HOT-FIRE TESTING OF 100 LB_F LOX/LCH_4 REACTION CONTROL ENGINE AT ALTITUDE CONDITIONS

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ABSTRACT

Liquid oxygen/liquid methane (LO_2/LCH_4) has recently been viewed as a potential green propulsion system for both the Altair ascent main engine (AME) and reaction control system (RCS). The Propulsion and Cryogenic Advanced Development Project (PCAD) has been tasked by NASA to develop these green propellant systems to enable safe and cost effective exploration missions. However, experience with LO_2/LCH_4 as a propellant combination is limited, so testing of these systems is critical to demonstrating reliable ignition and performance. A test program of a 100 lb_f reaction control engine (RCE) is underway at the Altitude Combustion Stand (ACS) of the NASA Glenn Research Center, with a focus on conducting tests at altitude conditions. These tests include a unique propellant conditioning feed system (PCFS) which allows for the inlet conditions of the propellant to be varied to test warm to sub-cooled liquid propellant temperatures. Engine performance, including thrust, c* and vacuum specific impulse ($I_{sp,vac}$) will be presented as a function of propellant temperature conditions. In general, the engine performed as expected, with higher performance at warmer propellant temperatures but better efficiency at lower propellant temperatures. Mixture ratio effects were inconclusive within the uncertainty bands of data, but qualitatively showed higher performance at lower ratios.

NOMENCLATURE

c* - Characteristic Exhaust Velocity [ft/sec (m/sec)]

I_{sp} – Specific Impulse [sec]

MR – Mixture Ratio (a.k.a O/F) [lb_m/s oxidizer / lb_m/s fuel (kg/s oxidizer / kg/s fuel)]

PCAV – Main Chamber Pressure [psia]

TOV/TFV – Temperature measured just upstream of Thruster Valves (Oxidizer/Fuel) [°R (K)]

vac – Vacuum condition

INTRODUCTION

To enable future exploration of the moon, Mars, and beyond next generation propellant system are being developed. With an emphasis on non-toxic, “green” propellants, LO_2/LCH_4 has risen to the forefront. Not only is there a potential for decreased vehicle mass, these propellants can also be produced on the moon and Mars using local resources. Prior work with these propellants is limited, so a goal of the current NASA Propulsion and Cryogenic Advanced Development (PCAD) project is to examine the feasibility and performance characteristics of these systems.[1, 2] In particular, there is interest in demonstrating repeatable and reliable ignition of the engine over a wide range of valve inlet temperatures (from liquid-liquid operation to gas-gas operation), especially at vacuum conditions.[3]

To facilitate this, a 100lb_f (445 N) LO_2/LCH_4 Reaction Control Engine (RCE) was developed by Aerojet.[3] Testing and analysis are being performed at the NASA Glenn Research Center’s Altitude Combustion Stand. The focus of testing is to determine the steady state and pulsed performance of the engine at simulated altitude conditions, over a wide range of valve inlet temperature conditions. A Propellant Conditioning Feed System (PCFS) developed by Sierra Lobo, Inc. (SLI) was used to control propellant conditions up to the RCE valves. This document will discuss the steady state (duration) testing that took place in November 2009 through February 2010 at NASA GRC. A total of 9 test cases were examined during the current test series.

RESULTS AND DISCUSSION

FACILITY OVERVIEW

Altitude testing of the RCE was conducted in the NASA GRC Altitude Combustion Stand (ACS) facility. This facility was originally part of the NASA Glenn (then NASA Lewis) RETF (B-Stand) facility, and was moved to its current location as part of the Cleveland-Hopkins Airport expansion in late 1990's/early 2000's. The relocated facility became active and operational during the summer of 2009.

The ACS facility is capable of accommodating up to a 2000 lb_f (8896 N) class engine and can simulate altitude conditions up to 130,000 ft. (39,624 m). The facility is able to test LO_2/GO_2/LH_2/GH_2/LCH_4 and RP propellants, up to chamber pressures of 1000 psia (6.8 MPa). A water spray cart and a water-cooled diffuser/multi-stage ejector system are used to condense the products of combustion and draw vacuum conditions inside the test capsule. A photograph of the spray cart and test capsule is shown in Figure 1.

Figure 1: ACS Facility Test Capsule, Spray Cart and Ejector Platform

Data acquisition is achieved through a National Instruments DIAdem™ module[4], and timing is controlled by a Modicon Quantum Programmable Logic Controller (PLC). The DIAdem™ module allows for real-time, streaming views of data at nominal sample rates of up to 1000 Hz and multiple computers in the control room allow for quick-view/post-test processing, providing researchers and engineers vital information to rapidly adjust test conditions and prepare for the next test.

PROPELLANT CONDITIONING FEED SYSTEM (PCFS)

Since the primary objective of the current test series was the performance characterization of the RCE at various propellant temperatures, a propellant condition system was developed to allow researchers to vary propellant inlet conditions to the engine in a controlled manner. The Propellant Conditioning Feed System (PCFS) was developed by Sierra Lobo, Inc. (SLI) and consists of two conditioning skids, one for oxygen and one for methane.[5] Each skid consists of a cryogenic bath, cryogenic heater, run tank, and vacuum jacketed lines leading to the test cell. The methane skid also incorporates a recirculation leg and cryogenic pump which allows the propellant to be re-circulated from the run tank, up to the thruster valves and back through the skid for additional temperature conditioning. A trace cooling system around the run lines allows for the system to be pre-chilled in order to maintain propellant conditions up to the engine. Both skids are capable of conditioning and maintaining propellant temperatures to +/- 5 °R. Each skid is independently controlled by a separate PLC and operator. A permissive signal is sent from each skid PLC to the facility PLC for a test ready condition. The PCFS was originally tested with the RCE in GRC's RCL Cell 32 during the summer of 2009, and then moved to ACS during the early fall of 2009. Photographs of the LO_2 and LCH_4 skids located outside of ACS are shown in Figure 2.

Figure 2: Photographs of LO_2 (left) and LCH_4 (right) PCFS Skids Outside of ACS

HARDWARE DESCRIPTION

The test article for this test series was the Aerojet designed 100 lb_f (445 N) RCE thruster.[3] For the tests described here, the injector and exciter were kept the same, and used a radiatively cooled, high-area ratio (altitude) nozzle made of columbium with an oxidation resistant coating. The altitude nozzle is a 45:1 area ratio, 80% bell nozzle with a 2.5″ L′. The injector is a split-triplet design, with fuel film cooling (FFC) ports and a barrier zone along the wall. A photograph of the engine with the altitude nozzle is shown in Figure 3.

