Northrop Grumman TR202 LOX/LH2 Deep Throttling Engine Technology Project Status

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NASA's Propulsion and Cryogenic Advanced Development (PCAD) project is currently developing enabling propulsion technologies in support of future lander missions. To meet lander requirements, several technical challenges need to be overcome, one of which is the ability for the descent engine(s) to operate over a deep throttle range with cryogenic propellants. To address this need, PCAD has enlisted Northrop Grumman Aerospace Systems (NGAS) in a technology development effort associated with the TR202 engine. The TR202 is a LOX/LH2 expander cycle engine driven by independent turbopump assemblies and featuring a variable area pintle injector similar to the injector used on the TR200 Apollo Lunar Module Descent Engine (LMDE). Since the Apollo missions, NGAS has continued to mature deep throttling pintle injector technology. The TR202 program has completed two series of pintle injector testing. The first series of testing used ablative thrust chambers and demonstrated igniter operation as well as stable performance at discrete points throughout the designed 10:1 throttle range. The second series was conducted with calorimeter chambers and demonstrated injector performance at discrete points throughout the throttle range as well as chamber heat flow adequate to power an expander cycle design across the throttle range. This paper provides an overview of the TR202 program, describing the different phases and key milestones. It describes how test data was correlated to the engine conceptual design. The test data obtained has created a valuable database for deep throttling cryogenic pintle technology, a technology that is readily scaleable in thrust level.

Nomenclature

A	=	Area
C^*	=	Combustion chamber characteristic velocity
c_p	=	Constant pressure specific heat
D	=	Diameter
h_g	=	Hot gas heat transfer coefficient
L^*	=	Combustion chamber characteristic length
L'	=	Combustion chamber length from injector to throat
$\dot{m}$	=	Mass flow rate
Pr	=	Prandtl number
R	=	Radius
V	=	Velocity
ρ	=	Density
μ	=	Viscosity
σ	=	Boundary layer correction factor

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I. Introduction

NASA's Propulsion and Cryogenic Advanced Development project is developing enabling technologies in support of future lander missions. On May 9, 2005 Northrop Grumman Aerospace Systems (NGAS) was awarded a contract responding to the need for "variable thrust rocket engines for landers and ascent vehicles that can use in-situ produced propellants." With this development goal in mind, NGAS has teamed with engineers from NASA for technology development associated with the advancement of the TR202 engine, a LOX/LH2 expander cycle engine driven by independent turbopump assemblies and featuring a variable area pintle injector.

For the TR202 engine, NGAS leveraged the throttling pintle injector technology from the Apollo Lunar Module Descent Engine (LMDE) as well as many other programs. NGAS heritage company TRW developed the Apollo LMDE, which was flown successfully on all crewed moon missions. Since that time, NGAS has developed cryogenic engines using LOX/LH2 and LOX/CH4 and tested the turbopump driven pintle injector engine, the TR108.[1] Also, NGAS developed expander cycle engine modeling tools as part of the Air Force's Upper Stage Engine Technology (USET) program.[2]

During Phase I of the TR202 program, the team developed a conceptual design of the engine. The key components of the conceptual engine are shown in Fig. 1. The engine features independent turbopump assemblies, a regeratively cooled combustion chamber that maintains hydrogen coolant above the critical pressure, and a throttle actuator that controls the pintle position. Figure 2 provides engine characteristics for the TR202. The engine provides a vacuum thrust of 8,725 lbf at 100% power and throttles to 1,600 lbf at 18.8% throttle. The engine was designed to nominally operate at 75% power with 4:1 throttling capability. This required capability was derived from vehicle level trade studies grounded in mission needs identified by NASA's Lunar Surface Access Module architecture studies. Phase I of the program culminated in a successful Conceptual Design Review of the engine system on April 5, 2006. Reference 3 provides details of requirement creation and design trades performed to reach the conceptual design.

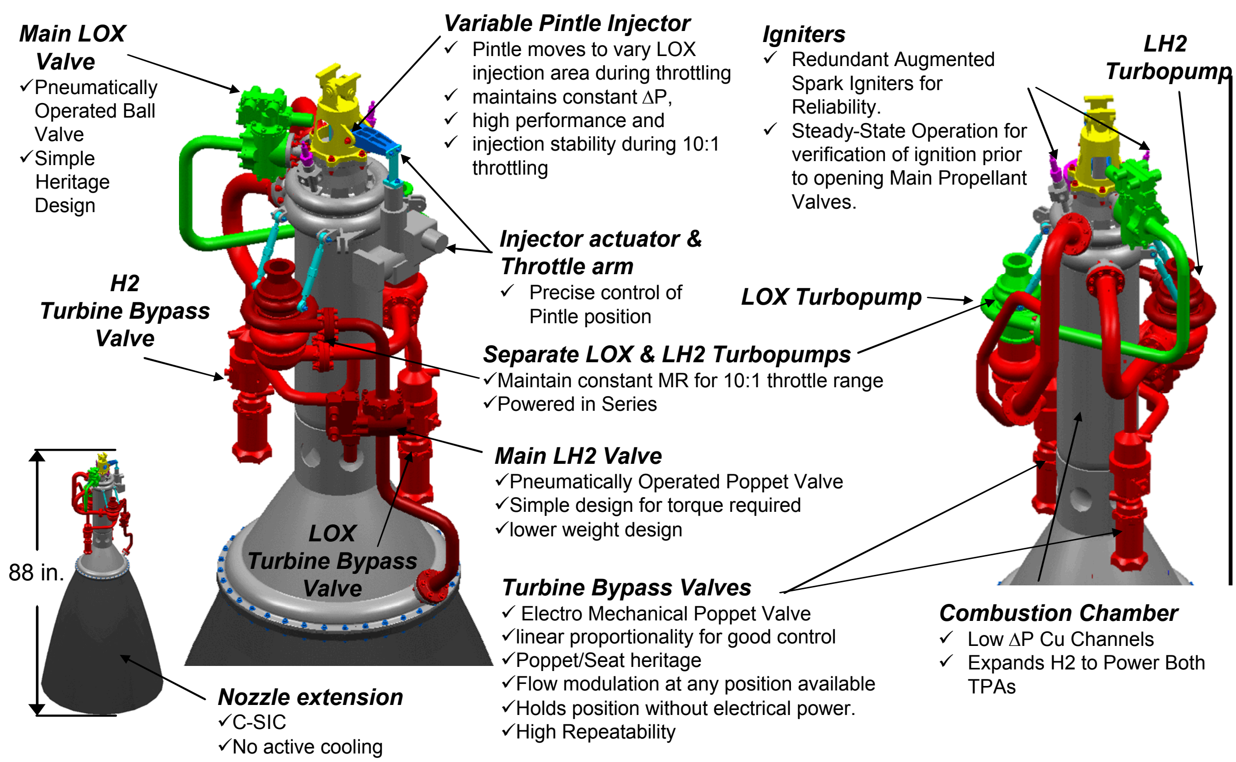

Figure 1. TR202 Engine Key Components

During Phase I, NGAS identified key technical challenges for cryogenic deep throttling engine designs, listed in Fig. 3. Since the conceptual design review, NGAS has been tasked to demonstrate LOX/GH2 variable area pintle injector technology, addressing the first four technical challenges and partially addressing challenges 5 through 7. During Phase II, Option 1, NGAS developed detailed designs for a test-bed pintle injector and igniter assembly, which incorporates the flexibility to demonstrate and optimize performance over a deep 10:1 throttle range. In November 2007, Phase II Option 2 was initiated. This option saw NGAS successfully build a test-bed pintle

Engine Cycle	Closed Expander	Nozzle Area Ratio	200:1
Propellants	LOX/LH2	Length:	88”
Engine Mixture Ratio	6.0 throughout the throttle range	Diameter:	40”
Engine Throttle Range	100% to 18.8%	Weight	280 lbm
Vacuum Thrust Range	8,725 to 1,600 lbf	Reliability Against Catastrophic Failure	0.999967
Engine Out Philosophy for Design	Shut down failed engine, gimbal remaining 3	Vacuum T/W	31 at 100% power 23 at 75% power
Engine Out Power Level	8,680 lbf	Chamber Pressure	700 – 130 psia
Nominal Rated Power Level	6,465 lbf	LOX Pump Discharge Pressure	890 – 270 psia
Vacuum Isp	453 sec to 436 sec	H2 Pump Discharge Pressure	1,785 - 240 psia

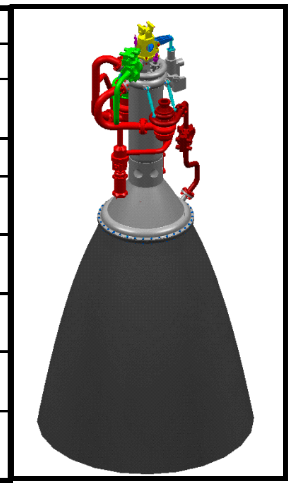

Figure 2. TR202 Engine Characteristics

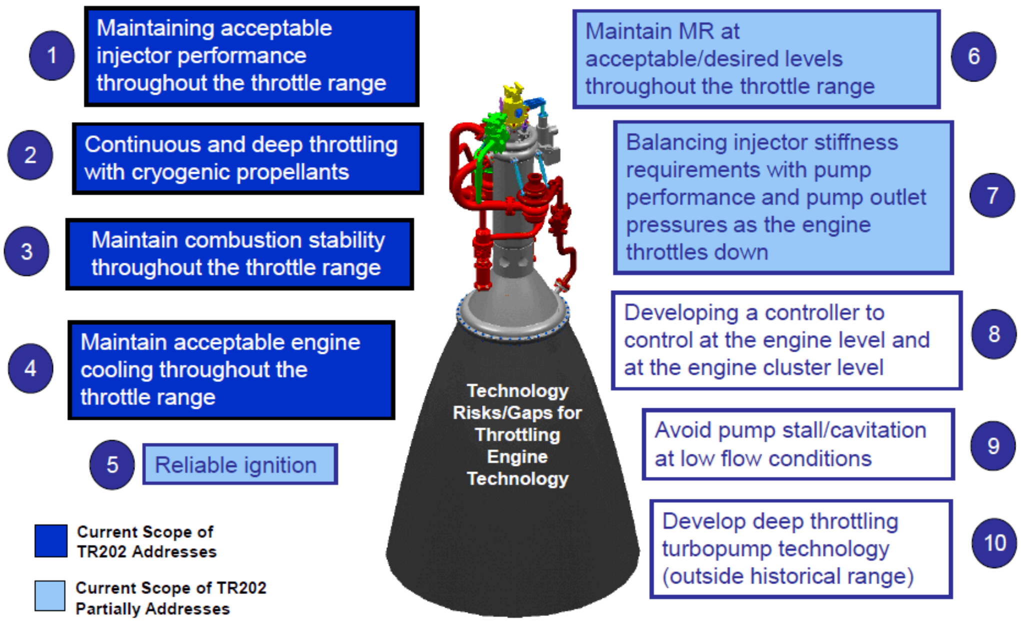

Figure 3. Technical Challenges for Cryogenic Throttling Engine Design

injector assembly for testing.[4] This paper provides an overview of the test results and shows how the results correlate back to the conceptual engine design.

