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ABSTRACT

Under NASA sponsorship, Northrop Grumman Space Technology (NGST) designed, built and tested two non-toxic, reaction control engines, one using liquid oxygen (LOX) and liquid hydrogen (LH_2) and the other using liquid oxygen and ethanol. This paper presents the design and testing of the LOX/LH_2 thruster. The two key enabling technologies are the coaxial liquid-on-liquid pintle injector and the fuel-cooling duct. The workhorse thruster was hot-fire tested at the NASA Marshall Space Flight Center Test Stand 500 in March and April of 2002. All tests were performed at sea-level conditions (see Figure 1). During the test program, 7 configurations were tested, including 2 combustion chambers, 3 LOX injector pintle tips, and 4 LH_2 injector settings. The operating conditions surveyed were 70 to 100% thrust levels, mixture ratios from 3.27 to 4.29, and LH_2 duct cooling from 18.0 to 25.5% fuel flow. The copper heat sink chamber was used for 16 burns, each burn lasting from 0.4 to 10 seconds, totaling 51.4 seconds, followed by Haynes chamber testing ranging from 0.9 to 120 seconds, totaling 300.9 seconds. The performance of the engine reached 95% C* efficiency. The temperature on the Haynes chamber remained well below established material limits, with the exception of one localized hot spot. These results demonstrate that both the coaxial liquid-on-liquid pintle injector design and fuel duct concepts are viable for the intended application. The thruster head-end design maintained cryogenic injection temperatures while firing, which validates the selected injector design approach for minimal heat soak-back. Also, off-nominal operation without adversely impacting the thermal response of the engine showed the robustness of the duct design, a key design feature for this application.

By injecting fuel into the duct, the throat temperatures are manageable, yet the split of fuel through the cooling duct does not

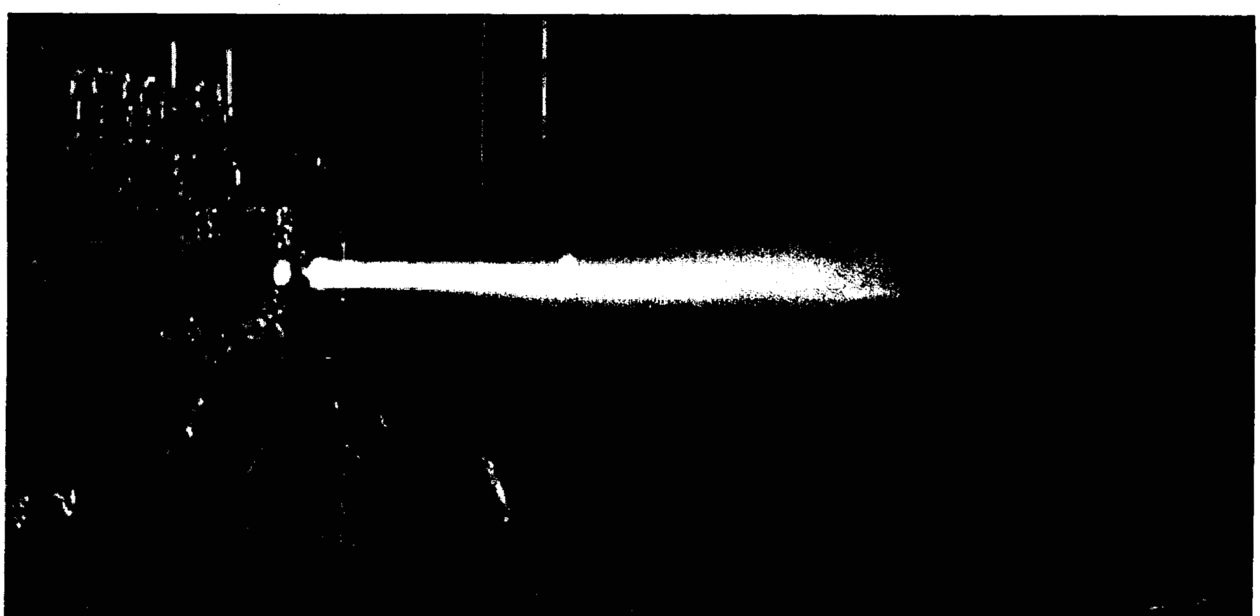

Figure 1: Burn P2280056

compromise the overall combustion efficiency, which indicates that, provided proper design refinement, such a concept could be applied to a high-performance version of the thruster.

INTRODUCTION

To support of the 2nd Generation Reusable Launch Vehicle (RLV) Program goals, this development project is structured to significantly increase the technology readiness of a high-performance reaction control system (RCS) thruster using non-toxic propellants for an operationally efficient and reusable auxiliary propulsion system (APS). This effort enables the development of an integrated primary/vernier thruster capable of providing dual-thrust levels of both 1000-lbf-class thrust and 25-lbf thrust. The intent is to reduce the risk associated with the development of an improved RCS flight design that meets the primary NASA objectives of improved safety and reliability while reducing systems operations and maintenance costs.

The current NASA Space Shuttle auxiliary propulsion system utilizes nitrogen tetroxide (NTO) and monomethylhydrazine (MMH), toxic and hypergolic propellants. These propellants require high levels of maintenance and precautions that contribute to cumbersome and costly launch operations, limiting access to space. By employing alternate non-toxic propellant combinations, the hazards and time required between missions can be significantly reduced, which in turn, will increase the efficiency and lower the cost in launch operations. The LOX/LH_2 propellant combination for auxiliary propulsion engine offers the advantage of a safe environment for maintenance, as well as the highest performance in conventional liquid rocket engines.

1000-LBF WORKHORSE ENGINE

The thruster design characteristics are summarized in Table 1. The thruster design includes a LOX-centered pintle injector, consisting of two rows of slots that create radial spokes into the chamber. The main fuel injection consists of a continuous sheet of LH_2 originating upstream of the LOX pintle tip.

Table 1: Thruster Design Characteristics

Thrust	1000 lbf
Mixture Ratio	4.0
Inlet Pressure	350 psia
Chamber Pressure	185 psia
Specific Impulse	350 lbf-sec//lbm
Total Flowrate	2.7 lbm/s
Duct Flowrate	20% of Fuel Flowrate
LOX Inlet Temperature	115 – 164 – 204 °R
LH2 Inlet Temperature	27 – 37 – 55 °R

The two propellant streams impinge and create the spray pattern that determines combustion efficiency and thermal conditions on the chamber walls. Another design feature of this thruster is the fuel wall cooling through a duct lining the inner wall of the chamber barrel section. The duct has cooling passages that discharge in the convergent section of the chamber creating a secondary fuel injection point. The variation in the amount of LH_2 used for the duct allows for adjustments in the cooling capacity for the thruster.

