NASA Technical Memorandum 107470
AIAA–97–2688

Hot Fire Ignition Test With Densified Liquid Hydrogen Using a RL10B–2 Cryogenic H_2/O_2 Rocket Engine

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Prepared for the
33rd Joint Propulsion Conference & Exhibit
cosponsored by AIAA, ASME, SAE, and ASEE
Seattle, Washington, July 6–9, 1997

National Aeronautics and
Space Administration

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Abstract

Enhancements to propellants provide an opportunity to either increase performance of an existing vehicle, or reduce the size of a new vehicle. In the late 1980's the National AeroSpace Plane (NASP) reopened the technology chapter on densified propellants, in particular hydrogen. Since that point in time the NASA Lewis Research Center (LeRC) in Cleveland, Ohio has been leading the way to provide critical research on the production and transfer of densified propellants. On October 4, 1996 NASA LeRC provided another key demonstration towards the advancement of densified propellants as a viable fuel. Successful ignition of an RL10B-2 engine was achieved with near triple point liquid hydrogen.

Introduction

This paper describes the successful ignition test of the cryogenic hydrogen/oxygen RL10B-2 rocket engine using densified liquid hydrogen at near triple point conditions and the potential impact of this test to the aerospace industry via engine and vehicle performance analyses. This demonstration test represents the next step in the advancement of densified propellant technology, the development of an engine that can operate using densified propellants.

Increased demand for launch vehicles for satellite deployment by the private sector and by governments through out the world has generated a fertile yet competitive environment from which advanced aerospace technologies are being incorporated into flight vehicles. One such technology on the verge of being utilized is the use of densified cryogenic propellants such as hydrogen and oxygen. The main advantage of densified cryogenic propellants is the increase in propellant mass fraction. Increased propellant mass fraction means increased payload mass to orbit and more revenue.

Densified cryogenic propellant technology development began in the early 1960's. Opportunities to utilize the technology included the Saturn IV upper stage and the Space Shuttle. However the technology was not considered advanced enough to be incorporated into the design cycle of these vehicles. More recently, research conducted by NASA for the National AeroSpace Plane program advanced slush hydrogen technology to the point that slush was selected as the fuel for the single-stage-to-orbit NASP X-30. These technology advancements focused on large scale slush production, vehicle related component testing, and computer code modeling. However the NASP program was cancelled before full scale engine testing utilizing slush hydrogen could be conducted.

A key issue that has not been adequately addressed is the demonstration of a rocket engine operating with densified propellants. During a recent test program, an opportunity arose to obtain some data using densified hydrogen with a Pratt & Whitney RL10B-2 engine. NASA in cooperation with McDonnell Douglas and Pratt & Whitney conducted two hot fire ignition tests in the Spacecraft Propulsion Research Facility (B2) at NASA Plum Brook Station, Sandusky, Ohio.

The first ignition test was called the nominal test. The inlet conditions and operating procedures were all

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considered "nominal" for the RL10B-2. The test was also the first hot fire of the RL10B-2 in the NASA Plum Brook B2 facility and provided a baseline to demonstrate that the engine and facility were operating properly. The second test that is presented was the densified hydrogen ignition hot fire test. This test was conducted under essentially the same conditions as the nominal test except for the difference in the hydrogen density and temperature.

The results of the two ignition tests are compared in this paper. Engine and vehicle performance analyses are also presented to quantify the potential performance benefits of densified propellants in an overall system.

Symbols

g gravitational constant of earth

Isp delivered specific impulse

m_d dead weight mass

m_L payload mass

m_o initial mass

m_p propellant mass

Δv delta velocity

δ dead weight ratio

λ payload ratio

Subscripts

1 first stage

2 second stage

i first or second stage

Experimental Apparatus

The ignition tests were conducted in the NASA Spacecraft Propulsion Research (B2) Facility. The RL10B-2 rocket engine that was tested is shown mounted inside the B2 facility vacuum chamber in Fig. 1. The B2 facility was designed to test full-scale upper-stage rockets up to 200 000 lb thrust in a simulated space environment. The B2 facility was initially used in the late 1960's and early 1970's to test the Centaur vehicle and is currently being utilized by several U.S. aerospace companies for the

