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*Steven J. Schneider, Jeremy W. John, and Joseph G. Zoeckler*
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November 2007

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## Abstract

A workhorse liquid oxygen-liquid methane ($LOX/LCH_4$) rocket igniter was recently tested at NASA Glenn Research Center's (GRC) Research Combustion Laboratory (RCL). These tests were conducted in support of the Reaction Control Engine (RCE) development task of the Propulsion and Cryogenics Advanced Development (PCAD) project. The igniter was a GRC in-house design used to evaluate the ignition processes for $LOX/LCH_4$. The test matrix was developed to examine the flammability of $LOX/LCH_4$ over a range of oxidizer-to-fuel mixture ratios, both in the core fuel flow and total flow. In addition, testing also examined the durability of the hardware by accumulating ignition pulses. Over the course of testing, a total of 1402 individual ignition pulses were successfully demonstrated over the range of mixture ratios. Testing was halted after the failure of the ceramic in the igniter spark plug.

## Introduction

The impetus for the development of liquid oxygen liquid methane ($LOX/LCH_4$) Reaction Control Engines (RCE) is based on several factors. First, while hypergolic propellants for spacecraft attitude control have been used since the 1960s, their use results in high operating costs for manned vehicles. With the development of Orion CEV, plans are underway to mitigate the safety concerns and operating costs that burden these propellants by using nontoxic cryogenic propellants (ref. 1). In-situ resource utilization on a future trip to Mars then creates a driver for the use of $LOX/LCH_4$, which also has the potential for improved performance and reduced cost over the hypergolic propellants (ref. 2). One of the key technical challenges to the use of $LOX/LCH_4$ in reaction control systems (RCS) on these vehicles is the demonstration of reliable ignition over the potential operating envelope of RCS thrusters. These thrusters have a minimum impulse bit, widely varying duty cycles, and high cycle life requirements. Reliable and repeatable demonstrations of cold start and restart with heat soak-back are required to mitigate the risk. Recent demonstrations (ref. 3) of $LOX/LCH_4$ ignition with hardware designed for LOX/ethanol propellants shows that this propellant combination has some versatility. This paper describes a NASA designed igniter and the test facility in which it was tested, along with the test matrix the facility was able to provide to the igniter. Details of the test results are presented.

## Igniter Design and Fabrication

The GRC Workhorse Igniter was designed around a bluff body tipped sparkplug. A sketch of the design is shown in figure 1. The exciter unit for the spark plug delivered 200 sparks per second at 20 kV and 70 to 150 mJ. Design flow rates to the igniter were in the 44 N thrust class. The igniter was fed by three separate propellant feed lines, two for fuel and one for the oxidizer. The LOX was injected with four doublets in the annulus behind the bluff body on the spark plug tip. The LOX flows past the spark plug tip and is excited by the spark arcing across the flow to the wall. One fuel line injected fuel with four doublets just downstream of the spark plug tip to provide an oxidizer rich core flow, while the second fuel line injected fuel with tangential swirl to supply film cooling flow to the chamber wall and an overall fuel rich torch. The propellant injection locations are indicated on the sketch. The core mixture ratio design point was oxidizer rich at 20, while the overall design mixture ratio was fuel rich at 2.0. Both of these mixture ratios were selected due to their flame temperatures being compatible with conventional

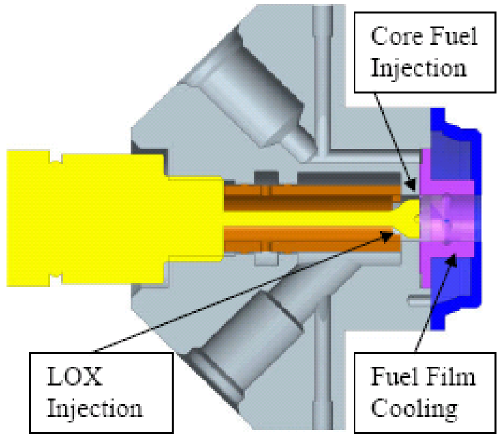


Figure 1.—Sketch of workhorse igniter.

materials. Propellant mass flow was controlled by cavitating venturis upstream of the thruster valves. Injector element pressure drops were low due to their design to accommodate the desired propellant mass flows as saturated vapor as well as saturated liquid.

## Facility Description

The LOX/$LCH_4$ ignition tests were conducted in Cell 21, which is part of the NASA GRC RCL. 4 A schematic of the Cell 21 layout for this series of tests is shown in figure 2 and a photograph is shown in figure 3 with the altitude can removed. All tests were conducted at a simulated altitude of 30 km, or approximately 1.4 kPa. The cell was upgraded for this series of tests to include capability for liquefying gaseous propellants in separate tanks by using a liquid nitrogen cooling system. Separate cooling lines were added to the feed system from the propellant tanks to the igniter inlet valves in order to maintain a liquid state. In addition, another cooling line was added around the igniter body in order to accelerate the chill in temperatures of the test hardware. Control relays cycled the liquid nitrogen on and off to each circuit based on the desired tank temperature (90 K for the liquid oxygen, 112 K for the liquid methane). Each bottle held approximately 2 liters of propellant. The liquid propellants were pressurized by the regulated gaseous propellant feed system with pressures up to 2760 kPa. Tests were initiated by a Quantum Programmable Logic Controller which triggered a Unison Controller to time the valves and spark.