The goal of the TR202 LOX/GH2 pintle injector test campaign was to establish baseline performance and heat transfer characteristics over a 10:1 throttle range and to develop an optimized scalable pintle injector configuration. During test planning the team developed a number of primary and secondary objectives that trace back to the key technical challenges outlied in Fig. 3.

Primary Technical Objectives

- Demonstrate high-performance (>98% C*) within 75%-100% power band
- Demonstrate stable combustion over 10:1 throttle range with high-performing injector
- Measure total chamber heat transfer over 10:1 throttle range
- Demonstrate adequate thermal power availability for engine cycle closure

Secondary Technical Objectives

- Demonstrate optimized performance and heat flux characteristics
- Obtain parameterized design data for performance & heat flux

- Measure performance at off-nominal MR
- Obtain L* (L') parametric performance data

II. TR202 Hardware

The TR202 injector hardware was designed and manufactured for flexibility on the test stand. It is workhorse hardware with high safety margins. Care was taken to ensure key injector components were easily accessible and easy to change out. This allowed for testing of multiple injector configurations and throttle levels in a single day. Internal flow geometry is similar to the conceptual flight engine. Throttle level is set by shims which can be removed and replaced. The chamber interface is compatible with the NASA MSFC "40K" calorimeter chamber, MSFC ablative chambers, and MSFC throttling test-bed chambers. The overall dimensions of the injector assembly are 18 in. by 18 in. by 9.5 in. and the weight is approximately 200 lbs. The injector hardware is depicted in Fig. 4.

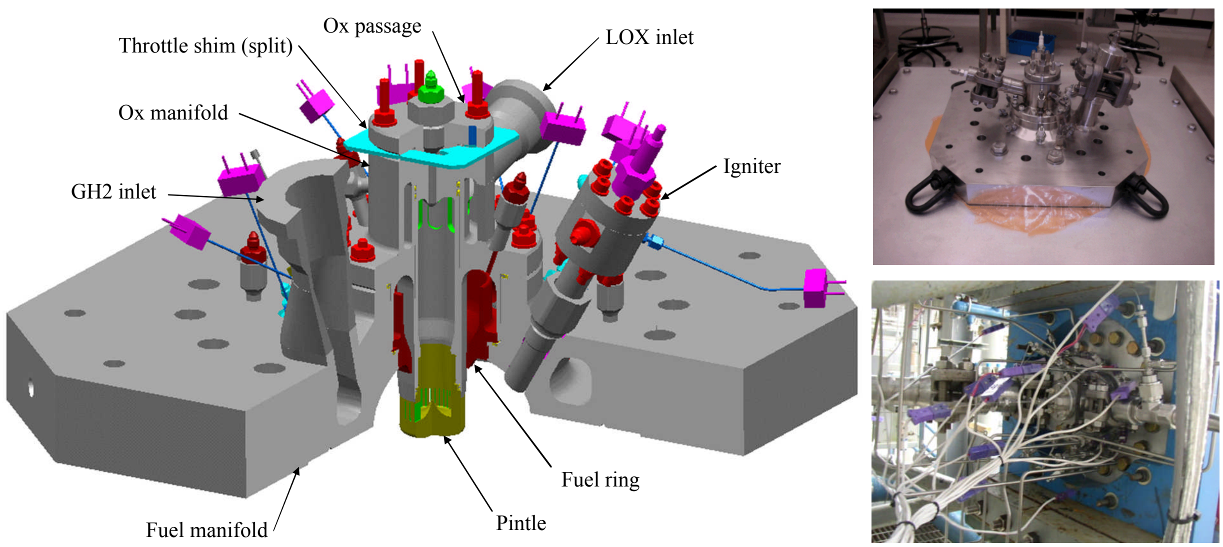

Figure 4. TR202 Test-Bed Injector

The two parts changed out to reconfigure the injector flow paths are the fuel ring and the pintle. The same basic design was used for the fuel rings with five test parts of incrementally increasing diameters (i.e. increasing fuel flow areas). Twelve variations (or dash number) of the same basic screw-on pintle design were built for test evaluation. The injection geometry variations involved 5 numbers of slots arranged in a single row using two different sizes of LOX injection total flow area and 6 length-to-width aspect ratios for the individual slots.

The -1 and -3 pintles were fitted with ablative tips to mitigate effects of adverse thermal conditions. The -11 and -12 pintles had a shortened tip, to reduce the conduction path from cold LOX to the combustion gasses. The -12 tip also was coated to assist in thermal management. Pintle and fuel ring blanks allowed new injector designs to be manufactured quickly. During testing, a new fuel ring configuration could be manufactured in 1 day and a new pintle design could be built in less than 2 weeks.

The TR202 team selected a rugged test-bed igniter design, shown in Fig. 5. It is based on an ox-rich GOX/GH2 igniter originally designed by engineers at NASA Glenn. For the TR202 test-bed application, the igniter flame tube is slightly longer than used previously, and the chamber pressure after main stage ignition is higher than had previously been demonstrated at NGAS. Igniter operation is discussed in detail in Ref. 5

To support TR202 test-bed hot-fire testing, the team utilized MSFC procured silica phenolic ablative chambers as well as existing NASA MSFC calorimeter hardware. The chamber configurations are shown in Fig. 6. The ablative chambers were based on previous MSFC chamber builds. They consist of ablative liners fabricated at ATK and a CRES liner. In the liner, 6 pressure ports were incorporated to measure static pressure along the barrel and just upstream of the convergent section. The full calorimeter assembly consisted of an eight-channel 4 in. long circumferentially cooled piece mated to the injector, two axially cooled twenty-eight channel 8 in. long spool pieces, and a seventeen-channel circumferentially cooled throat piece. For reduced length calorimeter chamber testing, one of the 8 in. long spool pieces was replaced with a multi-piece 4 in., modular, low cost calorimeter concept designed and built by MSFC.

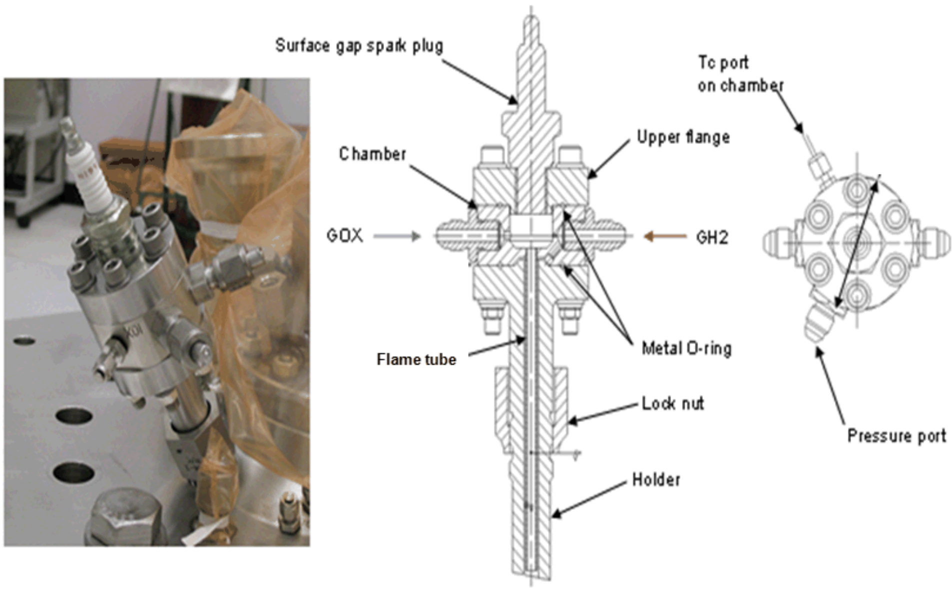

Figure 5. TR202 Test-Bed Igniter

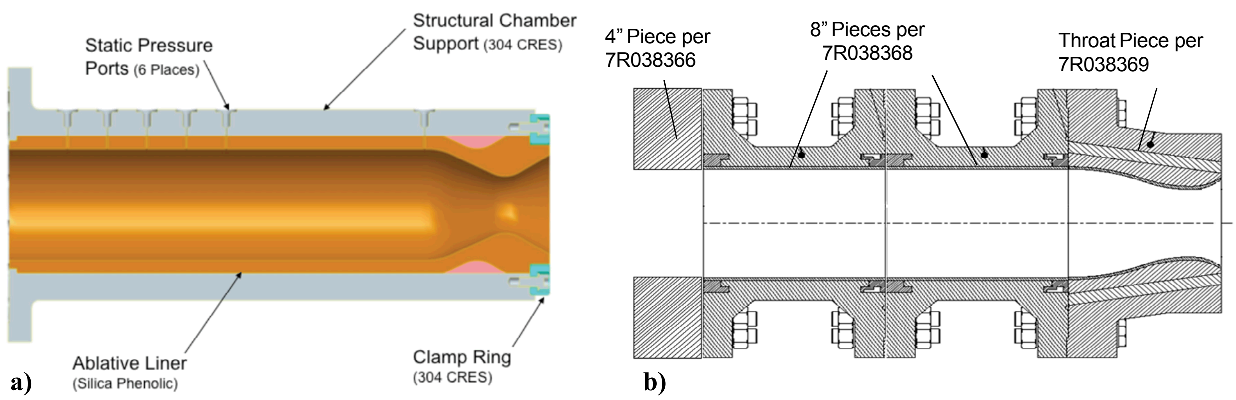

Figure 6. a) Ablative and b) Calorimeter Chamber Configurations for the TR202 Test-Bed Program

NASA MSFC engineers worked to procure a calorimeter nozzle spool with a throat diameter approximately equal to the design size of the conceptual engine throat. However, a series of fabrication issues led to cost and schedule impacts that precluded development of the hardware for this project. Therefore, an existing larger diameter throat piece was used to support testing. The implication of this was two-fold. First, chamber pressure as a function of mass flow rate was reduced, and as plumbed, the test stand was not capable of LOX flows associated with 100% power (Pc of ~700 psia). Second, injector performance optimization was done on a chamber with a lower than typical contraction ratio. The full calorimeter chamber assembly, with plumbing installed, is shown in Fig. 7.