The 100 lb_f (445 N) RCE was designed to meet the requirements specified in Table 1. Initial testing of the engine at both Aerojet and NASA White Sands Test Facility (WSTF) demonstrated that the engine could meet the requirements given.[3] The injector was designed with the igniter manifold system integrated into the main manifold, such that only one set of valves was needed for flow control, reducing system mass and volume. The injector assembly was fabricated from nickel and nickel alloy for oxygen compatibility.

Figure 4 shows a test article flow schematic for altitude tests in ACS. Class A Resistive Temperature Detector (RTD) temperature probes were used on all critical temperature (flow calculation) measurements. Initial tests did not include the turbine flow meter data; these were included in later tests. Flow measurement was calculated by non-cavitating venturi flow, and good agreement between venturi and turbine flow meter data was observed in the later test series. The thrust stand consisted of a tri-load cell arrangement which was summed for a total thrust measurement and the measured thrust was adjusted for a true vacuum thrust level. The thrust stand was calibrated at the start of each test day, as well at the end of the day. Early tests also did calibrations between each hot-fire test, and data showed little drift in the calibration constant for the load cells throughout the test day. Table 2 lists the various target performance parameters and flowrates for nominal operation. Nominal operation is defined as a mixture ratio (MR) of 2.5, main chamber pressure (measured at PCAV) of 175 psia (1.2 MPa), and a vacuum thrust level of 100 lb_f (445 N).

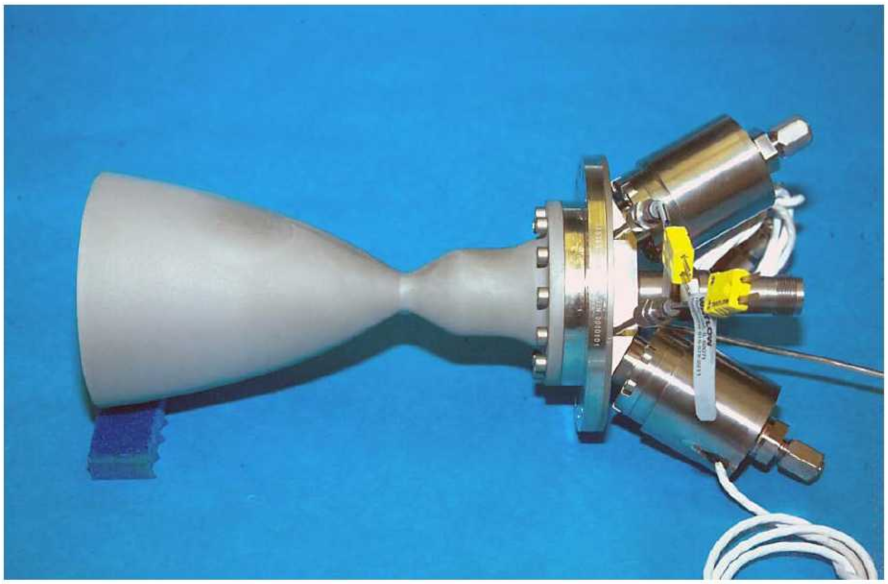

Figure 3: Photograph of the 100 lb_f RCE with Altitude Nozzle

Table 1: LO_2/LCH_4 RCE Engine Requirements

Parameter	Specification Value	Option 1 Test Results[3]
Thrust	100 lb_f (445 N)	84 to 115 lb_f (374 to 512 N)
Chamber Pressure	175 psia (1.2 MPa)	163 to 210 psia (1.12 to 1.44 MPa)
$I_{sp,vac}$ (80:1, 80% bell nozzle)	>317 sec	>317 sec*
Mixture Ratio	2.6 to 3.5	2.3 to 3.5
Nozzle Area Ratio, % Bell	7 x 16 in. envelope	80:1, 80% bell
EPW, Minimum	≤ 80 msec	40 msec
Impulse Bit, Minimum	4 lb_f-sec (17.8 N-sec)	≤ 4 lb_f-sec (≤ 17.8 N-sec)
Valve Life Cycle	25,000 cycles	55,000 Cryogenic Cycles
Mass	Minimize	11 lb_m (5 kg) w/ Flight-Type Components

*Demonstrated with sea-level chamber and extrapolated for an 80% bell, 80:1 exit ratio nozzle

Table 2: Target Nominal Flowrates for 100 lb_f (445 N) RCE

Overall Performance Characteristics	US	Metric
Vacuum thrust	100 lb_f	445 N
Nominal chamber pressure	175 psia	1.21 MPa
Vacuum Specific Impulse	317 sec	317 sec
Total Engine Flow (including igniter)		
Mixture ratio	2.48	2.48
Total flow rate	0.3159 lb_m/sec	0.1433 kg/sec
LO_2 flow rate	0.2251 lb_m/sec	0.1021 kg/sec
LCH_4 flow rate	0.0908 lb_m/sec	0.0412 kg/sec
Main Injector Flow (excluding igniter)		
Mixture ratio	2.50	2.50
Total flow rate	0.308 lb_m/sec	0.140 kg/sec
LO_2 flow rate	0.220 lb_m/sec	0.0998 kg/sec
LCH_4 flow rate	0.088 lb_m/sec	0.0399 kg/sec
Igniter Flow (~2.5% total flow)		
Mixture ratio	1.82	1.82
Total flow rate	0.0079 lb_m/sec	0.0036 kg/sec
LO_2 flow rate	0.0051 lb_m/sec	0.0023 kg/sec
LCH_4 flow rate	0.0028 lb_m/sec	0.0013 kg/sec