The propellant flow rates were controlled using cavitating venturis. These venturis were installed in each of the three propellant feed lines upstream of the injector valves. Temperature and pressure measurements taken at the inlet of the venturis were used to calculate the mass flow rates, using equation (1) (ref. 5).

$$\dot{m} = C_d \left( \frac{\pi d_t^2}{4} \right) \sqrt{2 g \rho_l \left( p_i - p_v \right)} \tag{1}$$

The vapor pressure, $p_v$, and liquid density, $\rho_l$, were determined for each test run using physical property data as compiled by the National Institute of Standards and Technology (NIST). The discharge coefficients, $C_d$ for each of the venturi throat diameters, $d_t$, were determined using a water flow test rig. The tank pressures were set to achieve the desired flow rate. The inlet pressures, $p_i$, were very close to the tank pressures. In order to ensure accurate flow control, i.e., for the venturis to cavitate, the pressures at the exits of the venturis had to fall within a pressure range determined by equations (2) and (3) (ref. 5). The maximum downstream pressure above which the venturi doesn't cavitate is $Pe$,max. The minimum downstream pressure below which the venturi still cavitates, but the fluid doesn't fully recover to the liquid state is $Pe$,min. These values were calculated on each propellant leg for each test conducted over the course of this investigation.

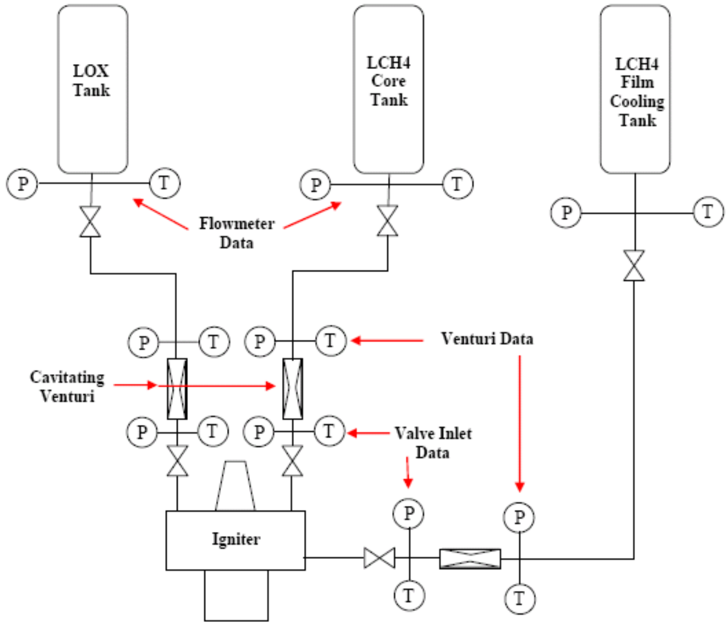


Figure 2.—Schematic of RCL-21 feed system.

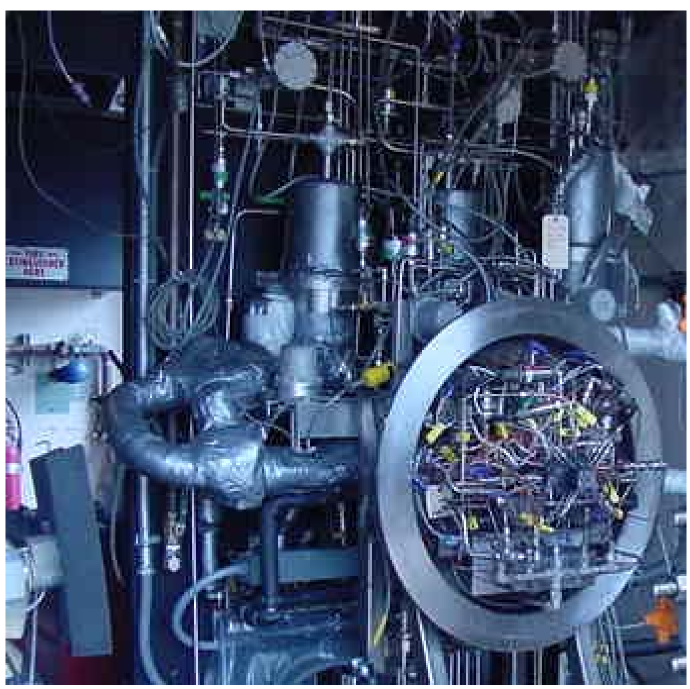

Figure 3.—Picture of facility with altitude can removed.

$$p_{e,\max} = p_v + C_d^2 \left(p_i - p_v\right)\left(1 - \left(\frac{d_t}{d_e}\right)^4\right) \tag{2}$$

$$p_{e,\min} = p_v + \frac{2C_d^2 \left(p_i - p_v\right)\left[\left(\frac{d_t}{d_e}\right)^2 - \left(\frac{d_t}{d_e}\right)^4\right]}{\left[1 - \left(\frac{d_t}{d_e}\right)^4\right]} \tag{3}$$

Turbine flowmeters were also installed into the system as an alternate means of measuring flow. They were located at the outlet of each tank as shown in figure 2 and had very low pressure drops. However, the flowmeters were found to be unreliable as they spun up, overshot the measurement, and didn't settle down within the pulse duration. Also, the range of the core methane turbine meter was too large to respond to the small flow rate most of the time. During the course of a test day extremely sluggish operation of the methane turbine meters was also noted and may have been caused by periodically freezing the methane by the liquid nitrogen cooling.

## Test Matrix/Objectives

Performance testing was addressed by the test matrix given in table 1. The tests were designed to evaluate the effect of mixture ratio variations, both in the core and overall, on the ignition process. A test ID was given for each test condition as shown in the table. The planned test matrix had the propellants condensed at ambient pressure using liquid nitrogen. Then, the nominal propellant temperatures in the tanks were 90 K for oxygen and 112 K for methane. Test condition (A) in the test ID was with a room temperature igniter body 298 K and test condition (D) was with a cold igniter body 112 K. Two different sets of cavitating venturis (I and II) were employed in an attempt to satisfy the exit pressure range requirements to ensure cavitation. Discharge coefficients ($C_d$) were determined as given in the table using a water flow rig with a beaker and a stopwatch. The planned variation of core mixture ratio was 15 to 31 and the planned overall mixture ratio range was 1.1 to 3.0. The planned overall flow rates were 12.18 to 17.15 g/sec.

The performance tests were conducted as single pulse tests with a 0.5 sec pulse duration in order to accommodate the slow response of the turbine meters. In these tests, the fuel valves were opened with a 0.02 sec lead over the oxidizer valves. Spark was initiated with the opening of the LOX valve and had a duration of 0.25 sec.