PROGRAM OBJECTIVES

The primary objective of this program was to establish and prove the LOX/LH_2 RCS thruster design concept through performance and thermal characterization tests at sea-level conditions. The test scope involved a screening of injector configurations over a range of operating conditions in short durations, using a copper heat-sink chamber. The performance (characteristic velocity, C*) and thermal trends were to drive the selection of the injector configurations for optimum performance compatible with acceptable heat loads for the Haynes chamber. With the pre-selected injector configurations, and a Haynes chamber, long duration burns were planned for evaluation of steady-state performance and thermal stability.

The secondary objective of this test program was to determine the feasibility of using fuel duct cooling to maintain the temperatures of the Haynes®188 thrust chamber within hardware limits set at 1850°F. The total fuel injected into the thruster is split between the main injector and the duct that is lined along the inside wall of the barrel section in the chamber. A range of 18.0 to 25.5% of total fuel flow was used for cooling, which varied the propellant mixture ratio inside the core zone of the combustion chamber, given the same total mixture ratio. This parametric study of percent fuel flow to duct determines the cooling effectiveness and the impact on overall performance.

The test program success criteria were as follows:

- Characteristic Velocity > 85% C* Efficiency
- Maximum Chamber Wall Temperature $\leq$ 1850 °F

TEST ENGINE DESCRIPTION

The 1,000-lbf-thrust test engine shown in Figure 2 has a building-block design approach and consists of a main stage injector, a cooling duct, a thrust chamber, and a torch igniter assembly. The injector configuration for this thruster is the oxidizer-centered coaxial pintle design, based on successful hot fire test demonstration with direct liquid-on-liquid injection of low-enthalpy hydrogen and oxygen propellants. The capability to easily change the injector body configuration to adjust injection parameters enables extensive development testing to be accomplished with minimum amount of hardware.

The oxidizer enters the injector through a centrally located passage and flows axially through a central pintle, where it is turned to uniform radial flow by the pintle tip's internal contoured surface. The oxidizer is then metered into the combustion chamber by a series of slots machined into the replaceable injector element. Thus the oxidizer is metered as radial spokes as illustrated by water flow test picture in Figure 3. Each extension sleeve has a primary row of slots, followed downstream by a row of secondary slots. The candidate pintle injectors have different slot geometries and replacing the threaded injector tip easily performs oxidizer injection geometry changes.

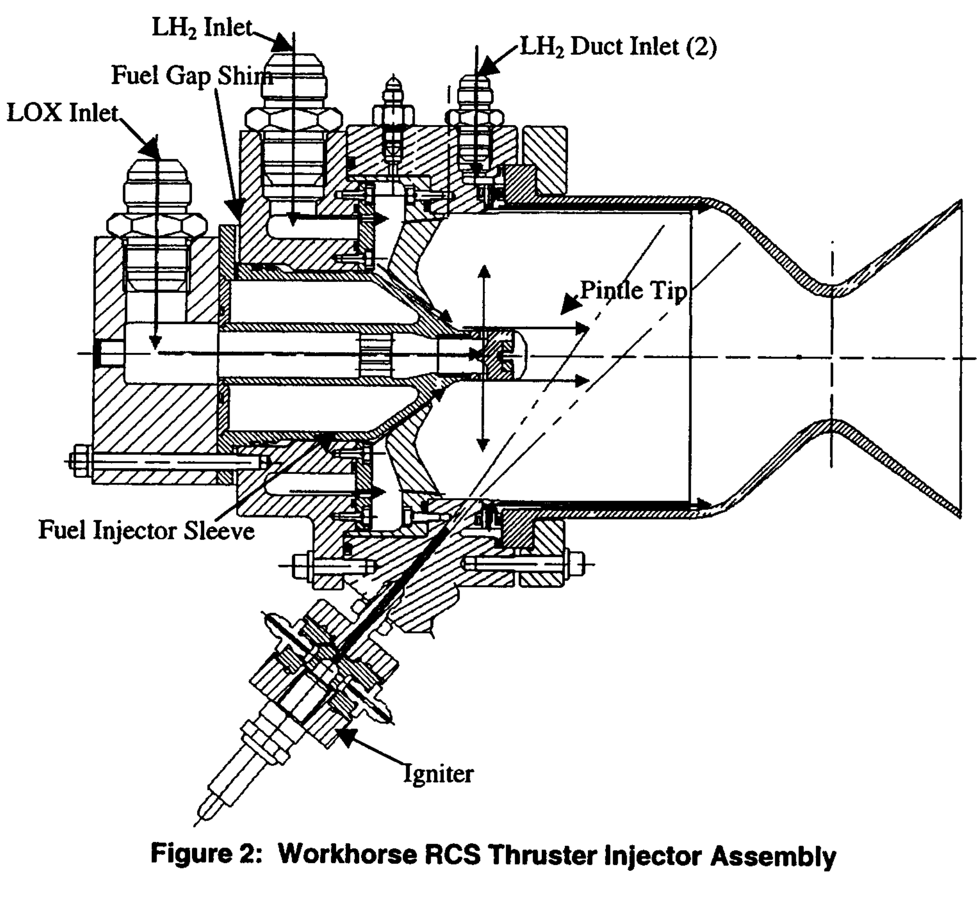

Figure 2: Workhorse RCS Thruster Injector Assembly

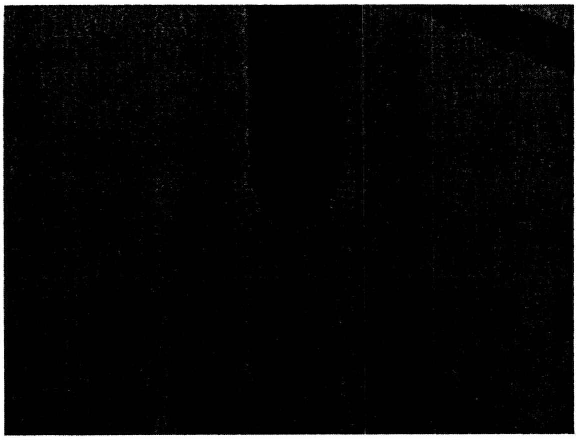

Figure 3: LOX Injector Flow Pattern

The fuel is fed through outer flow passages of the injector assembly into a circumferential annulus – formed between the injector body "snout" and the central injector element – which meters the flow into the combustion chamber. The fuel exits the injector as an axially flowing annular sheet that arrives at the oxidizer impingement point with a circumferentially uniform velocity as illustrated by Figure 4. The fuel injector metering area can easily be adjusted by changing the position of the sleeve with respect to the body. Replacing accessibly located shims changes the fuel annular passage and adjusts the fuel injection velocity.