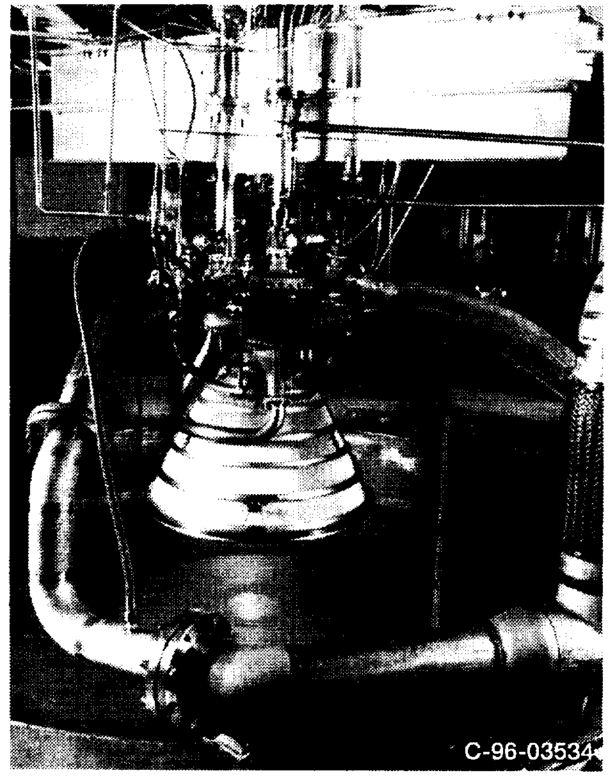


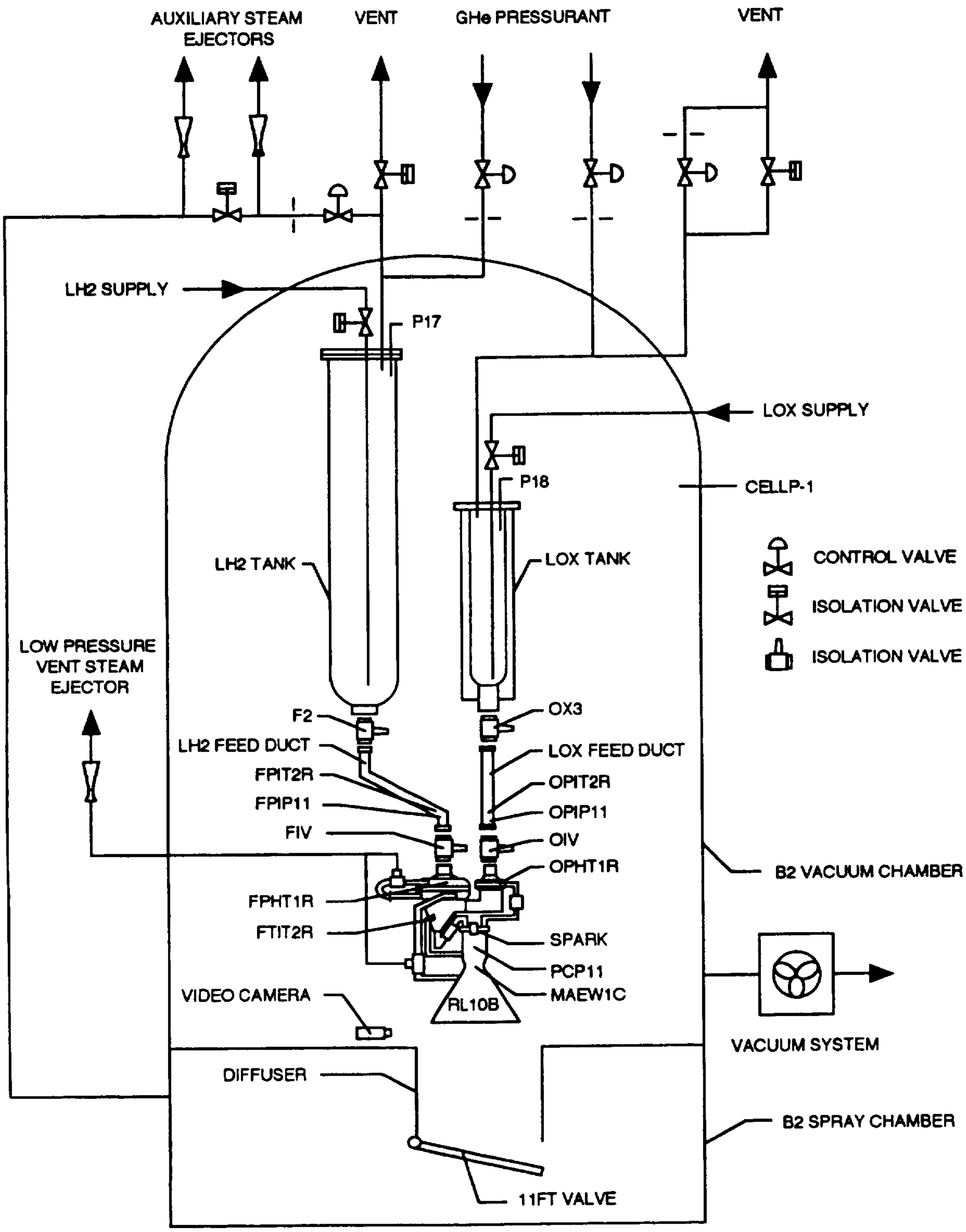
Figure 1 - RL10B-2 Rocket Engine Mounted in the B-2 Facility