The test of durability of the research hardware by accumulating ignition pulses was conducted by running pulse trains of varying duration and duty cycle. The goal of this testing was to gage the expected lifetime of the igniter components and valves. In addition, the repeatability of the ignition pulses was examined. In order to put a large number of pulses on the igniter, a 0.25 sec long pulse was used. These tests also had the fuel valves opening with a 0.02 sec lead over the oxidizer valves and spark for 0.12 sec initiated with the opening of the LOX valve. During the bulk of these tests, the igniter was operated at a single propellant flow set point in multiple pulse strings at a 10 percent duty cycle.

TABLE 1.—PLANNED TEST MATRIX AND NUMBER
OF PULSES AT EACH CONDITION

| Test ID | # pulses | LOX Pres. | Fuel Core Pres. | Fuel Film Pres. | MR Core | MR | Mass Flow |
|---|---|---|---|---|---|---|---|
| | | kPa | kPa | kPa | | | g/sec |
| I-A1 | 7 | 2070 | 1900 | 1720 | 26 | 1.9 | 15.26 |
| I-A2 | 2 | 2070 | 1550 | 1720 | 29 | 1.9 | 15.23 |
| I-A4 | 8 | 1900 | 1900 | 1720 | 25 | 1.8 | 14.81 |
| I-A5 | 4 | 1900 | 1550 | 1720 | 27 | 1.8 | 14.77 |
| I-A6 | 2 | 1900 | 1210 | 1720 | 31 | 1.8 | 14.73 |
| I-A7 | 9 | 1720 | 1900 | 1720 | 24 | 1.7 | 14.36 |
| I-A8 | 6 | 1720 | 1550 | 1720 | 26 | 1.7 | 14.32 |
| I-A9 | 4 | 1720 | 1210 | 1720 | 30 | 1.7 | 14.27 |
| I-A10 | 4 | 1550 | 1900 | 1720 | 22 | 1.6 | 13.86 |
| I-A11 | 4 | 1380 | 1900 | 1720 | 21 | 1.5 | 13.31 |
| I-A12 | 2 | 1210 | 1900 | 1720 | 19 | 1.4 | 12.77 |
| I-A13 | 2 | 1040 | 1900 | 1720 | 18 | 1.3 | 12.18 |
| I-A14 | 2 | 1040 | 2240 | 1720 | 16 | 1.3 | 12.22 |
| I-A15 | 2 | 1040 | 2590 | 1720 | 15 | 1.3 | 12.25 |
| I-A16 | 2 | 1040 | 1210 | 1900 | 23 | 1.3 | 12.37 |
| I-A18 | 1 | 1040 | 1210 | 2240 | 23 | 1.2 | 12.87 |
| I-A20 | 1 | 1040 | 1210 | 2590 | 23 | 1.1 | 13.28 |
| I-A21 | 1 | 1040 | 2590 | 2590 | 15 | 1.1 | 13.43 |
| I-A22 | 7 | 2420 | 2420 | 2420 | 25 | 1.7 | 17.13 |
| I-A23 | 4 | 2420 | 2590 | 2420 | 24 | 1.7 | 17.15 |
| I-D1 | 1 | 2070 | 1900 | 1720 | 26 | 1.9 | 15.26 |
| I-D4 | 3 | 1900 | 1900 | 1720 | 25 | 1.8 | 14.81 |
| I-D5 | 2 | 1900 | 1550 | 1720 | 27 | 1.8 | 14.77 |
| I-D7 | 3 | 1720 | 1900 | 1720 | 24 | 1.7 | 14.36 |
| I-D8 | 4 | 1720 | 1550 | 1720 | 26 | 1.7 | 14.32 |
| I-D9 | 1 | 1720 | 1210 | 1720 | 30 | 1.7 | 14.27 |
| II-A1 | 2 | 2590 | 2240 | 2070 | 24 | 3 | 13.35 |
| II-A2 | 2 | 2590 | 2070 | 2070 | 25 | 3 | 13.34 |
| II-A3 | 2 | 2590 | 1900 | 2070 | 26 | 3 | 13.31 |
| II-A4 | 2 | 2420 | 2240 | 2070 | 23 | 2.9 | 12.99 |
| II-A5 | 2 | 2420 | 2070 | 2070 | 24 | 2.9 | 12.97 |
| II-A6 | 2 | 2420 | 1900 | 2070 | 25 | 2.9 | 12.95 |
| II-A7 | 2 | 2240 | 2240 | 2070 | 22 | 2.8 | 12.62 |
| II-A8 | 2 | 2240 | 2070 | 2070 | 23 | 2.8 | 12.61 |
| II-A9 | 2 | 2240 | 1900 | 2070 | 24 | 2.8 | 12.59 |
| II-A10 | 2 | 2070 | 1900 | 2070 | 23 | 2.7 | 12.22 |
| II-A22 | 2 | 2420 | 2420 | 2420 | 22 | 2.7 | 13.23 |
| II-A29 | 9 | 2070 | 2070 | 2070 | 22 | 2.7 | 12.25 |
| II-A30 | 3 | 2070 | 2240 | 2070 | 21 | 2.7 | 12.26 |
| II-A31 | 18 | 2070 | 2420 | 2070 | 20 | 2.7 | 12.28 |
| Total pulses | 140 | | | | | | |
| Notes: | | | | | | | |
| All: LOX Temp-90 K, LCH4 Temp-112 K | | | | | | | |
| A: Body of Igniter-298 K | | | | | | | |
| D: Body of Igniter-112 K | | | | | | | |
| I: Cavitating Venturis LOX-0.474 mm dia., Cd=0.839 | | | | | | | |
| LCH4 Core-0.147mm dia, Cd=0.584, LCH4 Film-0.419mm dia., Cd=0.962 | | | | | | | |
| II: Cavitating Venturis LOX-0.419 mm dia., Cd=0.962 | | | | | | | |
| LCH4 Core-0.417mm dia., Cd=0.584, LCH4 Film-0.366mm dia., Cd=0.677 | | | | | | | |

# Test Results

The igniter successfully lit over the entire range of propellant conditions in table 1 and was repeatable. The number of successful ignitions at each test condition is given in table 1. The total number of pulses put on the igniter in performance tests examining the ignition range of the propellants was 140 (approximately 10 percent of the total number of pulses). As mentioned above, single pulse, 0.5 sec burns were performed during this investigation of flammability range. A typical ignition pulse pressure trace is shown in figure 4. The data used to determine the propellant flow rates and propellant conditions were taken 0.46 sec into each pulse. This was done to allow the system to overcome the initial transients and reach a steady state condition.