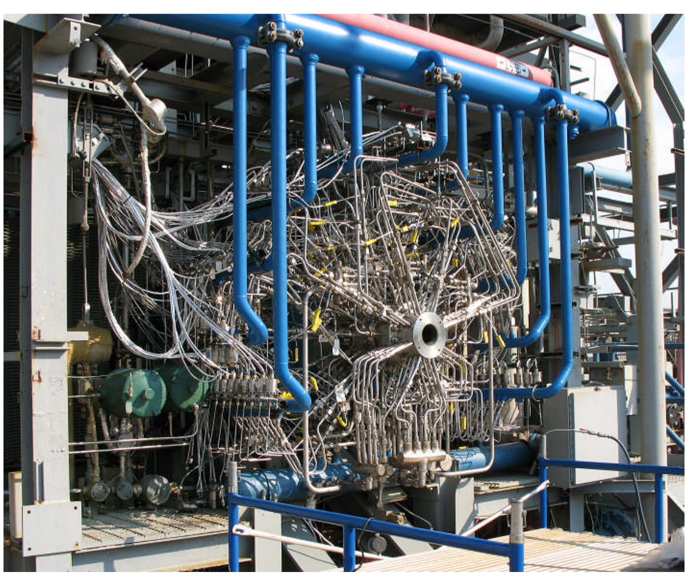

Figure 7. Calorimeter Chamber Installed on Test Stand 116

III. Ablative Test Series

TR202 test-bed injector testing with ablative chambers took place in the first half of 2009 as outlined in Fig. 8. The injector and the first of 5 chambers utilized were installed on the stand and instrumented starting in February. Cold flows of GH2 and LOX and igniter checkouts were conducted to establish valve sequencing and to determine facility and test article pressure and temperature responses. Hot-fire testing began on April 7th, 2009 and consisted of 22 tests with 6 pintles and 2 fuel rings utilized. The ablative chamber testing is described in detail in Reference 5.

Initially, ablative testing was meant to characterize and optimize pintle performance. Soon after hot-fire testing began, data analysis showed that ablative throat erosion was much greater than anticipated. The MSFC supplied ablative chambers had been used before, but not without film cooling at the pressures used for TR202. The inability to determine throat diameter caused performance uncertainty to reach levels that did not allow the team to easily distinguish performance trends with different pintle configurations. Therefore, performance characterization was accomplished during calorimeter testing.

The ablative testing did provide valuable risk reduction for calorimeter testing and demonstrated compliance to a number of test objectives. During cold-flow testing and igniter checkouts, a detailed facility system model was created and characterized in the ROCETS code. This model and the initial hot-fire tests were used to create robust start and shutdown sequences that were then used during calorimeter testing. Ablative testing also demonstrated stable 10:1 throttling for the first time.

During Tests 12 and 14, the -3 and -1 copper injectors experienced overheating at the pintle tip. Both of these pintles had fewer slots then the pintles used in all preceding tests. It is hypothesized that the large distance between slots allowed more hydrogen flow into the chamber core and changed the thermal environment near the pintle tip. This type of thermal issue was anticipated to be a possible problem prior to testing. To reduce risk, several ablative tips were manufactured during Phase II Option 1 that could be screwed onto the copper pintles. Existing copper pintles could be readily modified to accept the ablative tips. This was done successfully for the -3 pintle that was used during calorimeter testing. Additionally, the larger -5 fuel ring used for high performance tests on the calorimeter chamber reduced the incoming hydrogen velocity by more than 25% compared to the -2 fuel ring used for the ablative chamber tests. The reduction in GH2 velocity likely helps to mitigate the thermal issue. The long-term solution is to utilize improved tip cooling, thermal barrier coatings, and/or alternate metal material on the pintle tip that can withstand high temperatures and oxidizing environments better than copper.

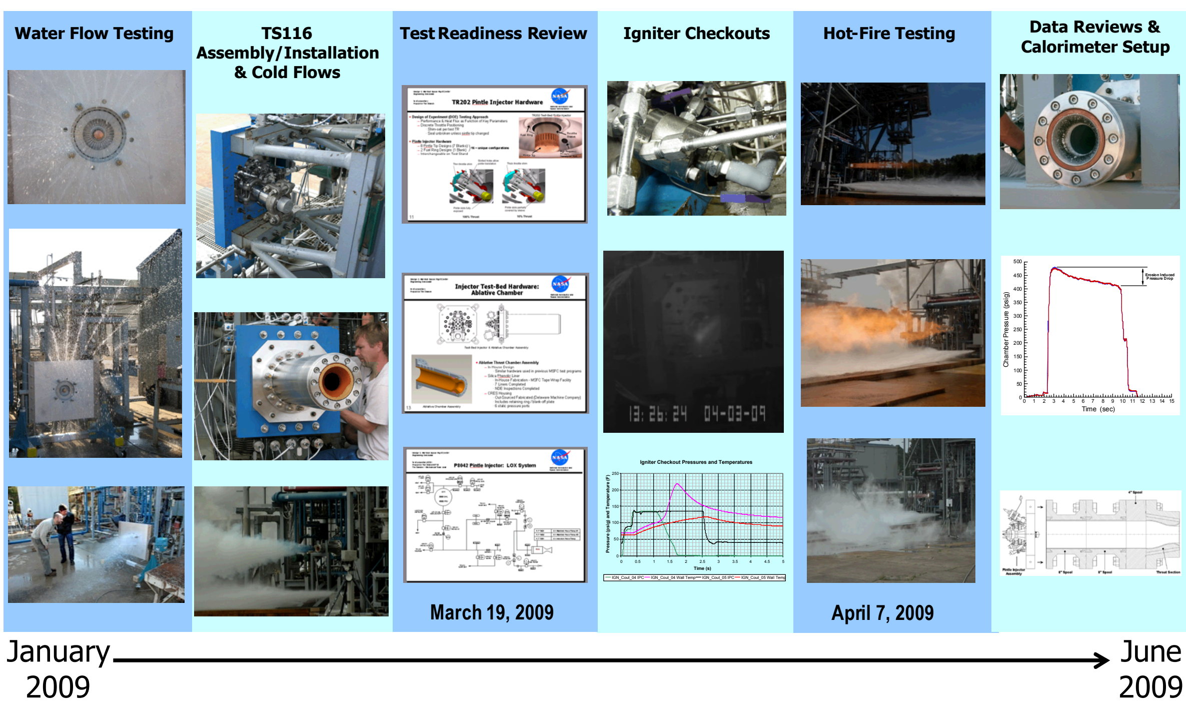

Figure 8. Ablative Test Series Highlights

IV. Calorimeter Test Series

After the ablative test series, the TR202 injector was connected to a calorimeter chamber for additional tests. Originally the goal for these tests was to characterize the injector heat transfer with an optimized injector configuration. Because the ablative throats ablated too rapidly to accurately determine performance, the calorimeter campaign added performance characterization to its objectives. Forty seven tests were performed during six test weeks that spanned three months from August until November 2009. During that period 15 unique injector configurations were utilized and five different power levels from 7.5% to 75% were tested. This was made possible by the flexibility of the hardware and the forward thinking hardware philosophy that was described above. Details of this test series are presented in Reference 6.

To characterize performance, a design of experiments (DOE) approach was utilized. These tests were performed at a power level of 75% and a target mixture ratio of 6.0. From these tests, the general performance trends were determined and high performance configurations were identified.. The highest performance configurations tested used the ablative-tipped -1 and -3 pintles. The -3A pintle with the -5 fuel ring achieved a 98.6 ± 1.2 % C* efficiency over the $5.96 < MR < 6.4$ range at 75% power. As shown in Fig. 9, this pintle was then successfully throttled 10:1. After these tests, the reduced length chamber was tested. With this chamber and the same injector configuration, performance at 75% power dropped 0.6 – 0.9% over the mixture ratio range. The injector configuration was again throttled successfully.

Another objective of the calorimeter campaign was to characterize heat flux for a LOX-GH2 pintle injector. This was done for all tests. The results for the -3A pintle and -5 fuel ring are shown in Fig. 10 across the throttle range. The figure shows that at all power levels, the heat flux reaches the average values seen in the long axially cooled sections within 4 in. of the injector/chamber interface. As highlighted in the figure, the first channel of the throat spool piece has an out of family high heat flux for each test. This is likely due to a slight misalignment of the spool piece that causes a local recirculation zone. The circumferential heat flux profile of the two 8 in. spool pieces showed an approximately ± 10% variation.

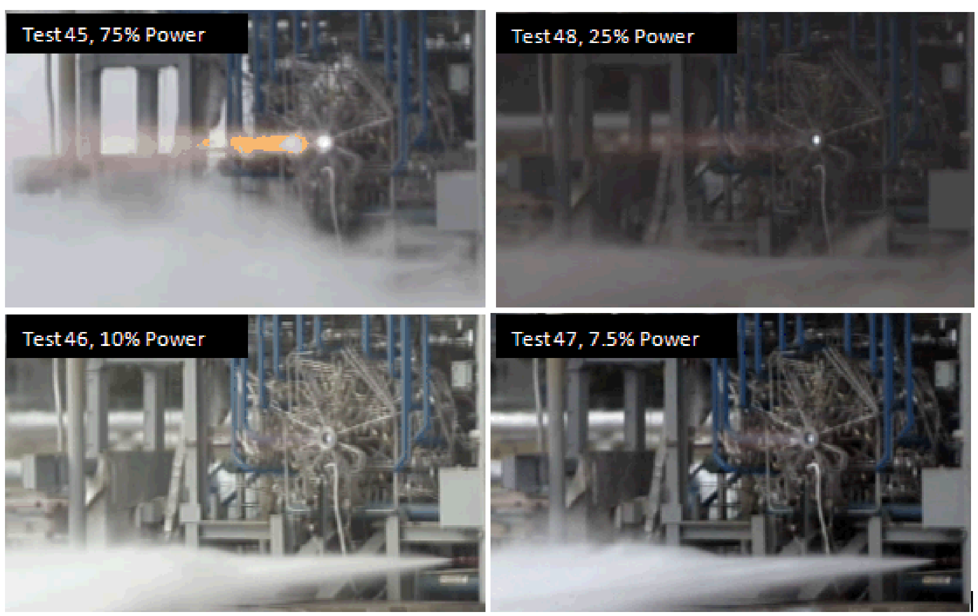

Figure 9. -3A Pintle with -5 Fuel Ring Hot-Fire Throttling Images

The TR202 injector showed excellent stability characteristics for all tests. This encompassed an MR range of 5.40 to 6.84 and a power range of 75% to 7.5% with the low power tests performed with nearly saturated state LOX injected. For all tests, the maximum oscillation was <1.0% of measured head-end pressure, with most tests having significantly lower maximum oscillations. For tests with maximum oscillations at 100-110 Hz, the response at frequencies below 100 Hz was investigated. Figure 11 shows a typical response for these tests at 75% and 7.5% power. These oscillations suggest bubbles in the LOX flow through the pintle slots locally changing the density and LOW flow conditions. Because the oscillations are not at a discrete frequency and of a very low magnitude, they are not a cause for concern for combustion stability, even for tests at low power with nearly saturated LOX injection.