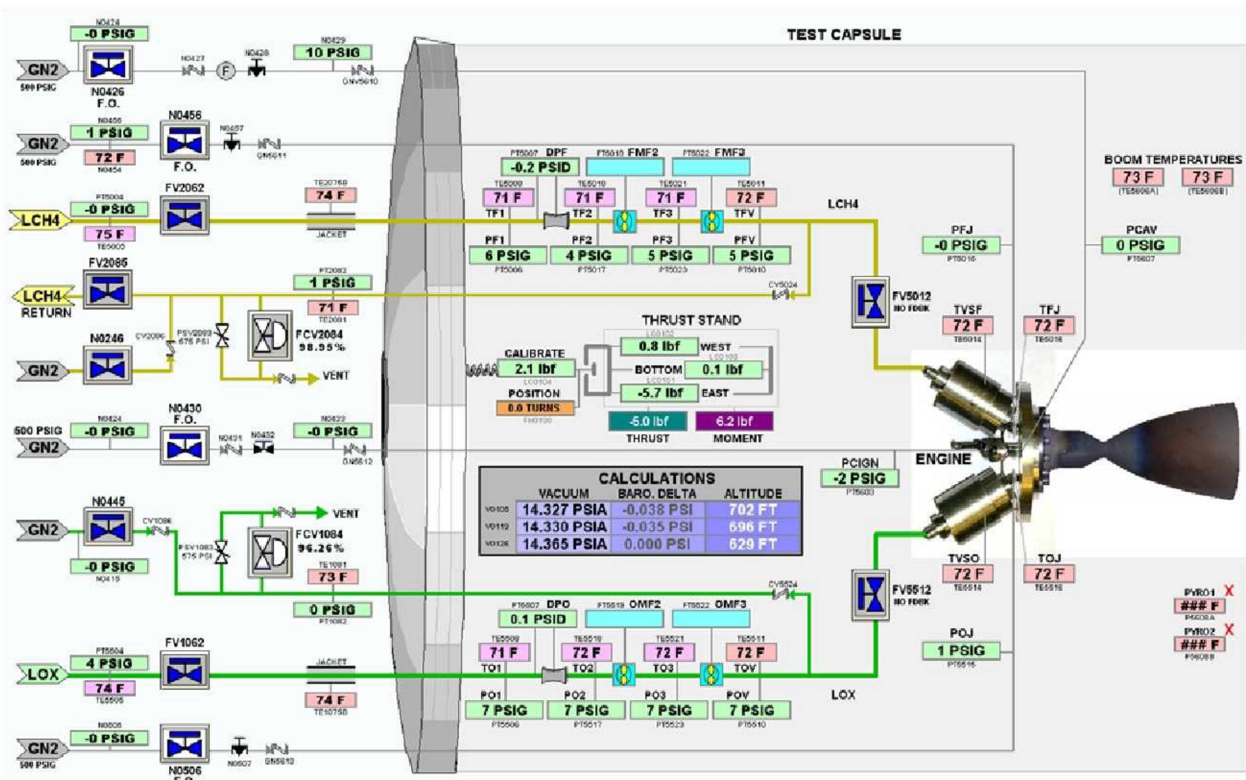

Figure 4: Test Article Flow Schematic (screenshot) for ACS Altitude Tests. Values are temperature (°F), pressure (psig) and load measurements (lbf) measured during testing.
(For the above screenshot, the engine was not operating, and the values are not meaningful)

TIMING

Timing for the engine at ACS is handled in four zones. Zone 1 consists of facility checks, drawing vacuum, and other related pre-fire checks. Zone 2 is the actual test period. Zone 3 consists of purges, valve closure, and other shut-down processes. In the event of an abort, the facility enters Zone 4, which is a timed quick shut down and system-safe process. For the purposes of this paper, all timing discussed is with respect to Zone 2, unless otherwise noted. The initial timing of the engine included a 10 ms LO_2 lead on startup, which was intended to bring both manifolds up to pressure at the same time. Cold propellant flow tests confirmed that a 10 ms LO_2 lead brought both manifolds up to pressure at the same time. A 35 psia (0.25 MPa) nitrogen gas purge was introduced through the igniter cavity pressure port (PCIGN) in order to ensure sufficient gas surrounded the spark plug gap to prevent corona discharge (Paschen) effects.[3, 6] The spark was timed to come on 20 ms before the LCH_4 flow, and remain on for 80 ms after both valves were open (a total duration of 0.100 sec.). Since the columbium chamber was identified as oxygen sensitive, a LCH_4 shutdown lag relative to oxygen shutdown of 40 ms was used in order to ensure a fuel-rich mixture upon shut down. Since the focus of these tests was on steady state performance, attempts to shorten and minimize the ignition/shutdown transients were deferred to later tests. Run time, "T", was defined from the time the methane valve opened until the oxygen valve was closed (both valves fully open). Thus, a "5 second run" was a run where both valves were open together for a total of 5 seconds. No attempt was made for a "pre-chill" (pre-flow of liquid oxygen to lower the injector hardware temperature) of the engine prior to start of test, however, the propellant lines were maintained at chilled conditions up to the thruster valves during and between runs. Since no pre-chill was conducted, the hardware often varied in initial temperature near ambient conditions. Figure 5 shows a basic timing sequence used for these series of tests. Unless otherwise noted, this timing was maintained for all tests described here.

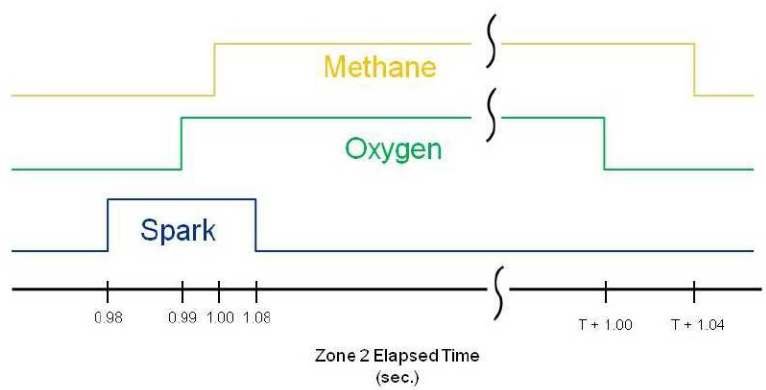

Figure 5: Basic Timing Sequence for Steady State Tests. "T" refers to planned test duration.

OVERALL TESTING PLAN

The purpose of testing the 100 lbf RCE at ACS was to determine performance behavior of the engine at altitude conditions. In order to determine this, testing was conducted at varying propellant inlet temperatures. Inlet temperature was defined as the temperature of the propellant measured just upstream of the thruster valve (TOV & TFV in Figure 4).

Table 3 lists the planned temperatures for ACS testing. Initial testing was conducted at nominal flowrates (for a MR = 2.5) and varying inlet temperature. Secondary testing investigated off-nominal mixture ratio excursions. Mixture ratio excursion tests examined performance at low MR (MR = 2.0) and high MR (MR = 3.0), also at varying propellant inlet temperatures.