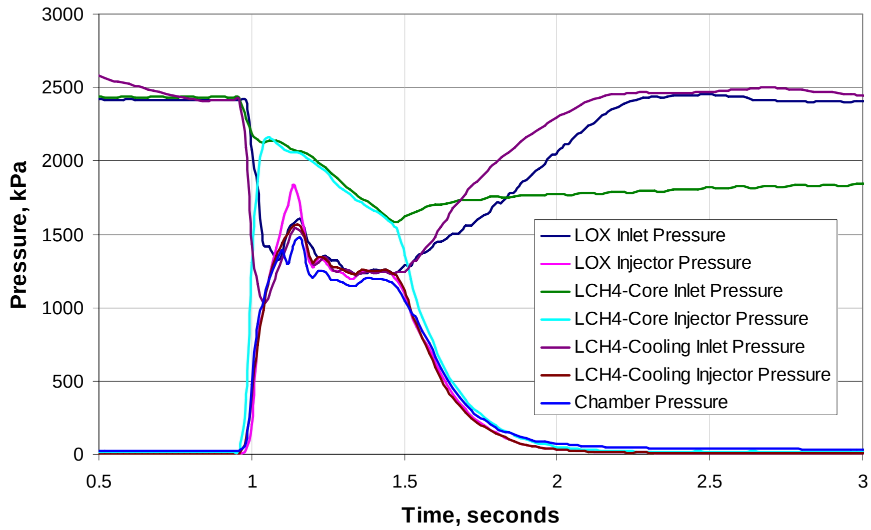


Figure 4.—Typical pressure trace from ignition pulse.

As shown in the feed-system schematic of figure 2, separate controls were used for the LOX, $LCH_4$ flow into the core, and $LCH_4$ flow into the film cooling injector. These allowed for a variation of the mixture ratio at the flame core as well as at the film cooling layer. Tabular results for selected run conditions are given in table 2.

TABLE 2.—MIXTURE RATIOS OBTAINED DURING SELECTED TESTS

| Run | Test profile | Target | | Calculated | | %Diff—Core | % Diff—Total |
|---|---|---|---|---|---|---|---|
| | | Core MR | Total MR | Core MR | Total MR | | |
| 278 | I-D9 | 30.0 | 1.7 | 29.29 | 1.68 | 2.36 | 1.28 |
| 302 | I-A1 | 26.0 | 1.9 | 33.46 | 1.67 | 28.70 | 12.28 |
| 308 | I-A5 | 27.0 | 1.8 | 26.77 | 1.35 | 0.84 | 25.15 |
| 311 | I-A7 | 24.0 | 1.7 | 21.81 | 1.35 | 9.11 | 20.35 |
| 313 | I-A8 | 26.0 | 1.7 | 31.92 | 1.40 | 22.78 | 17.74 |
| 315 | I-A10 | 22.0 | 1.6 | 19.36 | 1.08 | 11.99 | 32.32 |
| 317 | I-A11 | 21.0 | 1.5 | 20.33 | 1.10 | 3.21 | 26.79 |
| 363 | II-A1 | 24.0 | 3.0 | 31.91 | 1.80 | 32.95 | 40.06 |
| 365 | II-A2 | 25.0 | 3.0 | 30.75 | 1.85 | 23.00 | 38.23 |
| 367 | II-A3 | 26.0 | 3.0 | 29.55 | 1.88 | 13.64 | 37.27 |
| 369 | II-A4 | 23.0 | 2.9 | 28.69 | 1.66 | 24.72 | 42.73 |
| 371 | II-A5 | 24.0 | 2.9 | 27.94 | 1.62 | 16.41 | 43.99 |
| 373 | II-A6 | 25.0 | 2.9 | 33.06 | 1.80 | 32.23 | 38.08 |
| 375 | II-A7 | 22.0 | 2.8 | 22.76 | 1.55 | 3.44 | 44.74 |
| 377 | II-A8 | 23.0 | 2.8 | 26.41 | 1.72 | 14.82 | 38.40 |
| 379 | II-A9 | 24.0 | 2.8 | 28.04 | 1.70 | 16.83 | 39.27 |
| 381 | II-A10 | 23.0 | 2.7 | 29.19 | 1.69 | 26.90 | 37.55 |
| 383 | II-A22 | 22.0 | 2.7 | 27.06 | 1.66 | 22.98 | 38.39 |
| | | Average % Difference: | | | | 17.05 | 31.92 |

The amount of the fuel injected from the cooling injectors had the effect of keeping the overall mixture ratio between 1.08 and 1.88, while the core mixture ratio varied from 19.36 to 33.46 for these selected tests, as shown in table 2. The corresponding chamber pressures ranged from approximately 1040 to 1720 kPa and were set by the combustion efficiency and the mass flow through a 5.0 mm diameter throat on the igniter combustion chamber.

The calculated mixture ratios (MRs) had a significant deviation from the target values in some cases as shown in table 2. Equations (2) and (3) were used to check that the cavitating venturi exit criteria were satisfied. The results are presented in table 3.