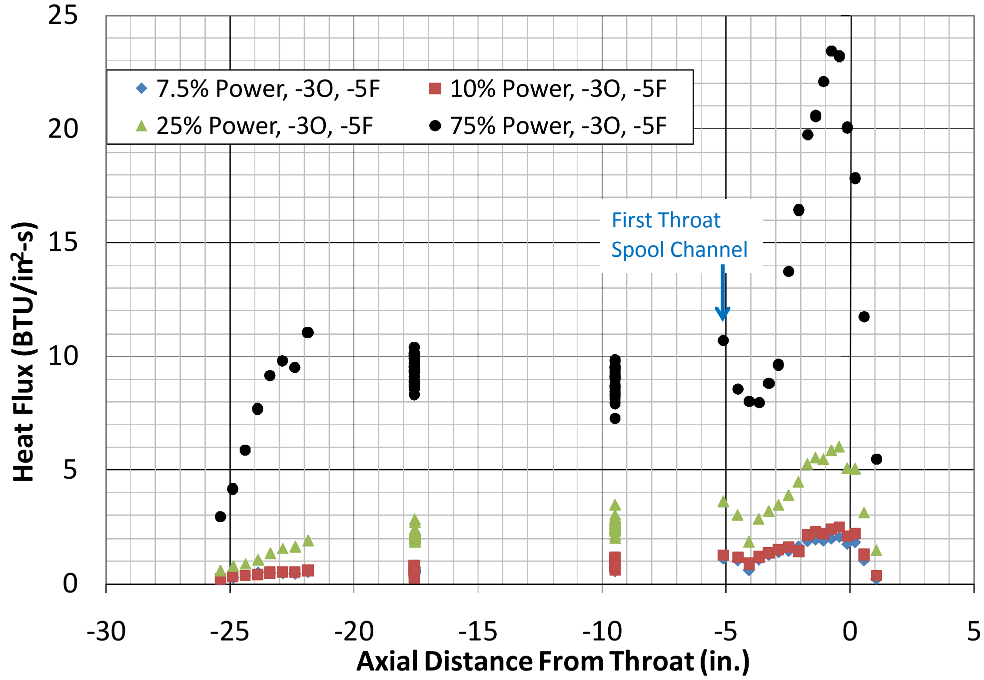

Figure 10. Axial Heat Flux Profile for -3A Pintle and -5 Fuel Ring with Nominal Chamber

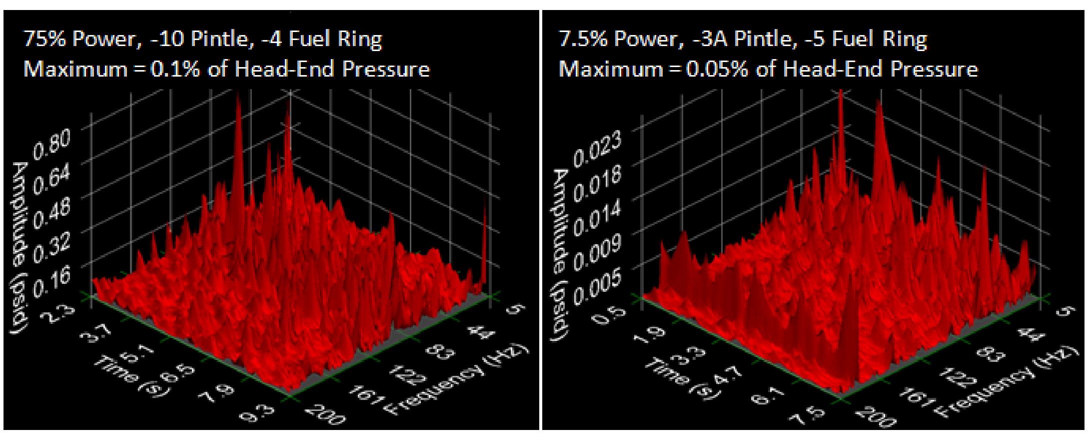

Figure 11. Typical Low Frequency Response of TR202 Injector

V. Post-Test Analysis

Because of the unique flow-field produced by pintle injectors, a NASA concern entering the program was accommodating the heat flux profile on the chamber wall. Testing alleviated those concerns, demonstrating that the heat flux profile was similar to that of a flat faced injector. For an expander-cycle engine, sufficient heat flow is necessary to drive the turbomachinery. During the conceptual design phase, a standard LOX-GH2 heat flux profile from a coaxial injector was assumed. A primary project objective was to demonstrate adequate thermal power availability for engine cycle closure. To do this, measured heat flows from calorimeter chamber tests were compared to cycle requirements defined during the engine conceptual design.

Three detailed power levels were examined during the conceptual design phase: 100%, 75%, and 18.8%. These power levels cover a 4:1 throttle from the nominal 75% power level and an overall 5.3:1 throttle ratio. The 100% power level was defined by the thrust required with one engine of a 4-engine cluster turned off. A constant 98% thrust chamber C* efficiency had been assumed with a mixture ratio of 6.0 across the throttle range. For the conceptual engine, % power level, % chamber pressure, % total flow rate, and % thrust were nearly identical, as indicated in Table 2. Cycle margin was defined as the percentage of total hydrogen flow that bypassed the fuel turbine. This margin increased with decreasing throttle setting, as indicated in Table 1. The conceptual design engine had a nozzle that was actively cooled out to an expansion ratio of 30:1. Approximately 17% of the total chamber heat transfer was predicted to come from the nozzle between and expansion ratio of 1.65 and 30. This portion of the nozzle heat transfer was not measured in the testing, as the sea-level testing required a short nozzle to inhibit flow separation.

Table 1. TR202 Conceptual Engine Throttling Characteristics

Engine System ConDR Power Balance Rev. D 04-2006											
% "PL"	Pc psia	% Pc	Thrust lbf	% Thrust	Flow lbm/s	% Flow	C* ft/s	C* Eff. %	Qdot BTU/s	Barrel Temp R	Power Margin % FTRB Bypass
100	700	100.0%	8,724	100.0%	19.25	100.0%	7,474	98.0%	4,000	700	20%
75	525	75.0%	6,526	74.8%	14.47	75.2%	7,457	98.0%	3,137		32%
50											
25											
18.8	131	18.7%	1,597	18.3%	3.66	19.0%	7,368	98.0%	967		61%
10											

C* trends with chamber pressure from test results are directly applicable to the cycle balance. The calorimeter nozzle throat tested was larger than the conceptual design configuration, meaning that heat transfer could not be mapped easily based solely on chamber pressure. Rather, heat transfer mapping requires a more sophisticated approach. The well-known Bartz correlation for hot gas heat transfer coefficient is

$$h_g = \frac{0.026}{D^{0.2}}\left(\frac{\mu^{0.2}c_p}{\Pr^{0.6}}\right)_0 (\rho_\infty V_\infty)^{0.8}\left[\sigma\left(\frac{D_t}{R_{WTU}}\right)^{0.1}\right] \quad \text{where} \quad \rho_\infty V_\infty = \frac{\dot{m}}{A} \tag{1}$$

The largest portion of test heat transfer comes from the barrel section, rather than from the nozzle. The tested and flight conceptual design barrel diameters and flow/surface areas are identical. Thus, the total heat flow is approximately proportional to the total chamber mass flow rate to the 0.8 power.

There are two key assumptions that must be true for this relationship to hold. First, it is assumed that thermodynamic properties don't change significantly at the different pressures and contraction ratios found for the test setup versus the conceptual engine. For frozen composition, the parameter does not vary at all with pressure. For equilibrium composition, the variation with pressure is slight, particularly at higher powers. The change in the thermodynamic parameter with contraction ratio was found to be negligible. The second key assumption is that the wall temperatures for the calorimeter chamber and the conceptual flight engine are similar. Except for a relatively small area near the throat, the temperatures are approximately equal.

Because chamber flow-rates were equal, the heat flux profile for the 75% power tests was compared to the 100% power level heat flux profile for the conceptual design. Two heat flux profiles were used during the conceptual design phase. A Bartz correlation based heat flux was used for the design of the chamber. The predicted heat flux in the chamber was low compared to flat-face injector test data supplied by NASA for comparison. Given the uncertainty of the pintle heat flux, a low heat flux ensured the cycle could be balanced because turbine power depends on heat pickup. Then, an elevated heat flux based on the flat-face injector test data was analyzed to ensure thermal margins were sufficient. Figure 12 shows the comparison between the design and test heat flux profiles. In the chamber, where direct comparison can be made, the

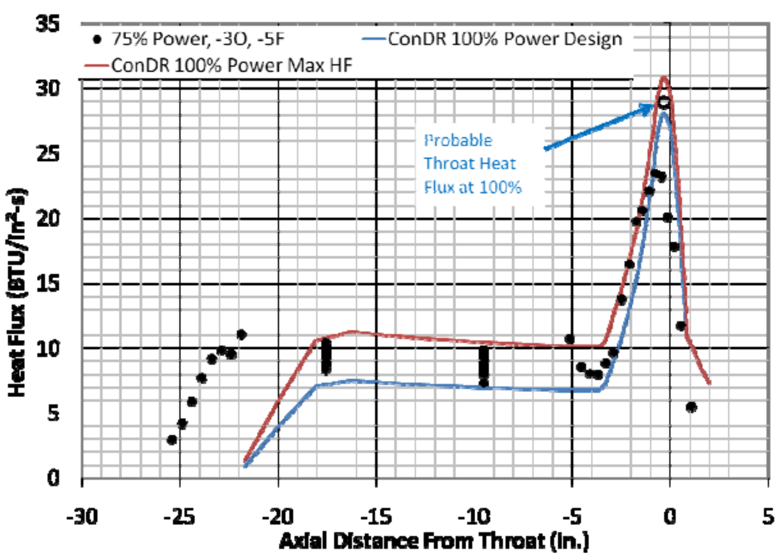

Figure 12. ConDR vs. Test Heat Flux Profile Comparison

pintle heat flux sits directly between the design and maximum heat flux cases. Also, at the head-end of the chamber the slope of increase matches the slope used for the ConDR analysis. Finally, multiplying the test throat heat flux by the anticipated Pc ratio between 100% and 75% and taking that ratio to the 0.8 power gives a heat flux value of 29.4 BTU/in2-s. Figure 12 shows that this value also sits between the design and maximum heat flux profiles analyzed.