Table 3: Planned Test Matrix of Inlet Temperatures

	Methane	Oxygen
Cold	170 °R (94 K)	163 °R (90 K)
Nominal	204 °R (113 K)	204 °R (113 K)
Warm	224 °R (124 K)	224 °R (124 K)

For the mixture ratio excursion tests, early testing focused upon maintaining chamber pressure while varying both propellant flowrates to achieve desired overall mixture ratio. However, during testing it was observed that the chamber experienced some minor spalling due to excessive heating at the high mixture ratio (MR = 3.0) case. This was attributed to insufficient wall cooling due to the reduced overall fuel flowrates (which is believed to significantly reduce FFC). After replacing the chamber with a spare one (same geometric dimensions), remaining MR testing focused upon maintaining nominal fuel flowrate to provide adequate FFC flowrates, adjusting MR by adjusting LO_2 flow only, and allowing main chamber pressure to vary accordingly.

Testing took place at GRC from November 2009 to February 2010, and all tests were completed at the ACS facility, with an average test capsule pressure of 0.21 psia (1447 Pa) over all tests. A total of 78 successful burns were completed. These tests varied in duration from 0.1 sec to 7.0 sec. While the PCFS were capable and successful at conditioning the propellants to the desired temperatures, maintaining those temperatures up to the thruster run valves proved difficult, especially for the cold propellant temperatures (LO_2 163 °R/ 90 K; LCH_4 170 °R/ 94 K). As a result, several non-ignition events and other transient behavior were observed. Due to the limited instrumentation available to diagnose ignition behavior, these non-ignition events will be further explored in testing later this year.

Figure 6 shows a plot of all valve inlet temperatures achieved for testing. The vertical axis is the methane inlet temperature and the horizontal axis is the LO_2 inlet temperature. The light grey box (surrounded by the dashed line) in the center of the figure represents the planned test range of the PCFS. The small, red boxes represent the ± 5 °R range at each temperature test point (represented by the "x" in the center of the red boxes). Each propellant condition is indicated by a specific color: red for warm/warm, green for nominal and blue for cold/cold. For the warm and nominal case, the majority of tests were within the range desired. However, the cold condition showed that there was some difficulty in reaching the appropriate methane temperature (LO_2 was just inside the desired range).

STEADY STATE (>3 SEC) TESTING

It was observed that it took approximate 3-5 seconds for the engine to fully chill in and flowrates to reach steady state conditions. After approximately 3 seconds, main chamber pressure, thrust, $I_{sp,vac}$ and methane flowrate had reached steady state conditions. LO_2 flowrate took a little longer, reaching steady state by 5 seconds into the run. Figure 7 shows the LO_2 and LCH_4 flow rates and chamber pressure with the steady state region highlighted. Although the LO_2 mass flowrate showed time varying behavior, and all other measurements showed steady behavior, the 3 sec. data was considered steady for the purposes of analysis.

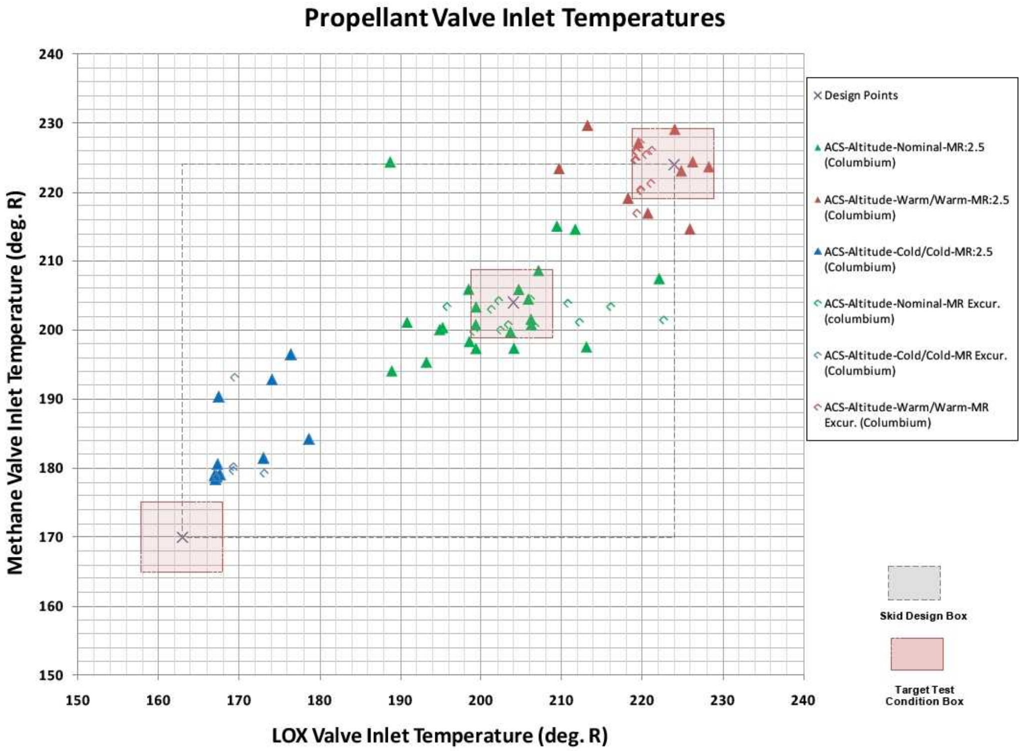

Figure 6: Plot of Valve Inlet Temperatures (TOV/TFV) for ACS Testing. MR Excursion tests include both high and low excursions.