TABLE 3.—VENTURI EXIT PRESSURE CAVITATION RANGE CALCULATION FOR SELECTED TESTS

| Run | LOX | | | | $LCH_4$—Core | | | |
|---|---|---|---|---|---|---|---|---|
| | Pcav_exit | Pe_max | Pe_min | Cavitating | Pcav_exit | Pe_max | *Pe*_min | Cavitating |
| | (kPa) | | | | (kPa) | | | |
| 278 | 1044 | 1293 | 292 | Yes | 329 | 833 | 442 | No |
| 302 | 1237 | 1591 | 526 | Yes | 1192 | 1251 | 1015 | Yes |
| 308 | 1282 | 1542 | 768 | Yes | 1248 | 735 | 632 | No |
| 311 | 1373 | 1405 | 705 | Yes | 1385 | 830 | 650 | No |
| 313 | 1170 | 1379 | 609 | Yes | 1175 | 1021 | 889 | No |
| 315 | 1238 | 1338 | 867 | Yes | 1358 | 983 | 804 | No |
| 317 | 1051 | 1161 | 665 | Yes | 1078 | 1103 | 828 | Yes |
| 363 | 1294 | 1471 | 563 | Yes | 1210 | 1297 | 960 | Yes |
| 365 | 1261 | 1372 | 382 | Yes | 1220 | 902 | 619 | No |
| 367 | 1127 | 1298 | 244 | Yes | 1068 | 552 | 276 | No |
| 369 | 1287 | 1331 | 444 | Yes | 1367 | 998 | 716 | No |
| 371 | 1167 | 1348 | 470 | Yes | 1167 | 833 | 537 | No |
| 373 | 1109 | 1245 | 285 | Yes | 1085 | 911 | 643 | No |
| 375 | 1316 | 1337 | 590 | Yes | 1398 | 726 | 425 | No |
| 377 | 1162 | 1130 | 212 | No | 1423 | 478 | 271 | No |
| 379 | 1082 | 1189 | 318 | Yes | 1101 | 633 | 377 | No |
| 381 | 992 | 1034 | 175 | Yes | 1038 | 747 | 463 | No |
| 383 | 1237 | 1205 | 204 | No | 1592 | 751 | 482 | No |

It was very difficult to satisfy these criteria for the core methane flow in the present test configuration. (Note: This situation can easily be alleviated in future tests by using smaller venturis or a larger throat at the exit of the igniter.) An investigation into the cause of the difference in the experimental MRs from the target MRs focused on the inlet conditions to the venturis given in table 4 and the mass flow calculation given by equation (1). The inlet temperature to the venturi affected mass flow in both the inlet density, $\rho_l$, and the vapor pressure, $p_v$ whereas, the controlling tank pressure only affects $p_i$ in equation (1). For example, the target density for the LOX was 1146 kg/m$^3$ and for the methane, it was 424 kg/m$^3$. The table shows that a temperature rise of 20 K results in a 10 percent change in these densities. A similar sensitivity of vapor pressure, $p_v$ can also be shown. Operationally, propellants were condensed at elevated pressure in the tanks rather than at ambient pressure, accounting for part of the propellant temperature rise in the system, the remaining being due to heat leaks.

TABLE 4.—CAVITATING VENTURI INLET CONDITIONS FOR SELECTED TESTS

| Run | $LCH_4$—Core | | | $LCH_4$—Film cooling | | | LOX | | |
|---|---|---|---|---|---|---|---|---|---|
| | Pcav_inlt | Tcav_inlt | rho | Pcav_inlt | Tcav_inlt | rho | Pcav_inlt | Tcav_inlt | rho |
| | kPa | K | kg/m$^3$ | kPa | K | kg/m$^3$ | kPa | K | kg/m$^3$ |
| 278 | 1475 | 131 | 391 | 1829 | 125 | 401 | 1717 | 101 | 1085 |
| 302 | 1887 | 147 | 363 | 1707 | 116 | 415 | 2043 | 108 | 1044 |
| 308 | 1549 | 137 | 380 | 1740 | 110 | 422 | 1869 | 114 | 1012 |
| 311 | 1912 | 138 | 379 | 1731 | 121 | 407 | 1701 | 112 | 1020 |
| 313 | 1564 | 144 | 368 | 1746 | 116 | 415 | 1706 | 110 | 1032 |
| 315 | 1881 | 142 | 372 | 1740 | 119 | 411 | 1537 | 115 | 1000 |
| 317 | 1887 | 143 | 371 | 1747 | 114 | 417 | 1372 | 111 | 1024 |
| 363 | 2198 | 146 | 365 | 2047 | 124 | 402 | 2549 | 109 | 1038 |
| 365 | 2051 | 137 | 381 | 2059 | 121 | 407 | 2548 | 104 | 1067 |
| 367 | 1880 | 124 | 402 | 2053 | 117 | 414 | 2549 | 99 | 1095 |
| 369 | 2195 | 140 | 376 | 2094 | 113 | 419 | 2385 | 106 | 1056 |
| 371 | 2036 | 135 | 385 | 2108 | 110 | 422 | 2390 | 107 | 1052 |
| 373 | 1871 | 138 | 379 | 2053 | 116 | 416 | 2386 | 101 | 1086 |
| 375 | 2284 | 131 | 392 | 2050 | 118 | 411 | 2224 | 110 | 1034 |
| 377 | 2031 | 124 | 403 | 2065 | 113 | 419 | 2220 | 97 | 1103 |
| 379 | 1854 | 129 | 395 | 2056 | 117 | 414 | 2223 | 102 | 1079 |
| 381 | 1871 | 132 | 389 | 2062 | 112 | 420 | 2054 | 95 | 1113 |
| 383 | 2382 | 133 | 388 | 2404 | 112 | 420 | 2394 | 97 | 1105 |