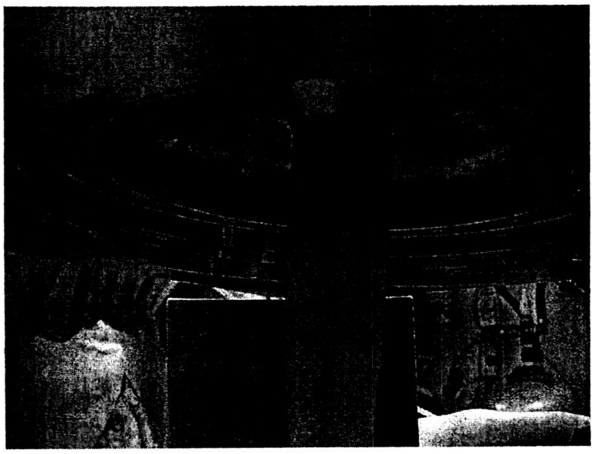

Figure 4: Main LH_2 Flow Pattern

The fuel is also fed through a fuel-cooling duct that lines the inner wall of the combustion chamber as illustrated in Figures 5 and 6. The duct has 180 cooling channels that extend along the chamber barrel section. It is designed to keep the chamber wall cool while LH_2 is passing through the duct and perform curtain cooling along the converging and throat sections of the nozzle upon discharge into the chamber.

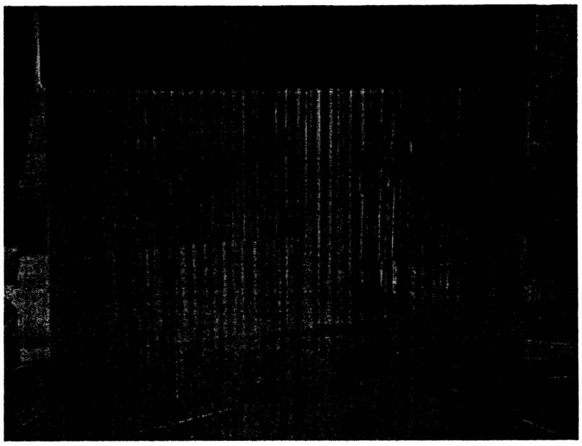

Figure 5 Fuel-Cooling Duct

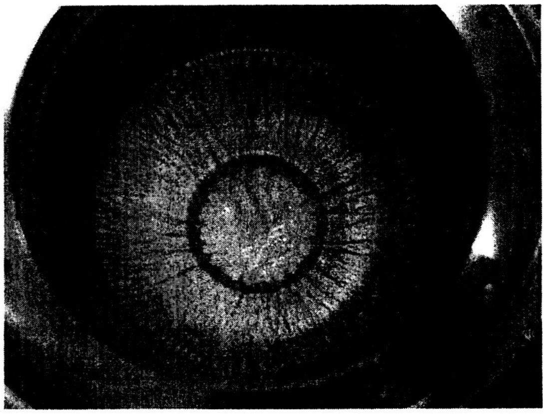

Figure 6: Duct Cooling – Chamber View

A GO2/GH2 igniter installed in the injector body as shown in Figure 7 performs the workhorse thruster main stage ignition. This spark-initiated torch igniter design has been successfully used for many years at various scales to provide ignition for rocket engine research hardware. The igniter assembly consists of an upper-flange containing a surface gap spark plug, a combustion chamber into which the hydrogen-oxygen propellants are introduced, the torch injection tube that directs the hot gases towards the main-stage chamber, and a port, which accepts the torch tube. Gaseous hydrogen and gaseous oxygen are introduced into the igniter chamber through two orifices and ignited using a spark plug. A third port in the chamber is used to monitor the igniter combustion pressure. Approximately 13% of the GH_2 flow rate enters the igniter chamber and the remaining 87% flows down along the outside of the torch injection tube to provide cooling. The oxygen flow makes the igniter chamber O/F equal to ~40. At the end of the torch tube, the cooling hydrogen mixes with the igniter chamber products of combustion and the resulting O/F is 5.

Spark energy for the igniter is supplied by a exciter box mounted on the side of the test stand. The high voltage from the exciter is supplied to the spark plug through a shielded ignition wire. The exciter operates on a 28 Vdc input and provides 100 sparks per second to the spark plug.

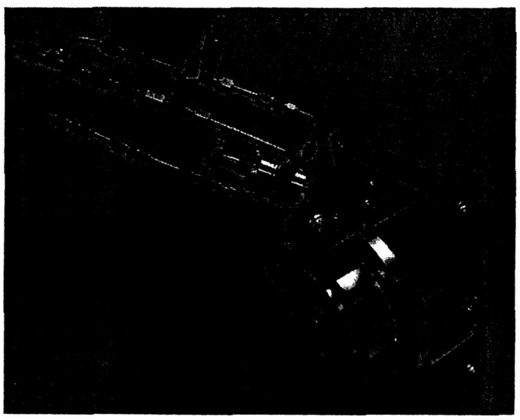

Figure 7: Igniter Assembly

The initial, injector-screening hot fire testing was performed in the workhorse heat sink copper chamber configuration as shown in Figure 8. The exit nozzle area ratio is 4:1. This robust engine configuration is capable of withstanding large pressure excursions and abnormal heat fluxes but is limited in run duration. On-times using this chamber were therefore, less than 10 seconds. Thermocouple probes were embedded in the copper chamber to assess heat fluxes.