Based on these analyses, a relatively simple approach to mapping test heat transfer data to the cycle balance was undertaken. Measured test heat flow was increased by 17% to account for the larger cooled expansion ratio on the flight engine conceptual design. The total raw and modified test heat flow was then plotted vs. percent design chamber flow rate and compared against engine cycle heat flow requirements and margins, as shown in Fig. 13. Plotting heat flow vs. flow rate, as opposed to (flow rate)$^{0.8}$, has no appreciable effect on the results. Using unadjusted test heat flow values, the pintle injector can be used to close the conceptual TR202 engine cycle at all power levels with all injector configurations tested. With +17% adjustment for the additional cooling out to an expansion ratio of 30:1 for the conceptual flight nozzle, the injector provides significant cycle power margin.

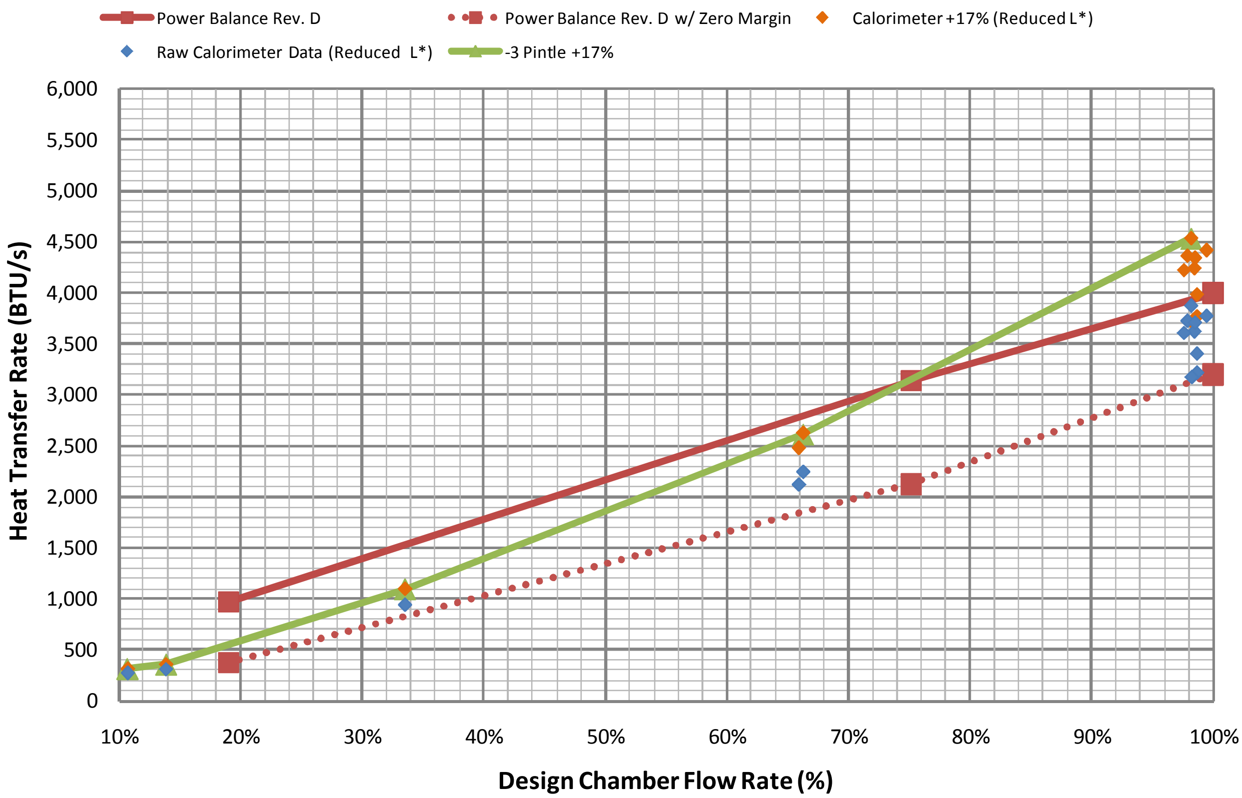

Figure 13. Total Heat Flow vs. Design Propellant Mass Flow Rate

VI. Future Work

The next step in development of the TR202 engine is to procure a linear actuator and attachment hardware to enable continuous throttling of the existing test-bed injector, shown schematically in Fig. 14. The actuator and associated brackets will have to fit within existing test-bed hardware and facility space limits. Unlike the flight design throttling system, which is pressure-balanced, the test-bed actuator will have to move against full combustion chamber pressures. Design and fabrication of the hardware has begun and is scheduled for completion in early 2011.

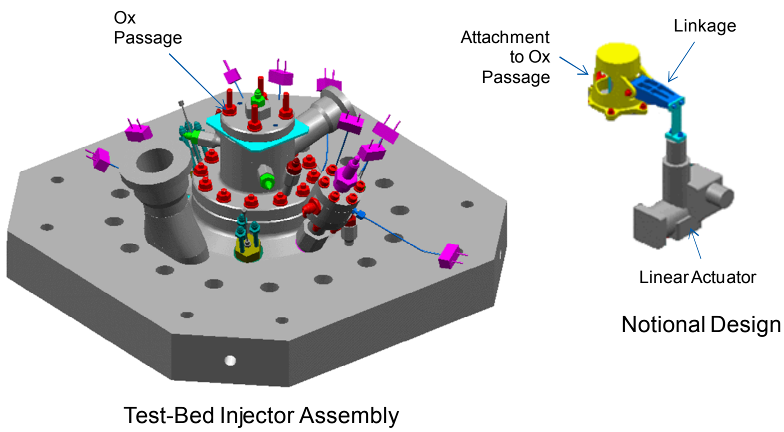

Figure 14. Test-Bed Injector Assembly and Notional Actuator

Conclusion

The Northrop Grumman TR202 test-bed injector was tested at NASA MSFC Test Stand 116 using MSFC ablative and calorimeter chambers, meeting all project objectives. The -3 ablative-tipped pintle combined with the -5 fuel ring was the best performance configuration, achieving a C* efficiency of 98.6% at 75% power. That injector configuration was successfully throttled from 10:1 (75% - 7.5%). Parameterized performance and heat transfer data was gathered for 15 unique injector configurations. Mixture ratios from 5.40 – 6.84 were tested. A reduced length chamber configuration was utilized to gather parametric performance data as well, testing the L' sensitivity of the injector. All configurations at all power levels tested over the 10:1 throttling range showed excellent stability characteristics. The total heat flow and axial heat transfer profile of the TR202 pintle injector are similar to other rocket injectors. The pintle injector can be used to close the conceptual TR202 engine cycle at all power levels with all injector configurations tested.

Acknowledgments

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Topics

- Background
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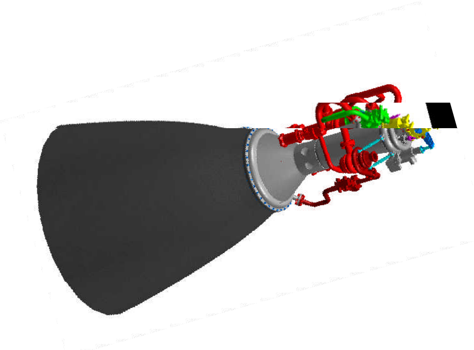
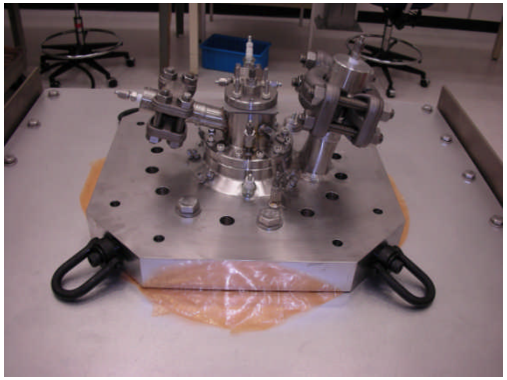
  - Conceptual Engine Design
  - Challenges to Development
  - Phase 2 Objectives
- Hardware
- Testing
- Post-Test Analysis
- Follow-On Activities





Program Overview

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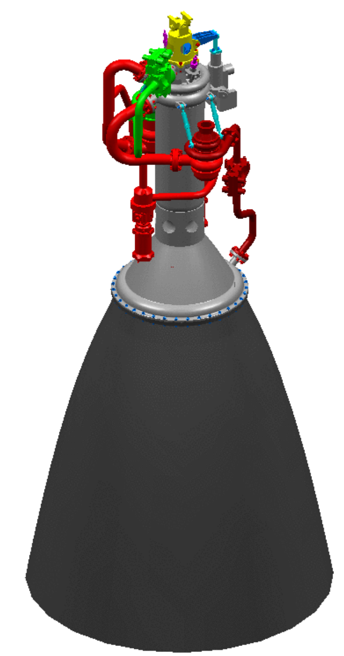
- Research and Development Opportunities in Human and Robotic Technology, BAA on August 31, 2004
  - Requested, “variable thrust rocket engines for landers and ascent vehicles
- NGAS “TR202 Variable Thrust Descent/Ascent Pintle Engine” technology development contract awarded
  - **Phase 1 (May 2005 to May 2006) COMPLETED**
    - NGAS teamed with industry partners and NASA engineers
    - Conceptual Design Review April 5, 2006
  - **Phase 2 (July 2006 – February 2010) COMPLETE**
    - Test-Bed Injector & Igniter Assembly Design (Option 1)
    - Test-Bed Injector Fabrication (Option 2)
    - NGAS teamed with NASA for test planning, test-bed chamber procurement and hot-fire testing (Option 2)
    - Test-Bed Hot-Fire Demonstration of high and stable performance over a 10:1 throttling range (Option 2)
  - **Phase 3 (February 2010 to February 2011) UNDERWAY**
    - Design and procurement of linear actuator hardware to continuously throttle the existing test-bed injector assembly

Design Characteristics of the TR202 Engine

LOX/LH2 closed expander cycle with independent LOX and LH2 turbopump assemblies

Engine Cycle	Closed Expander	Nozzle Area Ratio	200:1
Propellants	LOX/LH2	Length:	88”
Engine Mixture Ratio	6.0 throughout the throttle range	Diameter:	40”
Engine Throttle Range	100% to 18.8%	Weight	280 lbm
Vacuum Thrust Range	8,725 to 1,600 lbf	Reliability Against Catastrophic Failure	0.999967
Engine Out Philosophy for Design	Shut down failed engine, gimbal remaining 3	Vacuum T/W	31 at 100% power 23 at 75% power
Engine Out Power Level	8,680 lbf	Chamber Pressure	700 – 130 psia
Nominal Rated Power Level	6,465 lbf	LOX Pump Discharge Pressure	890 – 270 psia
Vacuum Isp	453 sec to 436 sec	H2 Pump Discharge Pressure	1,785 - 240 psia



- Throttling range at the engine level is dependent on engine out philosophy for the vehicle.
- Pintle injector developed for a throttle range of 10:1