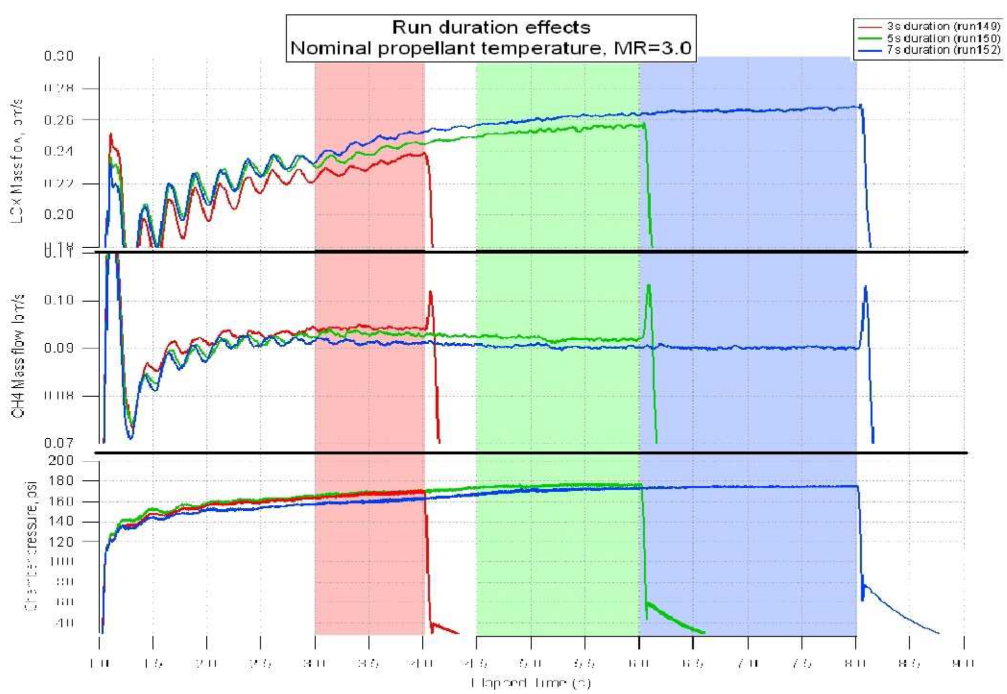

Figure 7: Plots to show steady state behavior of data. Top: Mass flowrate for LO_2. Middle: Mass Flowrate for LCH_4. Bottom: Mean Chamber Pressure. Red: 3 sec. test. Green: 5 sec. test. Blue: 7 sec. test. Highlighted regions show averaging period for each test duration shown.

Since there was considerable variation in the achievable experimental conditions (as can be seen in Figure 6), the data was classified using the ranges specified in Table 4. The acceptable range is shown along with the actual minimum and maximum values. Some steady state tests (approximately 11) fell outside these ranges and were not considered in the analysis.

Table 4: Table of Classification Ranges for Data Analysis.

Fuel Temperature Condition	Target Value	Acceptable Range	Actual Range
Cold	170 °R (94 K)	175 °R – 190 °R (97 K – 105 K)	178.4 °R – 184.2 °R
Nominal	204 °R (113 K)	194 °R – 214 °R (107 K – 119 K)	197.5 °R – 205.9 °R
Warm	224 °R (124 K)	215 °R – 234 °R (119 K – 130 K)	216.9 °R – 229.0 °R
Oxidizer Temperature Condition	**Target Value**	**Acceptable Range**	**Actual Range**
Cold	163 °R (90 K)	165 °R – 180 °R (92 K – 100 K)	166.9 °R – 178.6 °R
Nominal	204 °R (113 K)	194 °R – 214 °R (107 K – 119 K)	194.9 °R – 213.1 °R
Warm	224 °R (124 K)	215 °R – 234 °R (119 K – 130 K)	218.3 °R – 228.4 °R
Mixture Ratio Condition	**Target Value**	**Acceptable Range**	**Actual Range**
Low	2.0	1.75 – 2.24	1.82 – 2.24
Nominal	2.5	2.25 – 2.74	2.26 – 2.69
High	3.0	2.75 – 3.25	2.75 – 3.12

ERROR ANALYSIS

An uncertainty analysis[7, 8] was performed to qualify the performance results, in order to fully meet JANNAF reporting standards[9, 10]. While a fairly rigorous approach was used, refinement and improvement is ongoing and has not been completed for this publication. The parameters of interest in this analysis were vacuum I_{sp}, Thrust, c*, and mixture ratio. Bias errors were calculated for all diagnostics used in performance calculations including pressure transducers, thermocouples, RTDs, load cells, and turbine flow meters. Bias uncertainties were broken down into: calibration error, which addresses the conversion to engineering units, instrumentation error which addresses the performance of the diagnostic instrument, and data acquisition error which includes signal conditioners. Correlated bias uncertainties were neglected. The following equations show an example of bias error calculation for vacuum I_{sp}.

$$ISP_{vac} = \frac{F+P_{amb}A_e}{\dot{m}_o+\dot{m}_f} \quad \text{(Eq. 1)}$$

$$B_{ISP}^2 = \left(\frac{d\,ISP}{d\,F}\right)^2\left(tb_{F,inst}^2 + tb_{F,cali}^2\right) + \left(\frac{d\,ISP}{d\,\dot{m}_O}\right)^2\left(tb_{mo,,calc}^2\right) + \left(\frac{d\,ISP}{d\,\dot{m}_F}\right)^2\left(tb_{mF,,calc}^2\right) + \left(\frac{d\,ISP}{d\,P_{amb}}\right)^2\left(tb_{P,inst}^2 + tb_{P,cali}^2\right) + \left(\frac{d\,ISP}{d\,A_e}\right)^2\left(tb_{A,meas}^2 + tb_{A,therm}^2\right) \quad \text{(Eq. 2)}$$

where B_{Isp} is the total bias error for the vacuum I_{sp}, F is the thrust force, A_e is the exit area, P_{Amb} is the ambient pressure, $\dot{m}_O$ and $\dot{m}_F$ are the mass flows for oxidizer and fuel respectively, t is the factor from student's t-distribution, and b is the uncertainty value for each category of bias. The category is indicated by the subscripts which for this example are: instrument uncertainty (*inst*), calibration uncertainty (*cali*), thermal expansion effects (*therm*), and uncertainty in the measured area (*meas.*). Calculated uncertainty (*calc*) is a bias uncertainty that is calculated separately with its own set of diagnostics. For example, the mass flow rates are calculated from the subsonic venturi, so the associated bias uncertainty would include its own set of thermocouples, pressure transducers, etc. Approximately 15 instruments were used to get $I_{sp,vac}$, each with at least 2 uncertainty terms.

Since multiple tests were performed for all but one test condition, precision uncertainty was calculated using the multiple test method. Therefore, precision uncertainty was determined from the standard deviation of the calculated performance parameter itself, rather than a summation of all the diagnostic measurements. The following equation was used:

$$P_{ravg} = \frac{t}{\sqrt{N}}\left[\sum_{k=1}^{N}\frac{(r_k - ravg)^2}{N-1}\right]^{1/2} \quad \text{(Eq. 3)}$$

Where *P* is the precision uncertainty, *N* is the number of data points, *t* is student's t-distribution factor, *r* is the performance parameter from each test, and *ravg* is the average performance parameter for all tests at a given condition. Note that the term in brackets is the standard deviation.