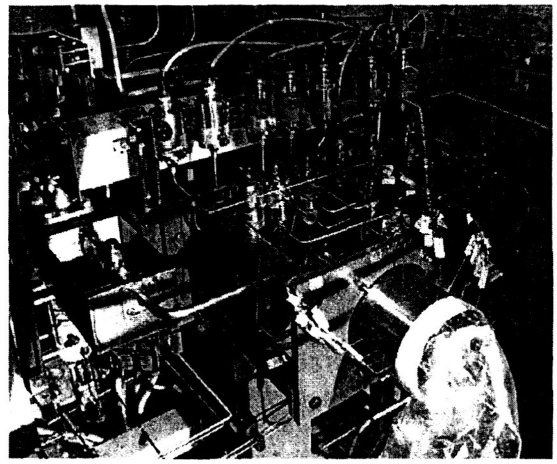

Figure 8: Copper Chamber Configuration Installed at MSFC TS500

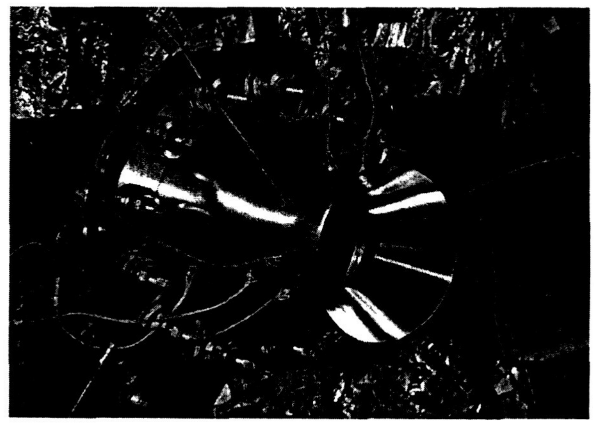

Figure 9: Haynes Chamber Configuration Installed at MSFC TS500

After the injector screening, the flight-like combustion chamber was installed onto the thruster as illustrated in Figure 9. This flanged bolt-up sea level thrust chamber (area ratio 4:1) was fabricated entirely from spin-formed Haynes® 188, which requires no oxidation resistant coatings. This material maintains adequate high-strength, oxidation and hydrogen embrittlement resistance, thus enabling the chamber to withstand steady-state operation at elevated temperatures. Control of chamber wall temperature is critical to engine performance and life. Multiple tacked-on thermocouples were installed to map the thermal response of the chamber wall during firing.

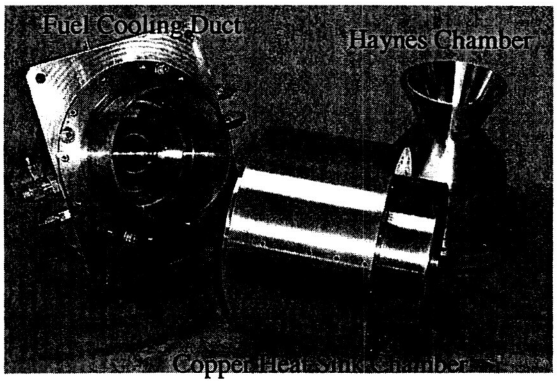

Figure 10 Chamber Components

A freestanding duct, shown attached to injector body in Figure 10, was inserted between the combustion gases and chamber barrel wall to maintain acceptable temperatures. Approximately 20% of the fuel flow is directed through cooling channels within the duct, and then injected onto the convergent section of the combustion chamber.

TEST FACILITY

This test program was conducted at NASA Marshall Space Flight Center. The NGST LOX/LH_2 workhorse thruster was horizontally mounted as an integrated assembly at the TS500 facility. TS500 is equipped for sea level testing with cryogenic propellant capability. The pressure system consisted of remotely operated valves, electro/hydraulic controllable pressurization system, and dry helium and nitrogen purge systems.

Propellant Interface Conditions

A liquid nitrogen (LN_2) bath was used for chilling of the LOX inlet feed line. All main injector LH_2 lines were insulated up to the test article inlet. The duct LH_2 lines were insulated up to the fire valve. The thruster was pre-chilled prior to each burn to ensure liquid propellant injection. The pre-chill operations can be summarized as follows:

1. Flow LH_2 through the fuel injector and duct while running a gaseous helium (GHe) purge through the LOX injector passages.
2. Flow LH_2 through the fuel injector while running the GHe purge through the LOX injector until target hardware temperatures are achieved. Flow momentarily through the duct just prior to firing.
3. Flow LH_2 through the fuel injector while purging the LOX injector with approximately until target hardware temperatures are achieved.

Table 2 summarizes the propellant inlet conditions for the test engine.

Table 2: Propellant Inlet Conditions

Propellant	Flowrate	Inlet Pressure	Inlet Temperature
Liquid Oxygen	1.40 to 2.14 lbm/s	200 to 350 psia	164 to 204°R
Liquid Hydrogen	0.42 to 0.62 lbm/s		
-Injector	0.32 to 0.47 lbm/s	240 to 370 psia	$\leq$55°R
-Cooling Duct	0.10 to 0.15 lbm/s	200 to 260 psia	~ 80°R

The actual inlet temperatures measured are shown in Figures 11 and 12. Due to the chill down procedure through the main fuel injector, the test article was sufficiently cold to ensure liquid injection for both propellants at the beginning of each burn. The duct propellant lines, however, were not chilled, and were not insulated. Therefore, the duct inlet temperatures were in a transient to steady state temperatures for approximately the first 10 seconds of every burn.

Facility and Engine Instrumentation

All pressure transducers for the thruster were strain gauge type transducers. Cryogenic fluid transducers were mounted with the transducer above the sense port. The thruster chamber pressure (P_c) measurement port is located at the injector head end. Stainless steel tubing (1/8" OD, 0.020" wall thickness, $\leq$ 12" length) was connected from the injector head end to the redundant pressure transducers. The majority of the temperature measurements on the thruster were measured with Type K thermocouple wires and probes. 0.030" K-Type thermocouples were welded on to the injector head and the Haynes chamber. The copper chamber was instrumented using embedded 0.062" K-Type thermocouples. The chamber was designed for the thermocouples to be seated 0.80" from the surface in contact with the combustion gases. Eight 0.030" K-Type thermocouples were embedded into the duct flange. Four 0.062" K-Type thermocouples were instrumented at the nozzle exit of the copper chamber for the injector screening tests in order to measure the gas temperatures for each burn. The thruster was also instrumented with 0.030" E-Type thermocouple probes for manifold and cryogenic injection fluid temperature measurements. Where high accuracy cryogenic temperature measurements were required on the test facility, specifically at the cavitating venturi location, resistance temperature devices (RTDs with ± 0.25°R calibration) were used. Setting the upstream cavitating venturis pressure controlled the propellant flow rates.