development of advanced upper-stage space vehicles and rocket engines.

A simplified drawing of the test configuration is shown in Fig. 2. The vacuum chamber is 38 ft in diameter, 62 ft high, and is constructed out of stainless steel. A mechanical vacuum pumping system is used to evacuate the vacuum chamber. It consists of one 28 100 cfm blower (first stage), two 1875 cfm blowers (second stage), and four 728 cfm mechanical vacuum pumps (third stage).

The rocket engine exhaust from the ignition tests was directed into the spray chamber located below the vacuum test chamber. The vacuum test chamber and spray chamber are connected via an 11 ft diameter, 37 ft long inconel diffuser duct. The vacuum test chamber and the spray chamber are isolated from one another by a 11 ft valve at the bottom of the diffuser. The 420 000 ft^3 spray chamber was filled with 70 ft of water prior to testing and was evacuated for the testing using steam ejectors.

Inside the vacuum test chamber were mounted a 250 gal liquid hydrogen (LH_2) test tank, a 40 gal liquid oxygen (LOX) test tank, propellant feed ducts, and an



DRAWING NOT TO SCALE

Figure 2 - Simplified RL10B Densified Hydrogen Ignition Test Configuration

RL10B-2 rocket engine. LH_2 and LOX were supplied to their respective test tanks via a 14 000 gal LH_2 dewar and a 12 000 gal LOX dewar located outside of the B2 test building. Gaseous helium (GHe) pressurant gas was supplied to each test tank via separate pressurant gas control systems supplied by a 70 000 scf GHe tube trailer. For the densified hydrogen ignition test steam ejectors were utilized to vacuum pump on the hydrogen test tank to densify the hydrogen.

The LH_2 test tank was isolated from the LH_2 feed duct by a shutoff valve mounted below the tank (F2). The LOX test tank was also isolated from the LOX feed duct by a shutoff valve (OX3). The LH_2 feed duct was constructed out of 2.5 in. diameter, 0.065 in. wall thickness, 300 series stainless steel tubing. The LOX feed duct was constructed out of 3 in. diameter, 0.065 in. wall thickness, 300 series stainless steel tubing.

A detailed propellant flow schematic of the RL10B-2 is shown in Fig. 3. The two-stage centrifugal fuel turbopump is isolated from the fuel feed duct with the fuel pump inlet shutoff valve (FIV). The single-stage centrifugal oxidizer pump is isolated from the LOX feed duct with the oxidizer pump inlet shutoff valve (OIV). The fuel pump is chilled down prior to ignition by flowing LH_2 through the pump and discharging the propellant out the fuel pump interstage cooldown valve and the fuel pump discharge cooldown valve and into the low pressure steam ejector vent. For the two tests conducted in this report the steam ejectors were not activated and the fuel was discharged to ambient pressure. To cooldown the LOX pump, LOX was discharged through the injector into the B2 vacuum test chamber. For further details on the characteristics of the RL10B-2 the reader is referred to Ref. 1.

Instrumentation and Data Systems

Strain-gage pressure transducers with an accuracy of ±1 percent of full scale were used to measure several parameters. The LH_2 test dewar pressure (P17), LOX test dewar pressure (P18), fuel pump inlet pressure (FPIP11), and combustion chamber pressure (PCP11) were measured with separate 0 to 50 psia transducers. The oxidizer pump inlet pressure (OPIP11) was measured with a 0 to 100 psia transducer. Vacuum test chamber pressure