Technical Challenges to Cryogenic Throttling Engine Development

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1. Maintaining acceptable injector performance throughout the throttle range
2. Continuous and deep throttling with cryogenic propellants
3. Maintain combustion stability throughout the throttle range
4. Maintain acceptable engine cooling throughout the throttle range
5. Reliable ignition

6. Maintain MR at acceptable/desired levels throughout the throttle range
7. Balancing injector stiffness requirements with pump performance and pump outlet pressures as the engine throttles down
8. Developing a controller to control at the engine level and at the engine cluster level
9. Avoid pump stall/cavitation at low flow conditions
10. Develop deep throttling turbopump technology (outside historical range)

- Dark blue: Current Scope of TR202 Addresses
- Light blue: Current Scope of TR202 Partially Addresses

Objectives for Phase 2 Option 2

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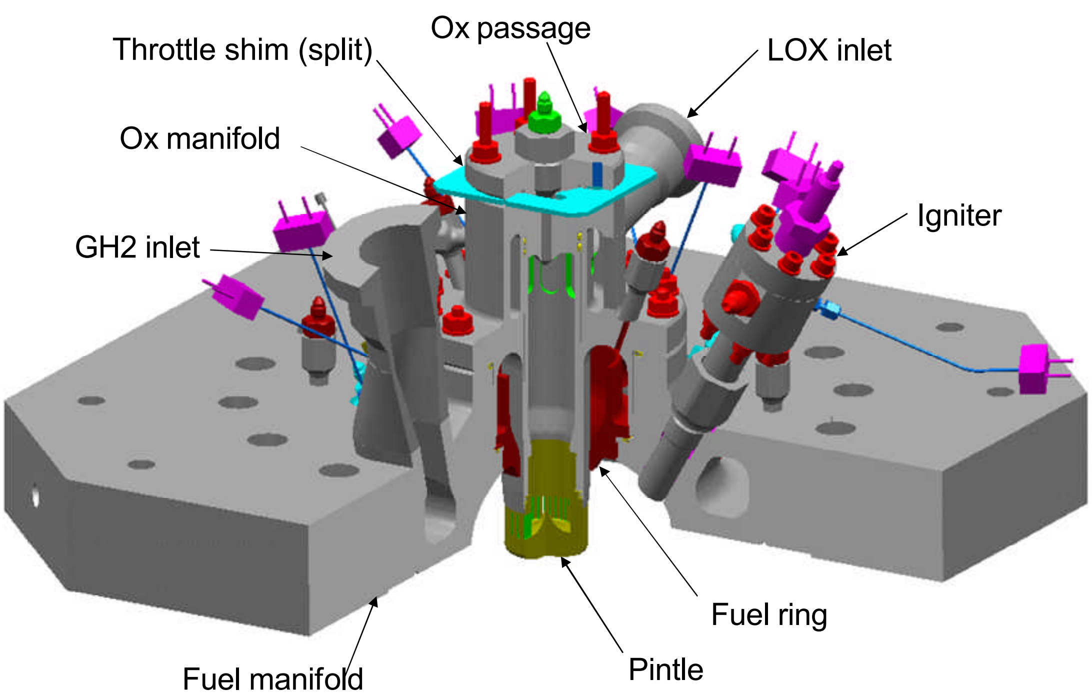
Primary Technical Objectives

- Demonstrate high-performance (>98% C*) within 75%-100% power band
- Demonstrate stable combustion over 10:1 throttle range with high-performing injector
- Measure total chamber heat transfer over 10:1 throttle range
- Demonstrate adequate thermal power availability for engine cycle closure

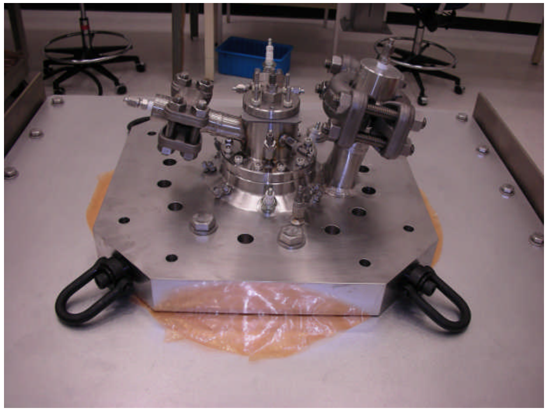
Secondary Technical Objectives

- Demonstrate optimized performance and heat flux characteristics
- Obtain parameterized design data for performance & heat flux
- Measure performance at off-nominal MR
- Obtain L* (L′) parametric performance data