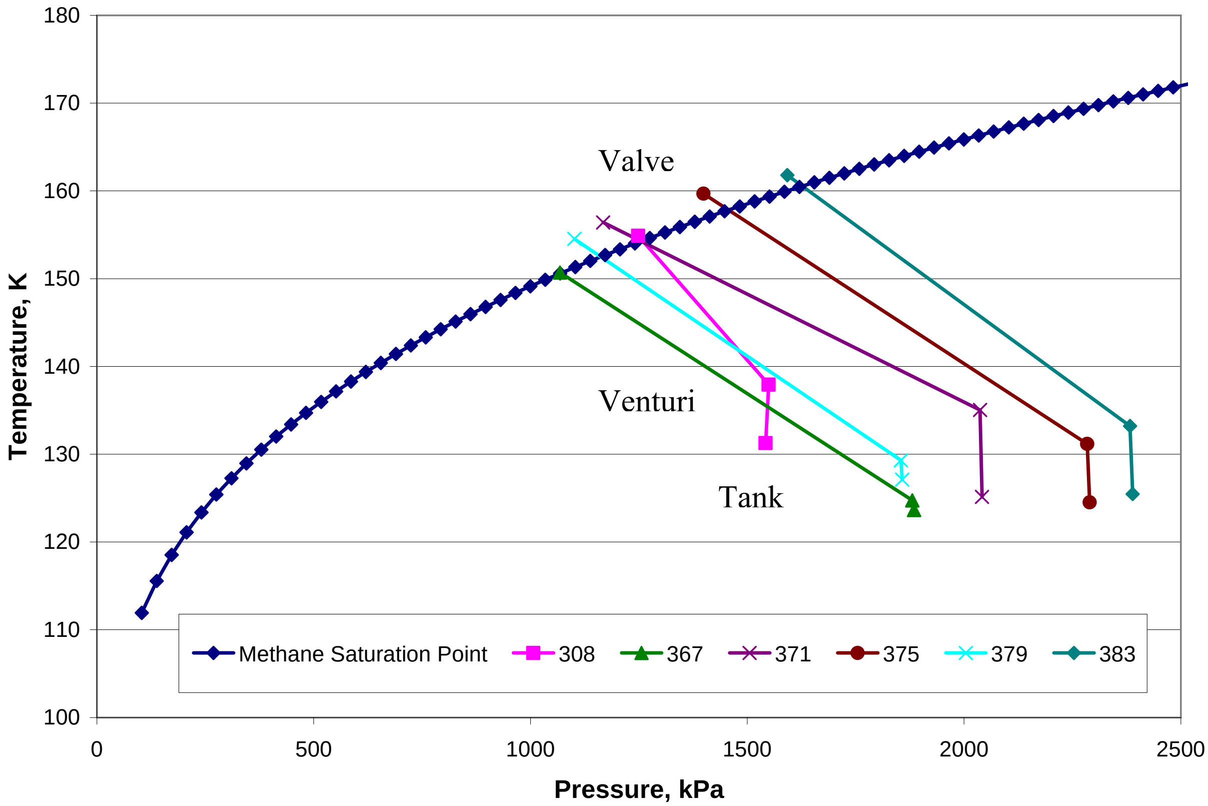


Figure 5.—Propellant phase in the core $LCH_4$ feed system during selected tests.

The effect of propellant storage temperature variation on the phase of the propellants entering the igniter manifold was also examined as part of this investigation. The goal was to determine whether the propellants remained in liquid phase going into the igniter manifolds. Figure 5 shows the temperature versus pressure relationship of the methane in the core leg as it moved through the feed system.

Three points are displayed for each test, i.e., tank condition, inlet to the cavitating venturi, and inlet to the igniter valves. The propellant saturation line is also shown. The data shows that during an ignition pulse, the methane remained in liquid state up to the igniter valves where it then began to transition to vapor. In all cases shown in figure 5, the fuel was injected into the chamber in gaseous phase, which was the expected state for ambient temperature hardware.

Non-ignitions were also experienced during the course of these tests. Approximately 5 percent of the total pulses attempted resulted in non-ignitions. This was usually due to clogging or freezing of the core fuel leg, but ignition was also observed in some cases with fuel transport from the cooling fuel into the fuel starved core. Ignition became increasingly difficult as the igniter hardware was pre-chilled to the methane temperature (approximately 112 K). Typically when these conditions were reached, the initial pulse would ignite nominally. However, subsequent pulses would not ignite. The cause of these non-ignition events was not fully examined during the course of the investigation. Many were the result of the valve’s sluggish operation or failure to open at 112 K. It was determined that additional testing at these conditions would be required in order to fully understand the non-ignition phenomena.

The durability of the igniter hardware was also examined during this investigation. A total of 1262 individual ignition pulses as described in table 5 were demonstrated in pulse trains to gage the expected lifetime of the test hardware.

TABLE 5.—DURABILITY TEST CONDITIONS AND NUMBER OF PULSES AT EACH CONDITION

| Test ID | Pulses | Pulse description |
|---|---|---|
| 1-A1 | 10 | 2 pulse trains, 5 pulses each at 0.5 sec on, 9.5 sec off (5% duty) |
| 1-A2 | 5 | 1 pulse trains, 5 pulses each at 0.5 sec on, 9.5 sec off (5% duty) |
| I-A22 | 15 | 3 pulse trains, 5 pulses each at 0.5 sec on, 9.5 sec off (5% duty) |
| II-A29 | 25 | 5 pulse trains, 5 pulses each at 0.5 sec on, 9.5 sec off (5% duty) |
| II-A29 | 25 | 5 pulse trains, 5 pulses each at 0.5 sec on, 4.5 sec off (10% duty) |
| II-A29 | 50 | 5 pulse trains, 10 pulses each at 0.5 sec on, 4.5 sec off (10% duty) |
| II-A29 | 10 | 2 pulse trains, 5 pulses each at 0.5 sec on, 2.0 sec off (20% duty) |
| II-A31 | 5 | 1 pulse train, 5 pulses each at 0.5 sec on, 4.5 sec off (10% duty) |
| II-A31 | 10 | 2 pulse trains, 5 pulses each at 0.25 sec on, 4.75 sec off (5% duty) |
| II-A31 | 20 | 2 pulse trains, 10 pulses each at 0.25 sec on, 4.75 sec off (5% duty) |
| II-A31 | 20 | 2 pulse trains, 10 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 240 | 12 pulse trains, 20 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 5 | 1 pulse train, 5 pulses each at 0.5 sec on, 4.5 sec off (10% duty) |
| II-A31 | 10 | 1 pulse train, 10 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 5 | 1 pulse train, 5 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 15 | 3 pulse trains, 5 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 80 | 8 pulse trains, 10 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 40 | 2 pulse trains, 20 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 8 | 1 pulse train, 8 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 27 | 3 pulse trains, 9 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 3 | 1 pulse train, 3 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 20 | 4 pulse trains, 5 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 20 | 2 pulse trains, 10 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 140 | 7 pulse trains, 20 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 3 | 1 pulse train, 3 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 13 | 1 pulse train, 13 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 8 | 1 pulse train, 8 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 15 | 1 pulse train, 15 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 20 | 4 pulse trains, 5 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 380 | 38 pulse trains, 10 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 2 | 1 pulse train, 2 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 9 | 1 pulse train, 9 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| II-A31 | 4 | 1 pulse train, 4 pulses each at 0.25 sec on, 2.25 sec off (10% duty) |
| | 1262 | Total Pulses |