TEST RESULTS

Hot-fire tests were conducted between March 08, 2002 and April 05, 2002. All runs were performed at sea-level conditions. During the test program, 7 engine configurations were tested, including 2 combustion chambers, 3 LOX injector pintle tips, and 4 LH_2 injection settings. The operating conditions that were surveyed were 70 to 100% thrust levels, mixture ratios from 3.27 to 4.29, and 18.0 to 25.5% LH_2 to fuel film cooling. The copper heat sink chamber was used for 16 burns, each burn lasting from 0.4 to 10 seconds, totaling 51.4 seconds, followed by Haynes chamber testing ranging from 0.9 to 120 seconds, totaling 300.9 seconds. The total accumulated burn time for the test program is 352.3 seconds. The performance of the engine ranged from 83 to 95% C* efficiency for burns lasting 5 seconds or longer. The temperature on the Haynes chamber remained below 1000°F, with the exception of the localized hot spot that persisted throughout the entire test program. This thermal anomaly limited the ability to perform a complete mixture ratio survey with the Haynes chamber as originally planned.

Injector Screening

Only one pintle tip was evaluated during the injector screening tests with the copper heat sink chamber. Due to schedule limitations, the objective of the injector screening series was limited to identify a combination of acceptable performance and temperature conditions for long duration burns with the Haynes chamber. A systematic mapping of injector configurations could not be accomplished. The performance ranged from 90 to 95% combustion (C*) efficiency for the 5 to 10-seconds screening burns. The data trends show that higher combustion efficiencies could still be reached with higher mixture and/or momentum ratios. The performance and thermal results for the injector screening are summarized in Figures 13 through 16.

Performance and Thermal Evaluation

The results for the performance and thermal evaluation are also included in Figures 13 to 16. These runs were completed with two tips and the Haynes chamber. After the first five burns were run with the initial pintle tip (-2) and fuel setting (0.0454" shim stack), it became apparent that a mixture ratio survey could not be performed beyond O/F of 3.7 while maintaining acceptable temperatures in the hot spot area as shown in Figure 13. An adjustment to the hardware to reduce the oxidizer to fuel momentum ratio was performed to reduced maximum wall temperatures while increasing the mixture ratio. A 0.0401" fuel shim stack was installed to reduce the oxidizer to fuel momentum ratio. A 4.0 mixture ratio was attempted in a 5 second burn, but the temperatures were still too high. The –5 pintle tip with a lower oxidizer pressure was then installed resulting in lower temperatures in the hot zone of the chamber (Figure 13). A test with a 4.0 mixture ratio was performed for the last burn of the test program. While most of the temperatures on the chamber remained within 1000°F during steady state, a temperature abort in the hot spot limited the burn duration to 15 seconds. The performance achieved in this test series ranged from 83 to 95% C* efficiency. Programmatic considerations and the thermal anomaly at the bottom of the chamber both limited the range of operating conditions that could be performed. .

Thermal Anomaly

As already mentioned, a thermal anomaly persisted throughout the test program, causing a hot spot at the bottom of the chamber throat when looking down in the chamber through the nozzle exit, and that limited the range of testing conditions. The Haynes® 188 chamber temperatures in the region from 9-3 o'clock did not exceed 1000°F for all the burns in this test program. On several burns, the temperatures at the 4:30 and 7:30 positions reached up to 1500°F. The hot spot was very localized at the very bottom of the nozzle as illustrated in Figure 17. Because the anomaly persisted after different changes to the hardware configuration, it is unlikely that the problem can be related to localized defects in the injector. The hot spot also existed during the injector screening tests with the copper chamber configuration, although the effect of the anomaly was reduced due to the conductivity of copper and shorter burn durations. All temperature readings, with the exception of those in the hot spot were consistent with each other, and reached steady state within approximately 25 seconds. Figure 18 shows the thermal trends of the temperatures at the throat, where TCX3 and TCX7 (6 and 7:30 positions, respectively) run away from the grouping of temperatures along the rest of the throat.

Chamber Pressure Dips

The combustion pressure dips that can be observed in both Figure 12 and Figure 18 were possibly due to the ice formation in the thruster. Frost and ice formation were recorded in post-test hardware inspections as illustrated by Figure 19. High-speed camera footage revealed debris flying out of the nozzle during the burns (Figure 20) that can only be ice since no hardware change could explain the debris. During the injector screening tests, it was also observed that some thermocouples placed in the gas stream were bent out after the tests. It was likely caused by ice entrained in the flow. These observations fit the pressure dip phenomena since debris also occur in the few initial seconds of the burn. No rigorous correlation such as time of occurrence for instances was established. The number of pressure dip occurrences per burn and the timing of the dips were random from burn to burn.

E-Type Thermocouple Inaccuracy

After reviewing data from test article LH_2 cold flows and main stage checkouts, it became apparent that the fuel inlet thermocouples were reading vapor temperatures when pressure measurements and flow characteristics indicated that the propellant was in liquid phase (the critical conditions of hydrogen are 59.7 R, and 190.7psia.) Since the quality of the propellant is critical in determining injector-operating conditions such as velocities and momentum ratio of the propellant injection further inquiry was needed. The investigation revealed that the accuracy of the calibration data, provided by the supplier is not guaranteed below 140°R, even though the thermocouples will continue to operate down to 40°R and that calibration services are not provided below liquid nitrogen temperatures. An in-situ calibration of the E-type thermocouples against RTDs calibrated to an accuracy of ± 0.25°R, was performed inside the facility lines using liquid hydrogen.

Only two E-Type thermocouples could be re-calibrated in the facility lines. The calibration cold flow consisted of a LH_2 chill from ambient down to 40°R in the facility fuel injector and duct legs. The temperatures rose to 200°R and the chill down was repeated two times from 200°R to 40°R for repeatability. The data shows that at the saturation range (40-50°R), the E-type thermocouples were reading 17°R higher than the RTDs. The data collected for each of the thermocouples showed repeatability for each of the 3 chill-down runs, however, the data was not repeatable between the two different E-type thermocouples. Based on these results, specific calibration curves were generated and applied to each corresponding thermocouple and the thermocouples were placed into the test article fuel injector temperature sense ports. The standard calibration curves for E-Type thermocouples were applied to all other E-type thermocouples in the test set up. From the LH_2 calibration data, it can be inferred that the temperature readings for the cooling duct injection were ~15°R higher than the actual temperature for the duct inlet.