All calculations are performed in a spreadsheet that is easily updated as new test data become available. Student's t-factor is currently 2, which assumes at least a 90% confidence interval for most test conditions. Table 5 shows the average values and associated uncertainties for the performance parameters at each test conditions. In the table "eff" refers to thrust chamber efficiency as defined from CEA calculations and JANNAF standards. [9, 11, 12]

	Mixture Ratio 2.25 to 2.74 (target = 2.5)											
	Cold/Cold				Nominal				Warm			
RESULTS	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max
Isp_ vacuum (sec)	299.10	4.12%	287.62	305.76	305.85	4.31%	299.94	310.65	323.97	4.71%	316.68	337.36
Thrust_vaccum (lbf)	106.15	3.95%	101.95	108.99	98.48	4.50%	94.20	102.62	103.59	4.59%	99.69	107.65
C* (ft/s)	5268.72	3.15%	5236.74	5308.76	5396.43	3.54%	5302.59	5493.57	5511.88	3.42%	5474.95	5570.29
MR	2.58	2.46%	2.42	2.69	2.48	3.45%	2.40	2.64	2.45	4.70%	2.26	2.55
# Data Points	8				7				5			
Isp_eff	87.89%				83.18%				84.23%			
C*_eff	90.47%				86.60%				85.37%			
	Mixture Ratio 1.75 to 2.24 (target = 2.0)											
	Cold/Cold				Nominal				Warm			
RESULTS	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max
Isp_ vacuum (sec)	303.54		303.54	303.54	306.17	4.45%	300.06	315.21	318.08	4.61%	311.01	334.33
Thrust_vaccum (lbf)	98.93		98.93	98.93	94.56	5.98%	85.47	104.17	96.91	6.16%	83.58	102.68
C* (ft/s)	5405.23		5405.23	5405.23	5419.50	3.48%	5328.62	5466.44	5521.41	3.52%	5471.30	5622.09
MR	2.18		2.18	2.18	2.05	3.62%	1.85	2.22	2.07	4.48%	1.82	2.24
# Data Points	1				10				8			
Isp_eff	93.07%				85.59%				82.93%			
C*_eff	95.38%				87.29%				83.93%			
	Mixture Ratio 2.75 to 3.25 (target = 3.0)											
	Cold/Cold				Nominal				Warm			
RESULTS	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max	Average	Uncertainty	Min	Max
Isp_ vacuum (sec)	292.11	5.05%	287.49	296.73	294.64	4.51%	288.80	302.23	316.83	4.17%	315.66	318.01
Thrust_vaccum (lbf)	106.65	3.74%	105.79	107.52	100.53	4.24%	97.65	102.95	101.47	3.96%	99.22	103.29
C* (ft/s)	5223.11	3.44%	5194.54	5251.68	5321.39	3.42%	5280.69	5365.70	5458.45	3.49%	5416.24	5477.64
MR	2.80	3.52%	2.75	2.84	2.85	3.91%	2.76	2.95	3.04	2.05%	3.00	3.12
# Data Points	2				4				4			
Isp_eff	85.62%				79.90%				82.05%			
C*_eff	90.28%				86.97%				87.12%			

Table 5: Summary Showing the Average Performance Parameter at Each Test Condition, and the Corresponding Uncertainties.

RESULTS

Figure 8 shows vacuum I_{sp} as a function of mixture ratio for the three temperature conditions. The average values for each condition is also plotted. Note that the line used to link the average values is not a curve fit, it is merely a simple line for aesthetic purposes. The error bars are the uncertainties for the average values (as opposed to individual values for each data point). The error bars on $I_{sp,vac}$ are ±4% and on mixture ratio are ±3% (actual values listed in Table 5). When looking at the nominal and cold temperature data, there is a decrease in performance as mixture ratio increases. However, this trend does not appear in the warm temperature $I_{sp,vac}$ data. It should be noted that the peak seen in the warm $I_{sp,vac}$ data may be due to two tests which, although appearing to be higher performing than the rest of the warm data, could not otherwise be ruled out as outliers based on test conditions. Overall, warm temperature propellants produced the highest performance, while the nominal and cold temperature propellants produced similar performance behavior. The uncertainty in $I_{sp,\,vac}$ (±4%) gives a ±12 sec band of uncertainty for the average overall performance of 305 sec. (min: 293 sec.; max: 317 sec.) Therefore, even between warm and cold propellant conditions, it is difficult to statistically say there is a significant difference in performance between the temperature conditions, as specific impulse by test case only varies from 292 sec. minimum average (nominal temperature; high MR) to 323 sec. maximum average (warm temperature; nominal MR). However, qualitatively there does appear to be an observable trend in the data. This is expected, as warmer propellants would have higher enthalpy and thus more energy available for performance. In addition, the density effects of propellant temperature could affect injector performance, since mixing characteristics would change with changes in density.