For this phase of testing, the pulse duration was set to 0.25 sec. Most tests were at 10 percent duty cycle, but a small number were at duty cycles of 5 and 20 percent. The tank pressure and temperature setting were approximately the same through the tests. Most pulse trains had 5, 10, or 20 pulses.

Figure 6 shows the feed system pressures and chamber pressure for a typical 10 pulse string.

Figure 7 shows the propellant mass flow rates and chamber pressures measured at 0.23 sec into each pulse.

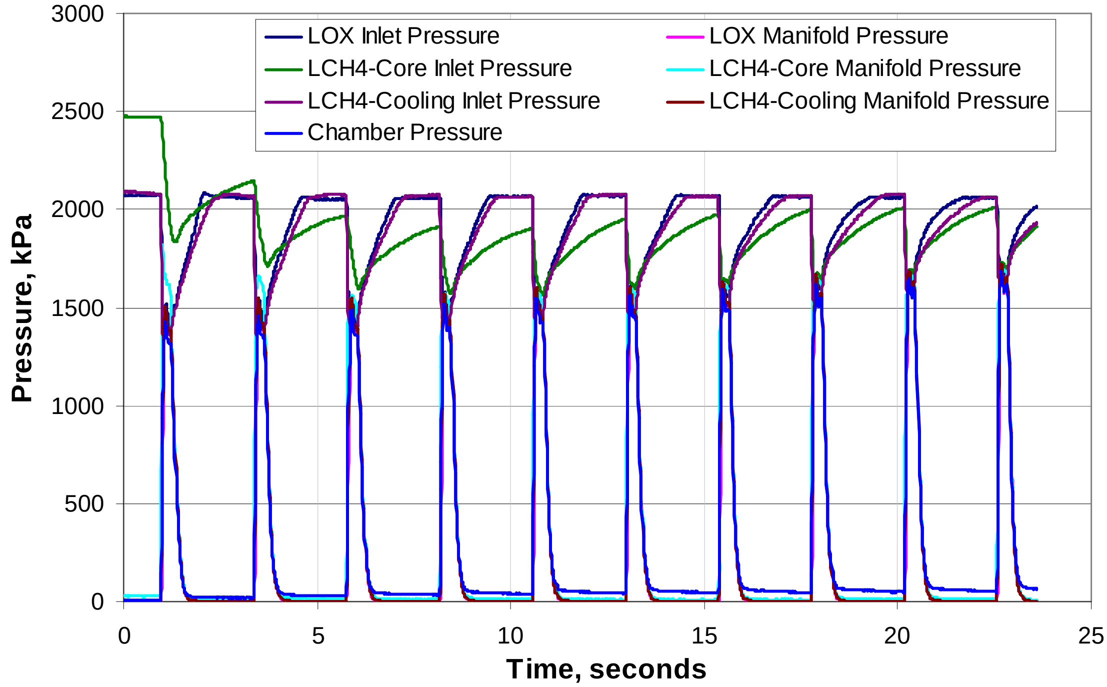


Figure 6.—Typical 10 pulse ignition train.

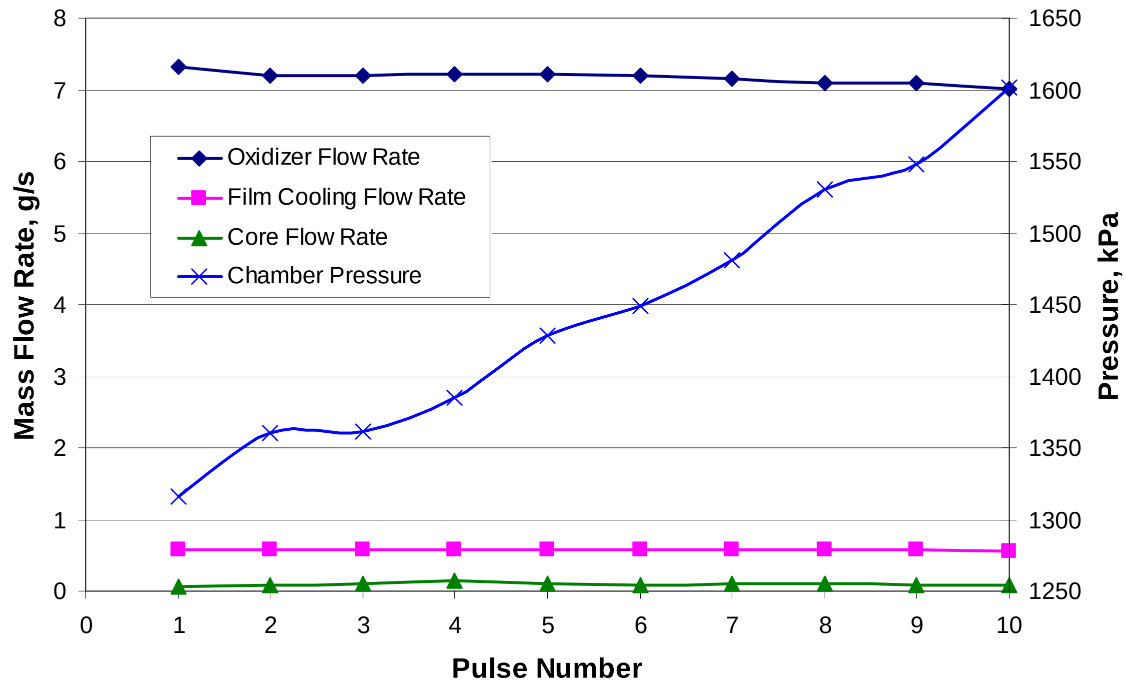


Figure 7.—Propellant flow rate and chamber pressure rise over the 10 pulse train.