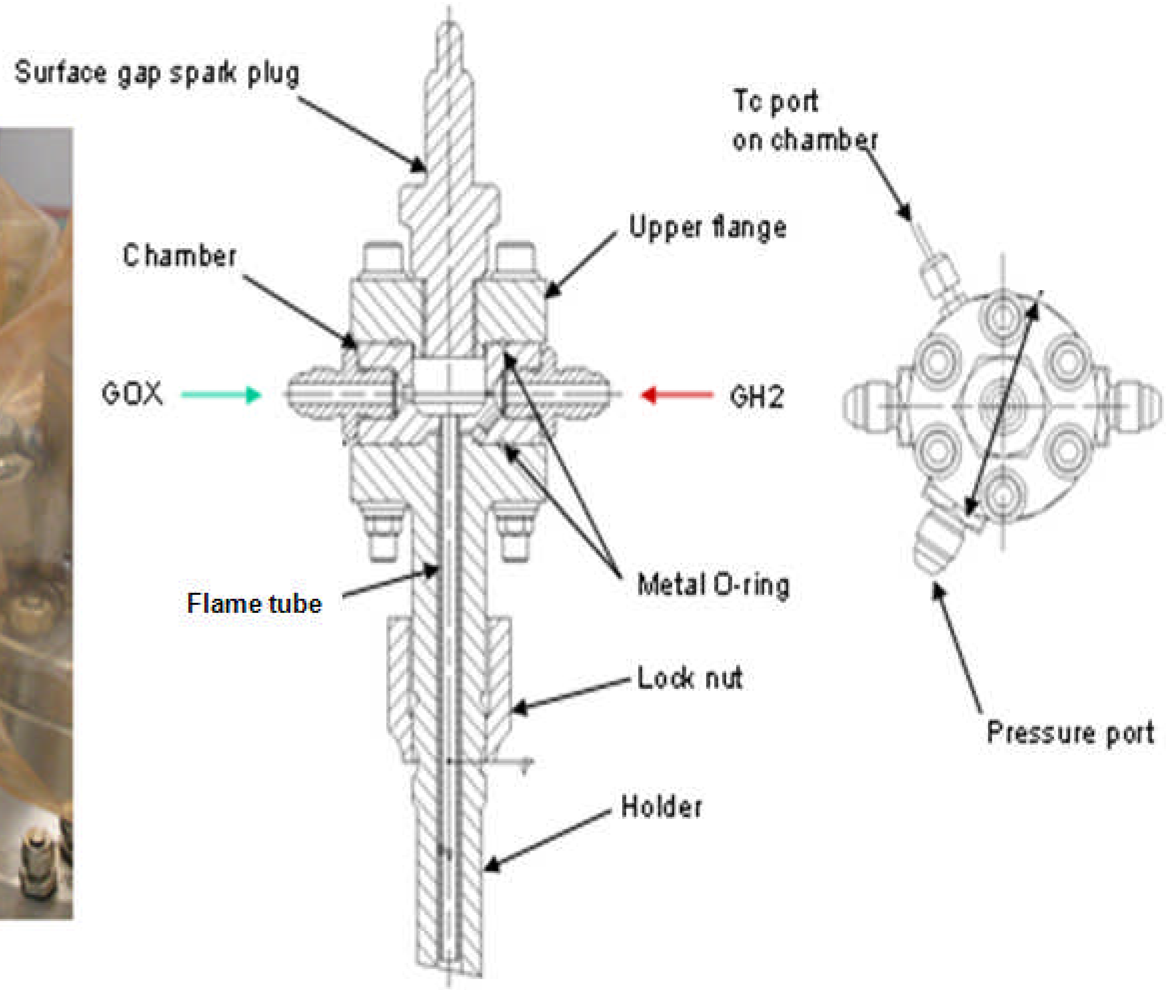
Injector Test-Bed Hardware: Injector



TR202 Test-Bed Injector

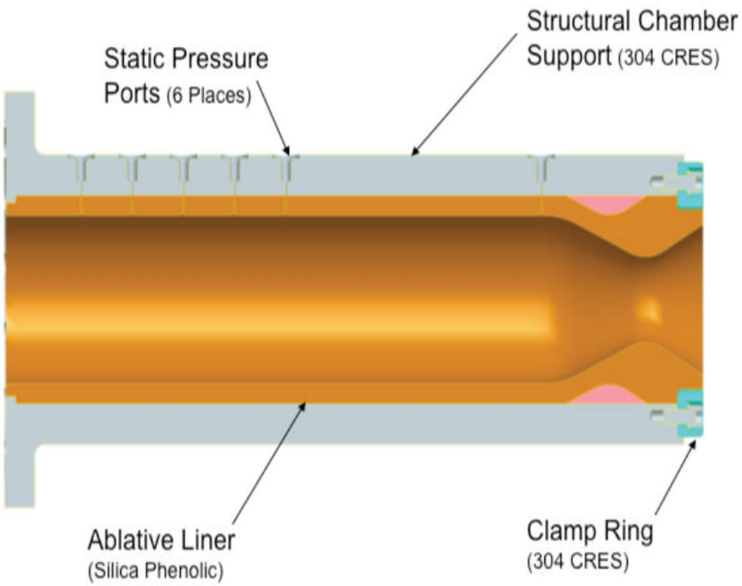


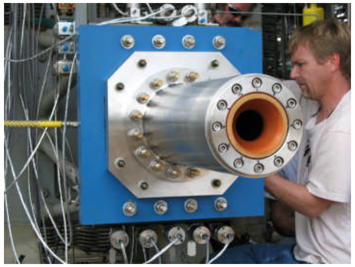
Injector Test-Bed Hardware: Igniter

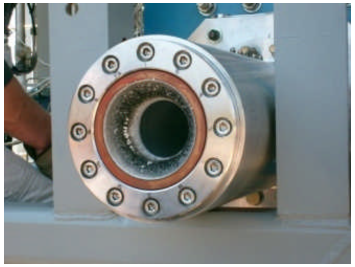


NASA MSFC Chamber Hardware

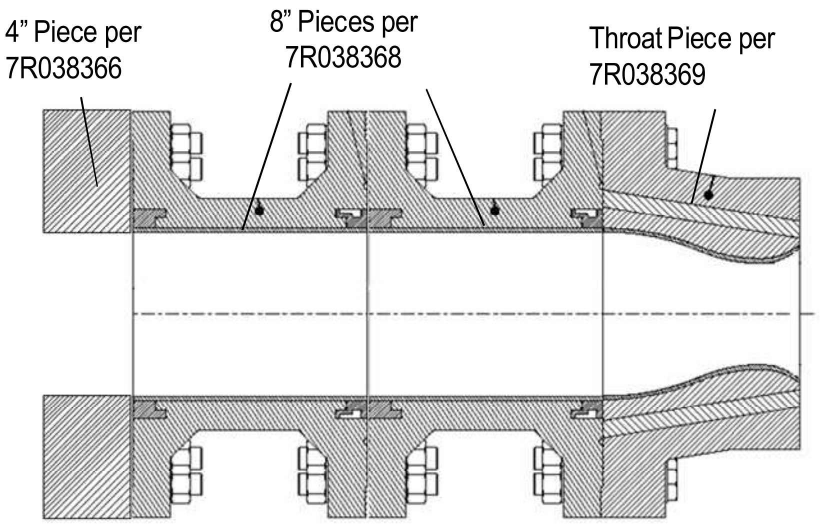
Ablative

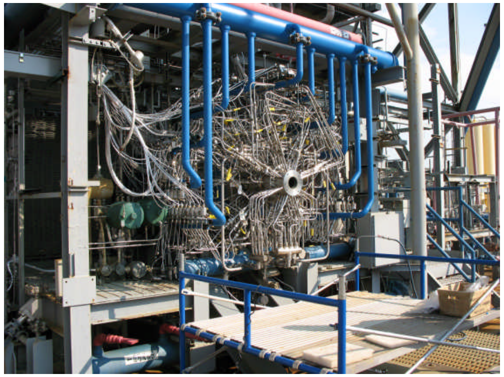






Calorimeter





Test Preparations and Test Series 1: Ablative Chamber

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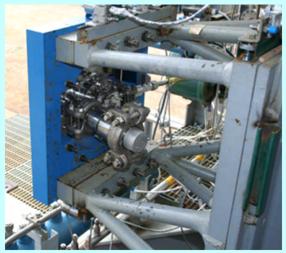
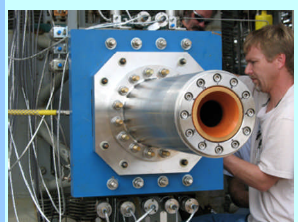
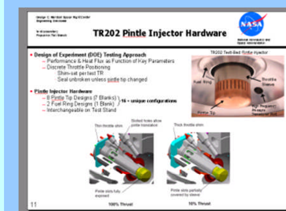
Water Flow Testing

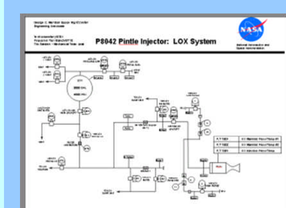
TS116 Assembly/Installation & Cold Flows





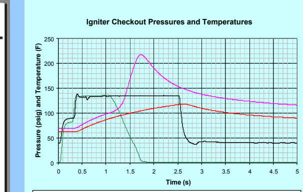
Test Readiness Review





March 19, 2009

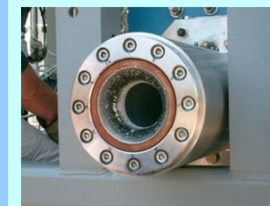
Igniter Checkouts

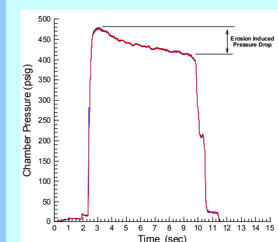


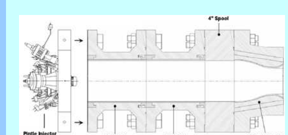
Hot-Fire Testing

April 7, 2009

Data Reviews & Calorimeter Setup







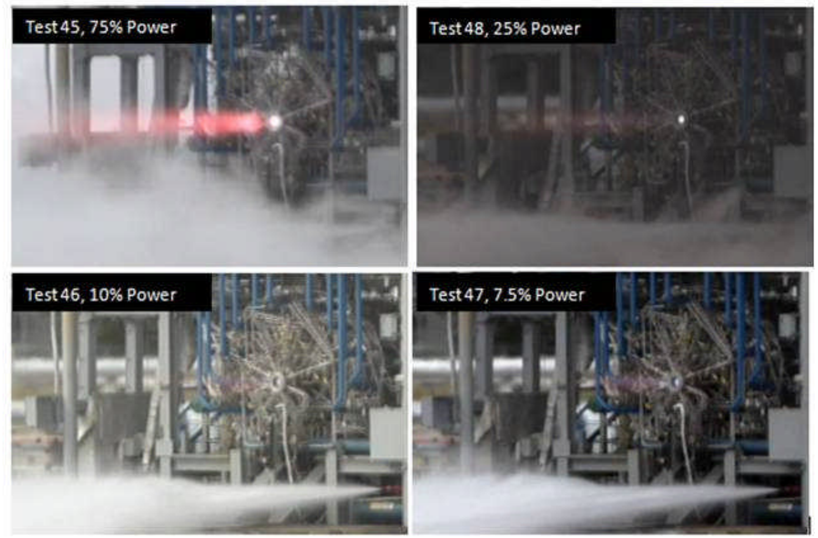
January 2009 → June 2009

Hot-Fire Test Series 1: Ablative Chamber

- 22 tests with 6 oxidizer pintles and 2 fuel rings
- Performance assessment complicated by high ablative throat erosion rates
- Provided risk reduction for calorimeter testing
- Start and shutdown sequences established
- Demonstrated stable combustion over a 10:1 throttle range
- Established relationship between throat stagnation pressure and head-end pressure measurements

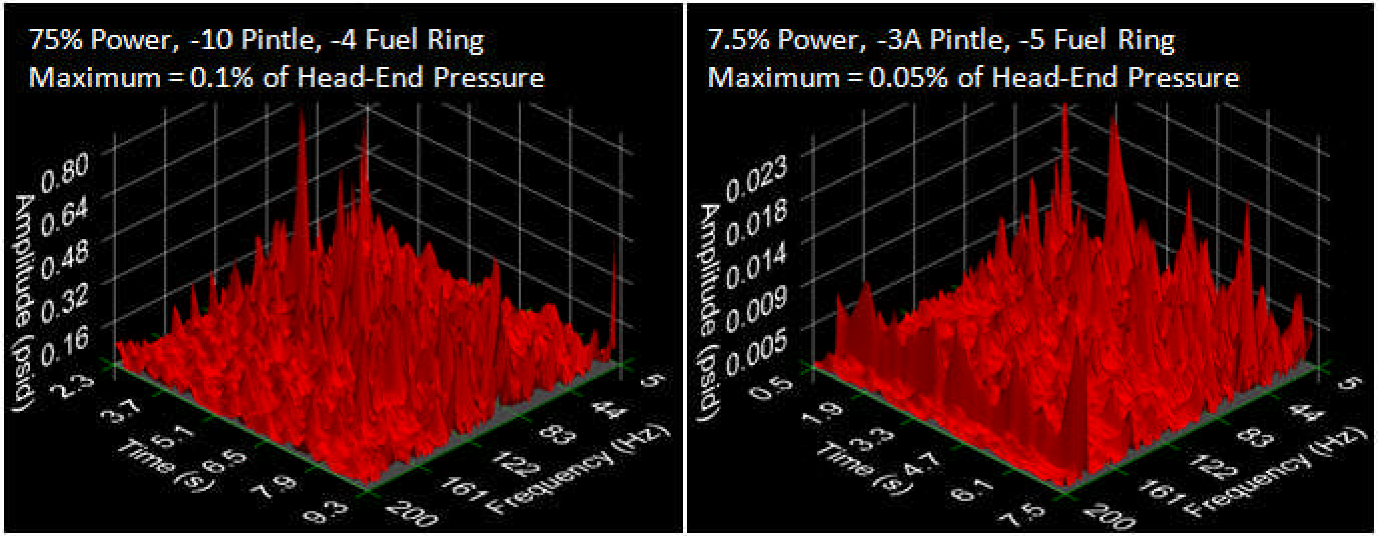
Hot-Fire Test Series 2: Calorimeter Chamber

- 47 tests conducted
  - DOE for performance with LOX/GH2
  - Heat flux characterization
  - L′ (L*) characterization
- 15 unique injector configurations
- 2 different chamber lengths
- Power levels from 75% down to 7.5%
- Mixture ratio range from 5.40 – 6.84
- Demonstrated C* efficiency of 98.6% over an MR range of 5.96-6.4
- No combustion stability issues over the entire envelope tested
- Heat flux adequate to drive a closed expander cycle engine over a 10:1 throttle range



Combustion Stability

- TR202 Injector showed excellent stability characteristics for all tests
  - Encompassed an MR range of 5.4 to 6.84
  - Power levels of 75% to 7.5%
- Maximum oscillation was <1% of measured head-end pressure