CONCLUSIONS

The performance success criteria of ≥85% C* efficiency was met and the data shows that performance is a strong function of momentum ratio. With the exception of the hot spot at the bottom of the nozzle the thermal success criteria of ≤1850°F was also met. The temperatures on the Haynes® 188 chamber were below 1000°F from the 9 to 3 o'clock positions, and reached steady state within ~25 seconds. These results confirm that Haynes® 188 is a good candidate for the RCS thruster application. Based on these test results several injector configurations can both generate high performance while still maintaining thermal margin compatible with the

RCS thruster design goals. Still, the data trends suggest that even higher performance can be achieved by increasing the mixture ratio and/or momentum ratio.

The workhorse engine program shows that both the coaxial liquid-on-liquid pintle injector and fuel duct design concepts are viable for the intended application. The thruster head-end design maintained cryogenic injection temperatures while firing, which validates the design for minimal heat soak and is critical for pulse mode operation. By injecting fuel into the duct, the throat temperatures were maintained below 1000°F, with the exception of the hot zone at the bottom, yet the split of fuel through the cooling duct does not compromise the overall combustion efficiency, which indicates that provided proper design refinement, such a concept can be applied to a high-performance version of the thruster. Although the duct design point used hydrogen temperature of 80°R at the cooling passage inlet, it was tested over a wide range of inlet conditions, including sub-critical (i.e. high-density) LH_2. These hot fire tests demonstrate the robustness of the duct design concept and good capability to withstand off-nominal operating conditions without adversely impacting the thermal response of the engine, a key design feature for a cryogenic thruster.

The limited available data indicates that ice formation may be the most probable cause for the thermal anomaly and the chamber pressure dips. The test article purge procedure may need to be modified to ensure that frost formation does not occur inside the combustor during chill down. The frost observed on the chamber walls could be responsible for the abnormal response, however, another source of water is the combustion products themselves; water vapor could solidify on the cold combustor walls at start up. In order to distinguish the root cause for ice formation, the chill-down procedure should be re-worked before proceeding onto future testing.

Acknowledgements

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1. Test Report - DRD RCST.951-DE-020-LH2, Revision A; dated September 20, 2002