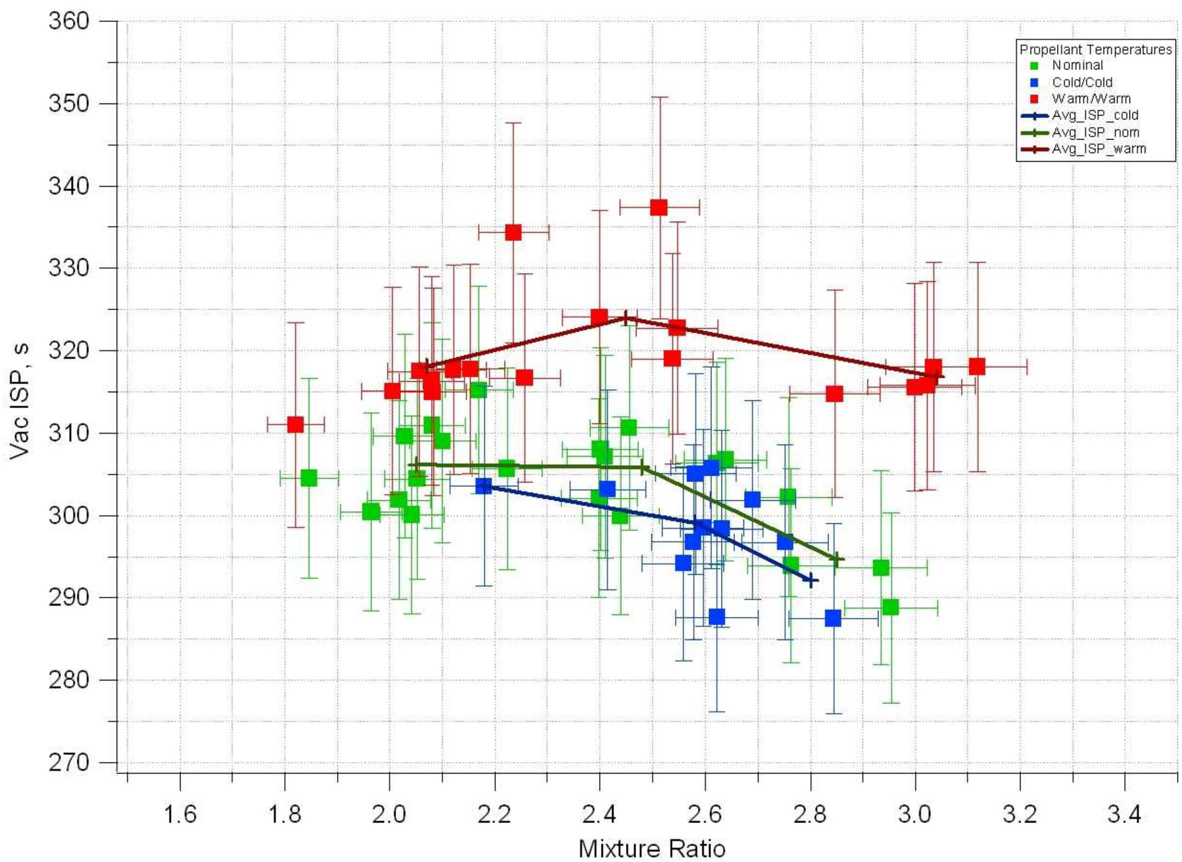

Figure 8: Vacuum I_{sp} versus mixture ratio with the propellant temperature conditions indicated by color (Red: Warm; Green: Nominal; Blue: Cold). Error bars represent the average uncertainty for $I_{sp,vac}$ (4%) and mixture ratio (3%). The lines are straight line interpolation between average values.

Figure 9 is vacuum I_{sp} plotted against propellant temperature (measured at the thruster valves, TOV and TFV). While the acceptance testing performed previously at Aerojet[3] demonstrated that the engine can perform >317 sec for vacuum I_{sp}, it should be noted that this value is derived from sea-level testing and extrapolated to vacuum conditions (80:1 bell nozzle), and those tests were conducted with warmer (~nominal) propellant conditions. Since colder propellant conditions demonstrated poorer relative performance, the lower temperature propellants are a potential cause for the observed average $I_{sp,vac}$ during this phase of testing being lower than the acceptance criteria. Considering the warm temperature propellant results for the present work, those average $I_{sp,vac}$ results were above the acceptance criteria (Ref: Table 5).

Figure 10 is an overview of performance with the efficiencies (left) and raw values (right) of both $I_{sp,vac}$ (top) and c* (bottom). Efficiencies are "thrust chamber" efficiencies as defined by the JANNAF standards.[9, 10] CEA[11, 12] was used to calculate the ideal performance for efficiency calculations. Ideal performance was calculated for a 45:1 exit ratio nozzle, and chamber contraction ratio of 5.85, with propellant temperature and enthalpy specified based upon actual propellant manifold conditions for a given test. As in Figure 8, the performance parameters are plotted against mixture ratio and grouped with respect to propellant temperature. Trends in c* are the same as those seen for $I_{sp,vac}$, with the exception that the warm propellant cases also show a decrease in c* as mixture ratio increases. The low MR, cold temperature propellant condition shows relatively good efficiency and performance. However, there is poor statistical resolution here since only a single test point is available here. The trade-off between efficiency and performance is otherwise clearly seen in the data, as the warmer propellants show greater relative performance (more relative energy available at warmer conditions) but lower relative efficiency. Here, the trend to improved efficiency with colder propellants could be related to mixing and vaporization efficiency, as the momentum ratio of the split triplet injector could be better optimized at colder conditions.

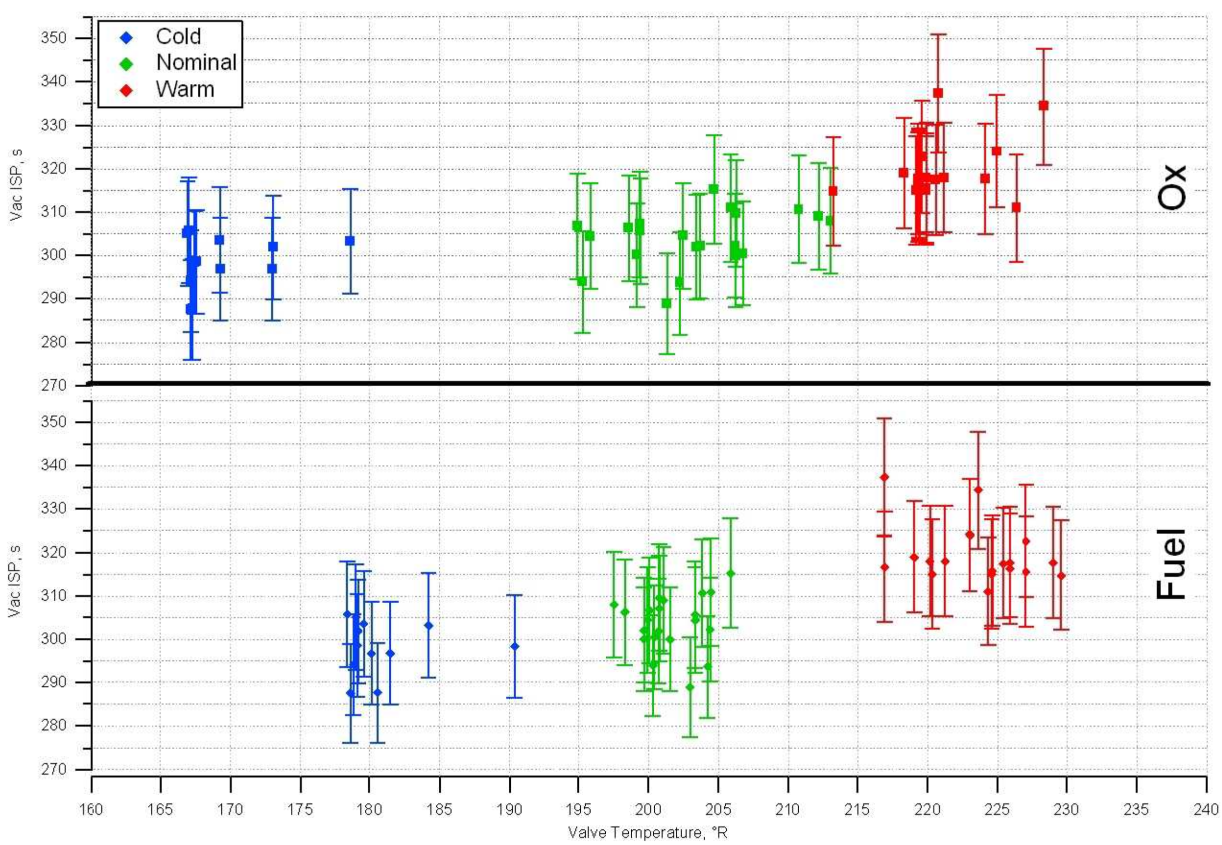

Figure 9: Plot of Vacuum I_{sp} versus propellant temperature (Top: LO_2; Bottom: LCH_4).