Post-Test Analysis: Heat Flux Profile

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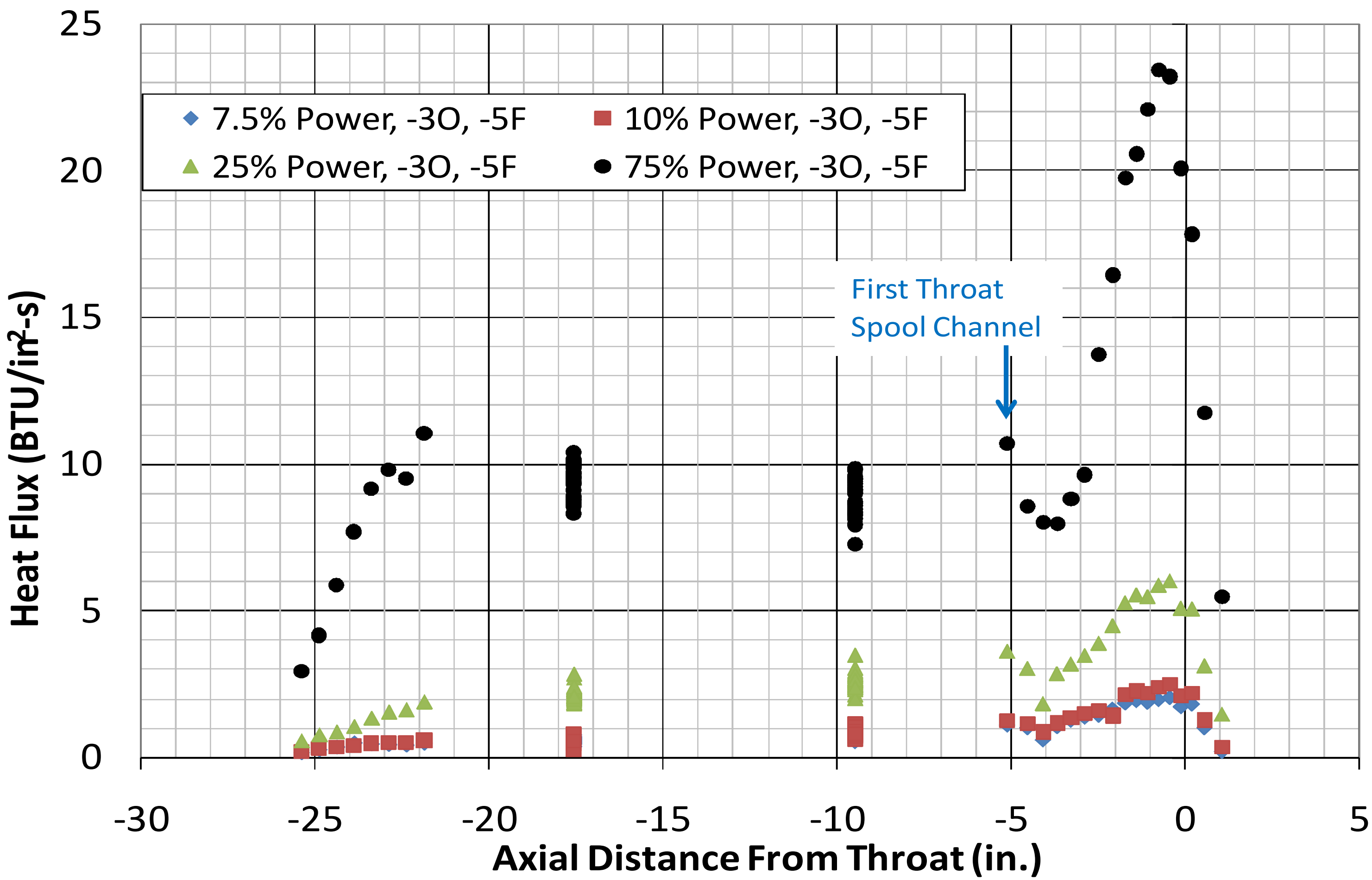

Post-Test Analysis: Total Heat Flux

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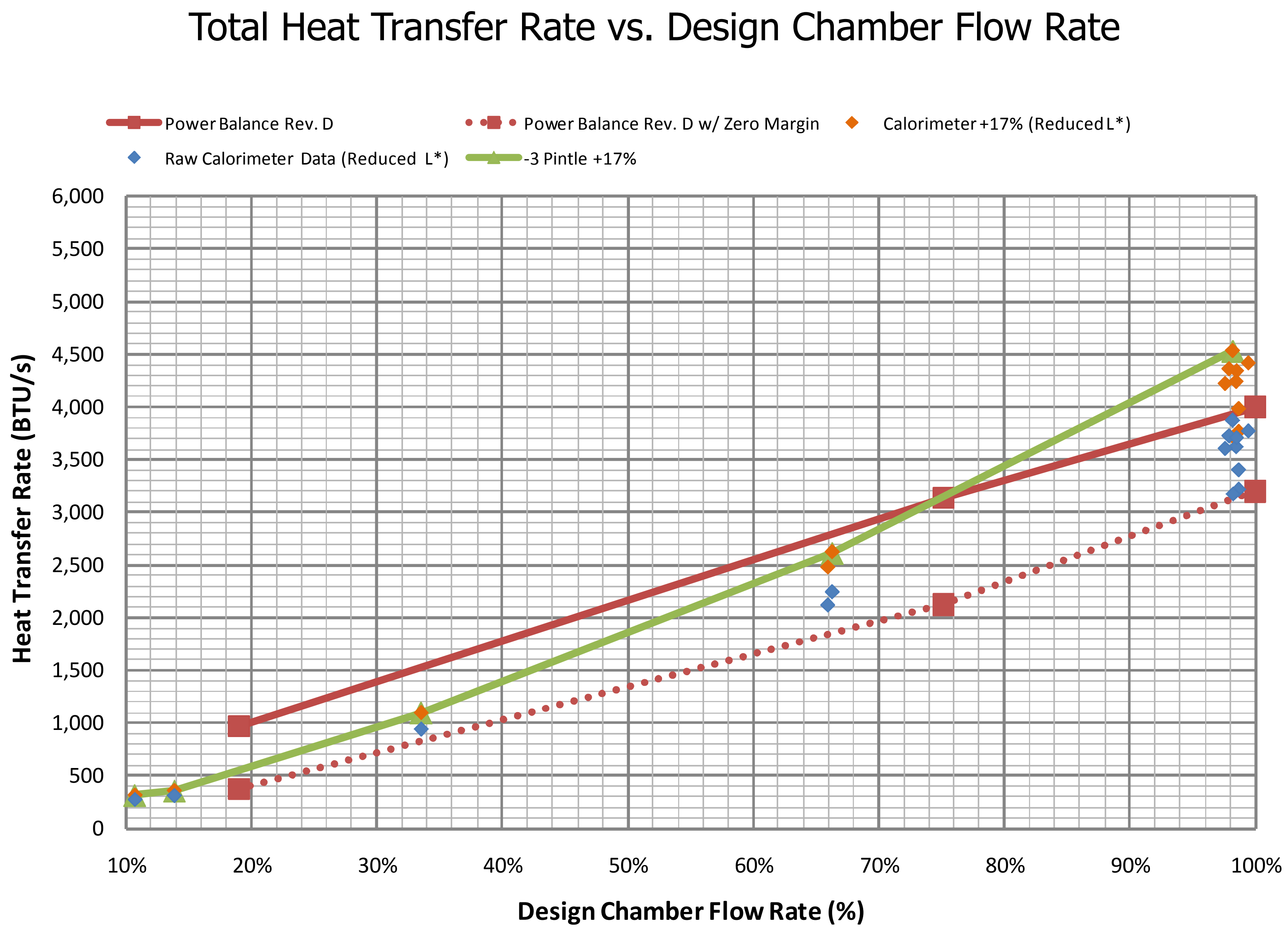

Summary against Objectives

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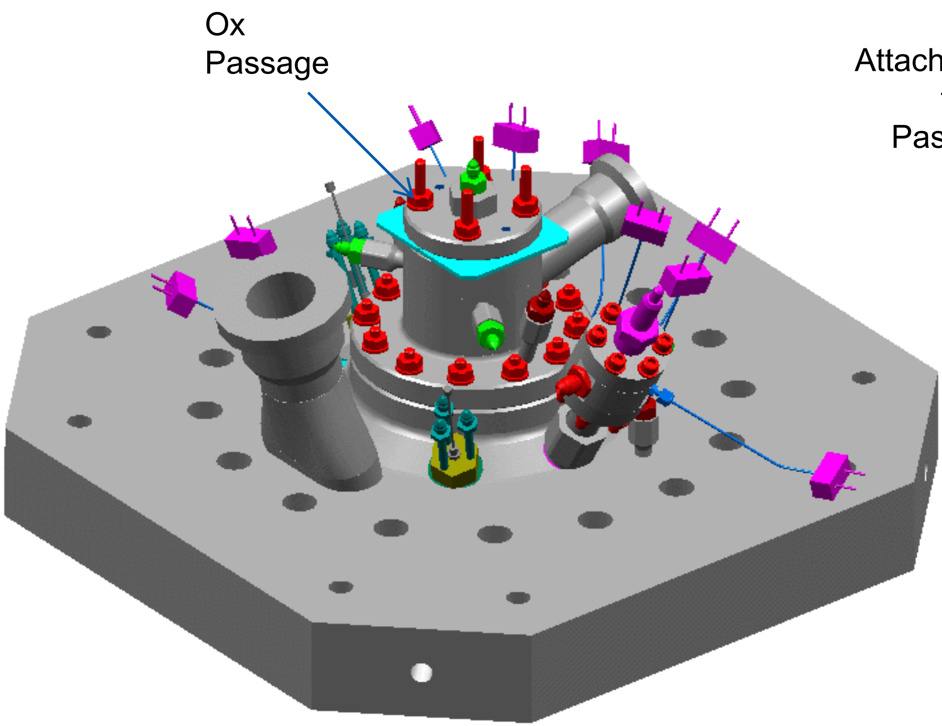
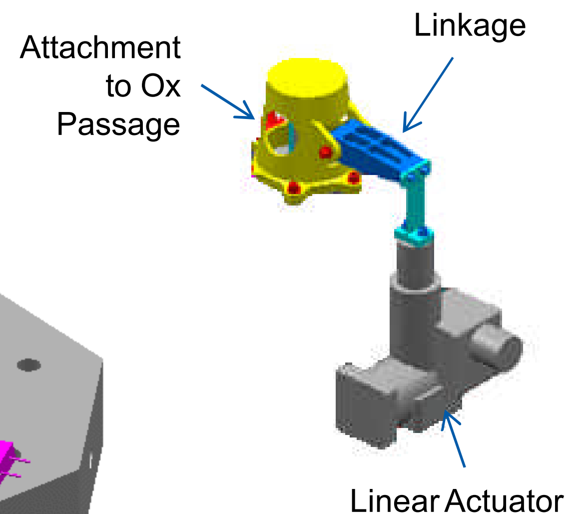
- ✓ **Demonstrated high-performance (>98% C*) within 75%-100% power band**
  - Demonstrated over 5.94 – 6.38 MR range
- ✓ **Demonstrated stable combustion over 10:1 throttle range with high-performing injector**
  - No stability issues in any of the tests conducted
- ✓ **Measured total chamber heat transfer over 10:1 throttle range**
- ✓ **Demonstrated adequate thermal power availability for engine cycle closure**
- ✓ **Demonstrate optimized performance and heat flux characteristics**
- ✓ **Obtained parametric design data for performance & heat flux**
  - 15 unique injector combinations tested
- ✓ **Measured performance at off-nominal MR**
  - Overall MR varied from a minimum 5.4 to a maximum 6.84
- ✓ **Obtained parametric performance data for shorter chamber length (L*)**

Follow-On Activities

- Phase 3 underway to procure a linear actuator and attachment hardware to enable continuous throttling of the existing test-bed injector on MSFC's Lander Integrated Engine Test-Bed



Test-Bed Injector Assembly



Notional Design

Acknowledgements

NORTHROP GRUMMAN

- NASA GRC's PCAD deputy project manager Timothy Smith
- Test Planning, Testing, and Post-Test Analysis: Ron Litchford, John Foote, Gregg Jones, Chip Kopicz, Joel Robinson
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- MSFC Machine Shop: Danny Holland and Danny Lemaster
- Test-Bed Injector Fabrication: Alfred Ramirez, Mike Anderson, Julio Ramirez, Shena Howell
- Chamber Fabrication: Tawna Laughinghouse, Darron Rice, Elizabeth McCollum

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