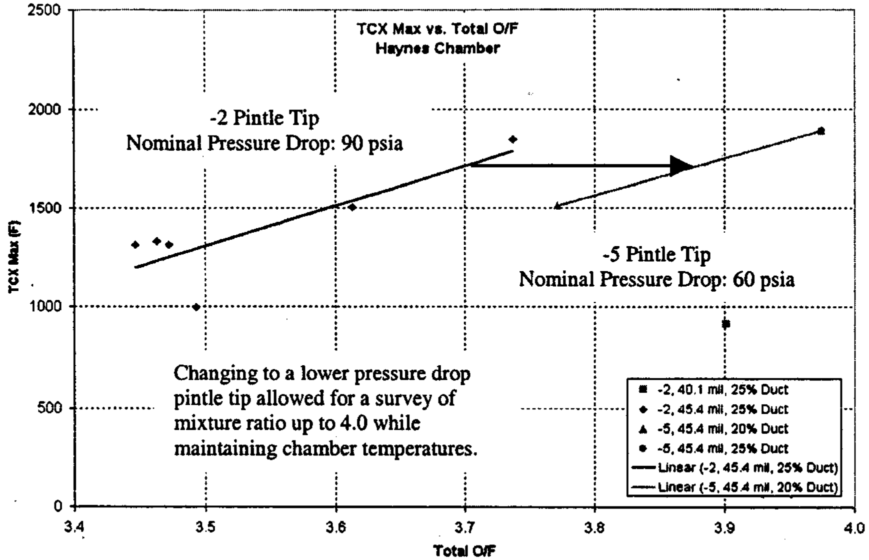

Figure 13: Performance & Thermal Evaluation – TCX Max vs. Mixture Ratio

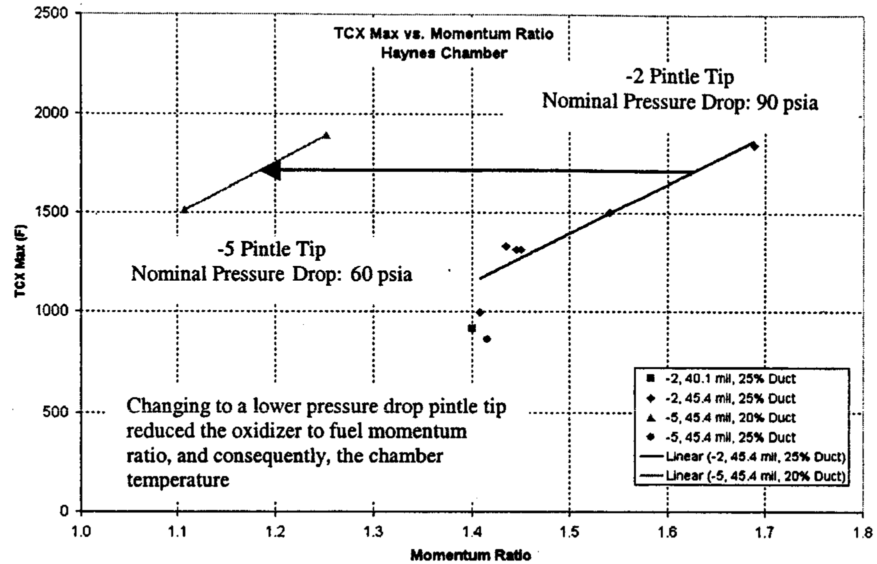

Figure 14: Performance & Thermal Evaluation – TCX Max vs. Momentum Ratio

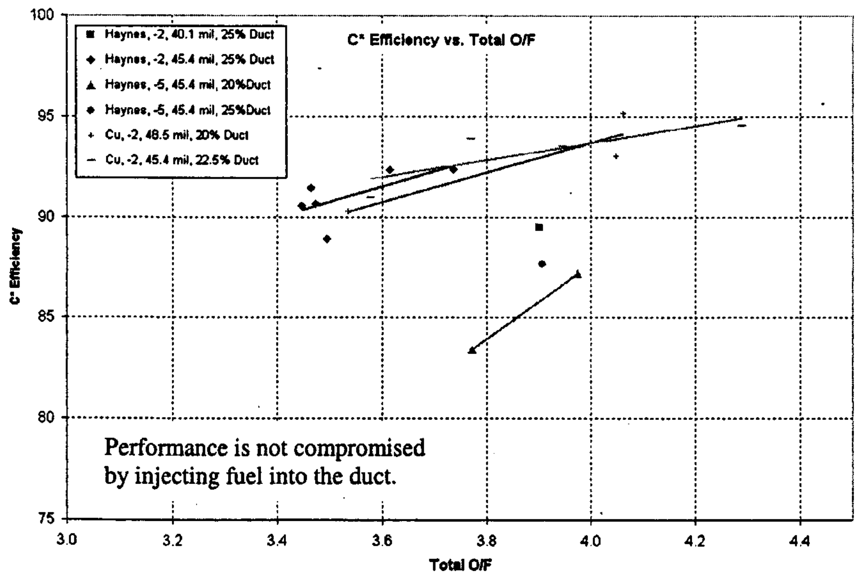

Figure 15: Test Program Performance Summary – C*Efficiency vs. Mixture Ratio

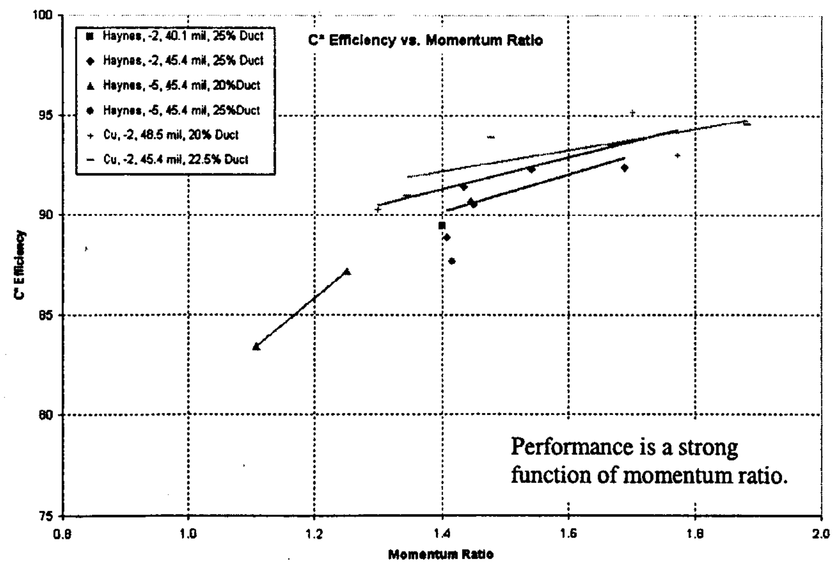

Figure 16: Test Program Performance Summary – C*Efficiency vs. Momentum Ratio

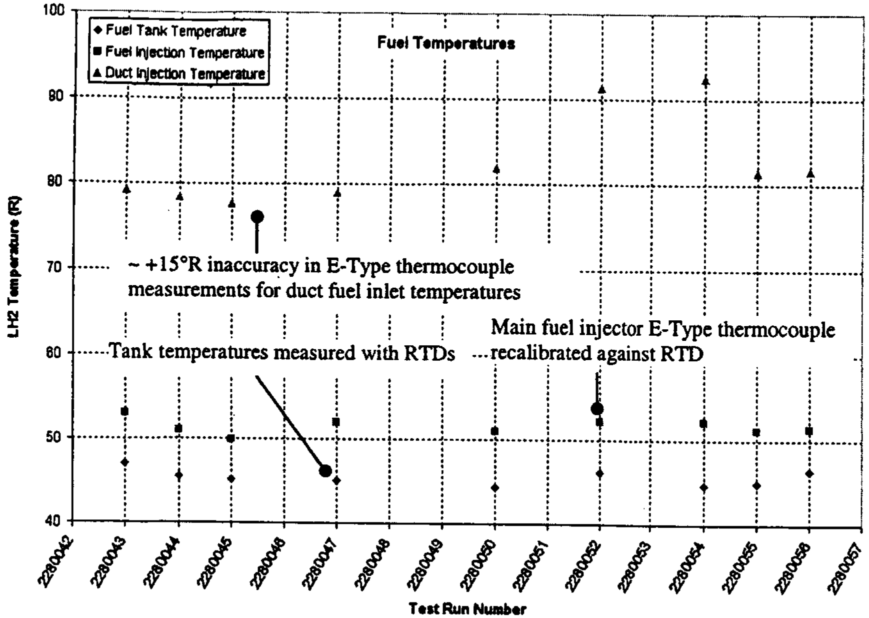

Figure 11: Fuel Inlet Temperatures