Note that chamber pressure increases slightly in each subsequent pulse even though the measured mass flows are equal or slightly decreasing. This could be due to increased combustion efficiency in subsequent pulses or there could be some undetermined error in the mass flow measurement, since the downstream pressure criteria for cavitation were not being satisfied.

Table 6 shows the measured mass flow rates and calculated core and overall mixture ratios. Historical data (ref. 6) gives the lean premixed flammability limit of oxygen and methane as MR = 38. The core MR in table 1 is well beyond this oxidizer rich (fuel lean) combustion limit, while we're still successfully igniting. This implies some entrainment of the fuel film cooling into the core ignition zone.

TABLE 6.—PROPELLANT FLOW RATES AND MIXTURE RATIOS IN THE 10 PULSE TRAIN

| Pulse | Core flow rate | Film cooling flow rate | Oxidizer flow rate | Core MR | Total MR |
|---|---|---|---|---|---|
| | (g/s) | | | | |
| 1 | 0.055 | 0.587 | 7.325 | 132.66 | 11.41 |
| 2 | 0.074 | 0.581 | 7.204 | 97.79 | 11.00 |
| 3 | 0.101 | 0.585 | 7.187 | 70.82 | 10.47 |
| 4 | 0.153 | 0.584 | 7.219 | 47.31 | 9.80 |
| 5 | 0.101 | 0.577 | 7.212 | 71.55 | 10.64 |
| 6 | 0.075 | 0.582 | 7.187 | 95.98 | 10.95 |
| 7 | 0.096 | 0.577 | 7.156 | 74.36 | 10.64 |
| 8 | 0.093 | 0.579 | 7.103 | 76.18 | 10.57 |
| 9 | 0.076 | 0.579 | 7.096 | 93.66 | 10.84 |
| 10 | 0.074 | 0.565 | 7.011 | 94.91 | 10.98 |

The durability test showed that the igniter was capable of producing repeatable ignition pulses in over a hundred separate pulse trains. There were slight variations from run to run due to the fluctuation of the propellant pressure regulators. These variations resulted is slight differences in the propellant flow rates. This phase of testing was halted due to the failure of the ceramic insulation surrounding the base of the spark plug tip. The spark plug with the failed ceramic is shown in figure 8. An investigation into the cause of the failure is underway with a focus on its use in a cryogenic environment.

Figure 8.—Photograph of spark plug with failed ceramic.

## Conclusion

Over the course of testing, a total of 1402 individual ignition pulses were successfully demonstrated in ignitability and durability tests over the range of overall mixture ratios from 1.08 to 1.88. The corresponding chamber pressures ranged from approximately 1040 to 1720 kPa and were set by the combustion efficiency and the mass flow through a 5.0 mm diameter throat on the igniter combustion chamber. This pressure affected the accuracy of the cavitating venturi, but flow control was established as demonstrated by pulse repeatability. Approximately 10 percent of the total number of pulses was conducted to examine the ignition range for the ignitability portion of the testing. Ignition attempts with a cold-soak of the igniter were examined with mixed results. In the durability portion of the testing, 1262 pulses were accumulated mostly in trains of 10 pulses. Testing was halted after the failure of the ceramic in the igniter spark plug.

## References

1. Judd, D.C., et al, “Development Testing of a LOX/Methane Engine for In-Space Propulsion,” AIAA–2006–5079, July 2006.
2. Sullivan, T.A., Linne, D. L., Bryant, L., and Kennedy, K., “In-Situ-Produced Methane and Methane/Carbon Monoxide Mixtures for Return Propulsion from Mars,” *Journal of Propulsion and Power*, vol. 11, no. 5, 1995, pp. 1056–1062; also AIAA Paper 94–2846, June 1994.
3. Studak, J.W., Schneider, S.J., and Grimm, T., LOX/$LCH_4$ Ignition Results with Aerojet RCS LOX/Ethanol Igniter, JANNAF JPM, Denver, CO May 2007.
4. Kearns, K.A., “The Revamping of an Ignition Test Facility,” AIAA–2003–7362, January 2003.
5. Wright, R.S., and Olicker, S.D., “Cavitating Venturi for Flow Control,” *Chemical Engineering*, November 1956.
6. Strahle, W.C., *An Introduction to Combustion*, Gordon and Breach Science Publishers, 1993, Chap. 4.

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| **14. ABSTRACT** A workhorse liquid oxygen-liquid methane (LOX/$LCH_4$) rocket igniter was recently tested at NASA Glenn Research Center's (GRC) Research Combustion Laboratory (RCL). These tests were conducted in support of the Reaction Control Engine (RCE) development task of the Propulsion and Cryogenics Advanced Development (PCAD) project. The igniter was a GRC in-house design used to evaluate the ignition processes for LOX/$LCH_4$. The test matrix was developed to examine the flammability of LOX/$LCH_4$ over a range of oxidizer-to-fuel mixture ratios, both in the core fuel flow and total flow. In addition, testing also examined the durability of the hardware by accumulating ignition pulses. Over the course of testing, a total of 1402 individual ignition pulses were successfully demonstrated over the range of mixture ratios. Testing was halted after the failure of the ceramic in the igniter spark plug. | | | | | |
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