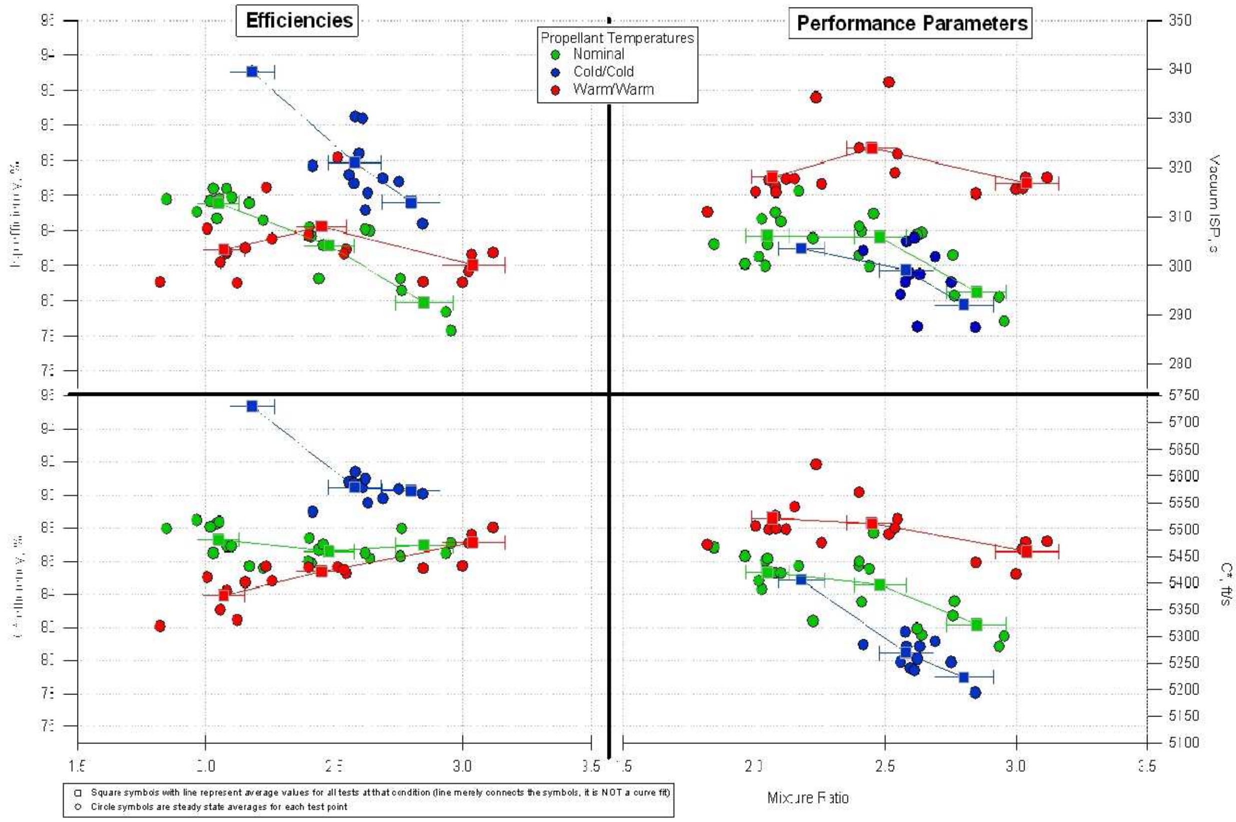

Figure 10: Plots of performance parameters. Left: Efficiencies; Right: Performance Parameters; Top: Vacuum Specific Impulse, $I_{sp,vac}$; Bottom: Characteristic Velocity, c*

SUMMARY AND CONCLUSIONS

A LO_2/LCH_4 100 lb_f (445 N) RCE was tested in an altitude combustion chamber at GRC. Steady state performance evaluations were conducted at three propellant temperature conditions and three mixture ratios. Measurement of pressure, temperature and thrust provided data to determine engine performance and efficiency.

The engine performed near to the programmatic goal of 317 sec for vacuum I_{sp} (ave. $I_{sp,vac}$ = 305 sec.), given the uncertainty present in the system (± 4%). While statistically it is difficult to come to any firm conclusion regarding performance degradation with either mixture ratio excursion or propellant inlet temperature, qualitatively there does appear to be a trend in behavior, with the colder propellants and higher mixture ratios showing the greatest decreases in performance. It is likely that warmer propellants have higher performance due to the increase in available enthalpy. While mixture ratio trends remained the same for efficiency (higher MR showed lower efficiency), colder propellants showed greater efficiency than warmer propellants. This is likely due to more optimized mixing and vaporization for the split triplet design at colder temperatures.

PLANNED FUTURE WORK

Additional testing of the RCE at NASA GRC is planned to investigate pulsed operation testing, ignition margin testing, and environmental (thermally conditioned hardware). Tests will continue to use the PCFS to study propellant temperature effects related to performance and hardware behavior.

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GLOSSARY

ACS – Altitude Combustion Stand

AME – Ascent Main Engine

CEA – Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications

ETDP – Exploration Technology Development Program

EPW – Electric Pulse Width

FFC – Fuel Film Cooling

GRC – NASA Glenn Research Center

JANNAF – Joint Army, Navy, NASA, Air Force

MR – Mixture Ratio (defined as mass flow oxidizer/mass flow fuel)

PCAD – Propulsion and Cryogenic Advanced Development Project

PCFS – Propellant Conditioning Feed System

PLC – Programmable Logic Controller

RCE – Reaction Control Engine

RCL – Research Combustion Lab

RCS – Reaction Control System

RETF – Rocket Engine Test Facility

SLI – Sierra Lobo, Inc.

WSTF – NASA White Sands Test Facility