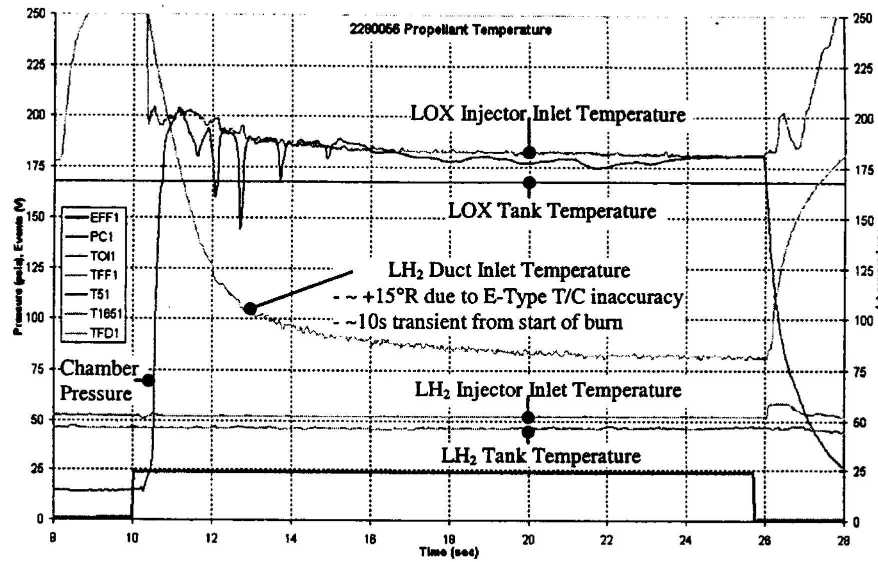

Figure 12: Burn 2280056 Propellant Inlet Temperatures

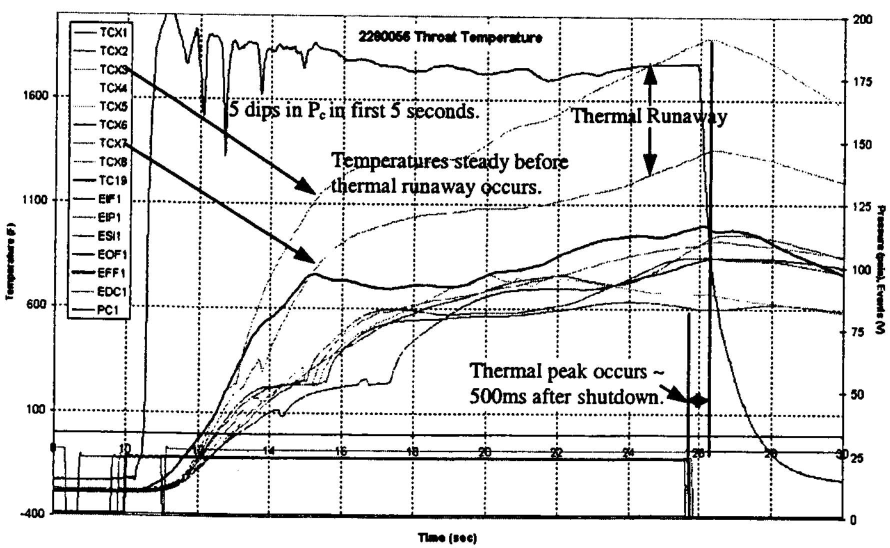

Figure 18: Throat Temperatures – Burn P2280056

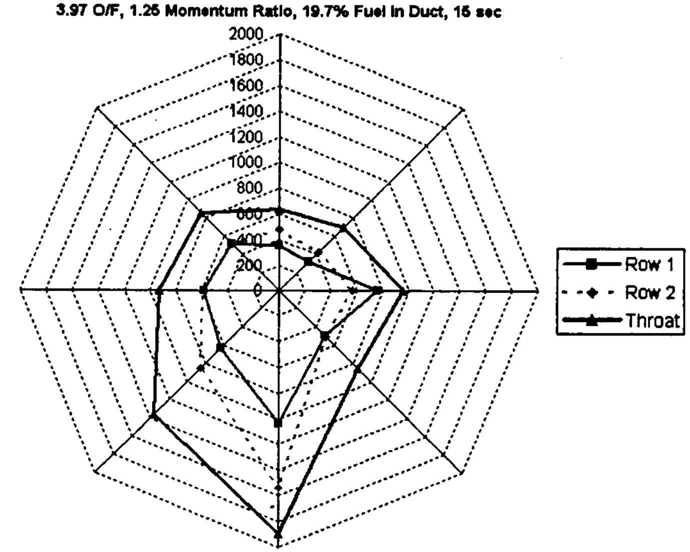

Figure 17: Burn P2280056 Haynes® 188 Chamber Temperatures

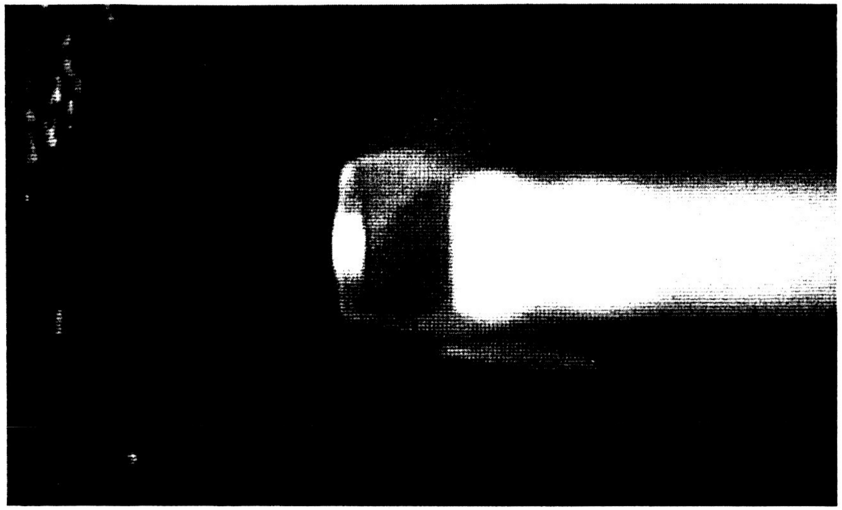

Figure 20: Haynes Chamber Checkout Burn P2280041 – Debris at Bottom of Plume

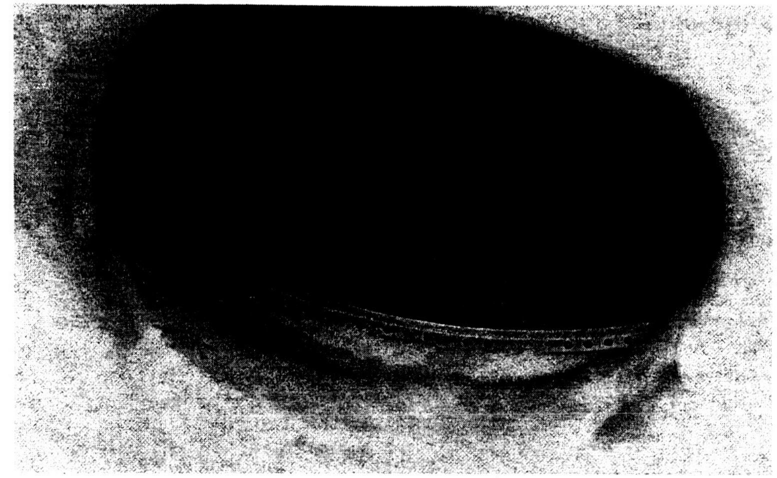

Figure 19: Ice Accumulation at Bottom of Chamber/Duct – Post Burn P2280020