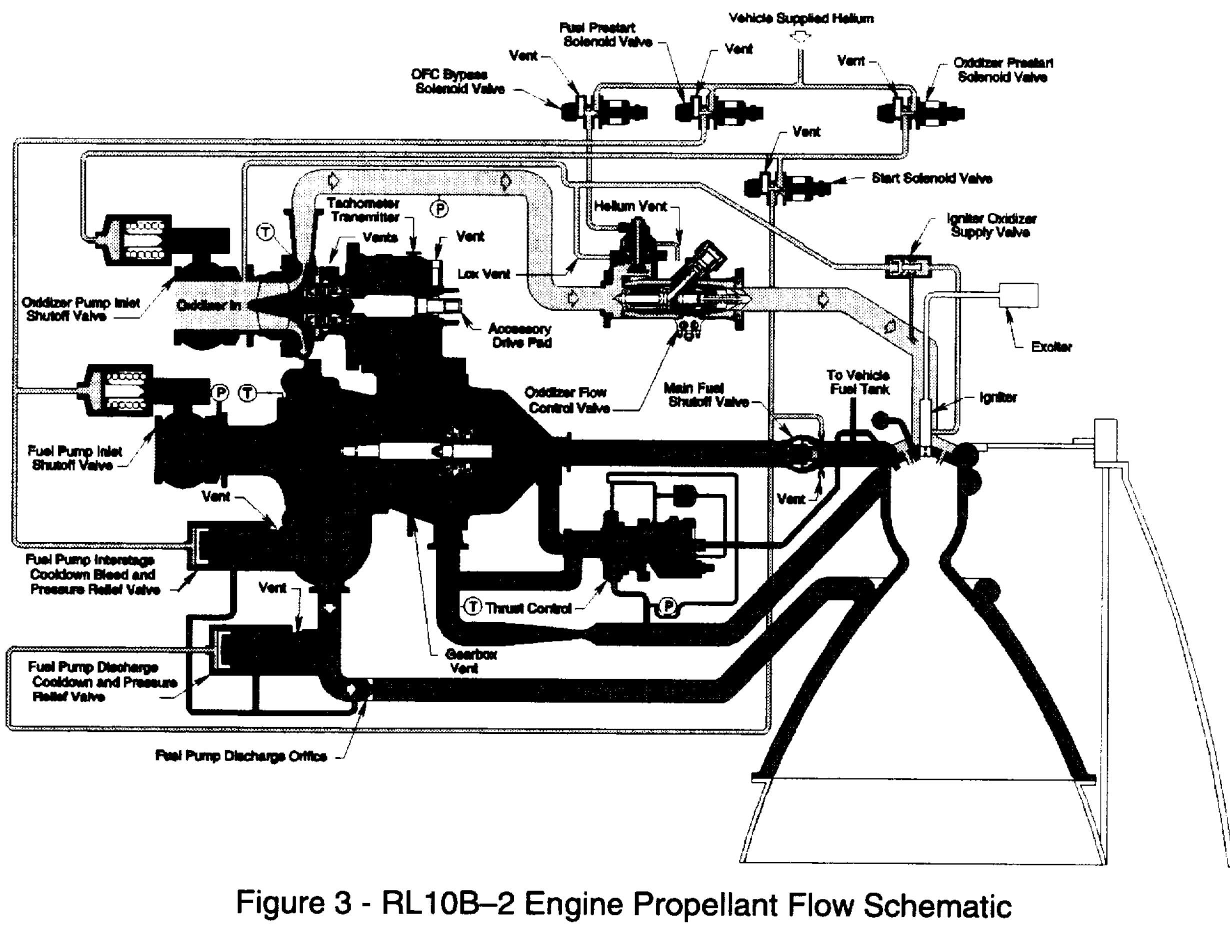


Figure 3 - RL10B–2 Engine Propellant Flow Schematic

(CELLP1) was measured using a Convection gage with a range of 1×10^{-4} to 990 torr and an accuracy of 2 significant digits.

The fuel pump inlet fluid temperature (FPIT2R) and the oxidizer pump inlet fluid temperature (OPIT2R) were measured using silicon diode temperature sensor probes that were inserted into the respective feed ducts upstream of the inlet valves. The accuracy of the silicon diode temperature sensors was ±0.2 °R. The fuel pump housing temperature (FPHT1R) and the oxidizer pump housing temperature (OPHT1R) were measured with platinum resistance thermometers with an accuracy of ±3.0 °R. The pump housing temperature sensors were the same sensors that are used for flight operations. The spark plug gap voltage (SPARK) was a direct measurement of the spark voltage in the ignitor. The accuracy of the measured voltage is within 0.1 V.

Two types of data systems were utilized. The ESCORT data recording system recorded 188 channels of data at 1 Hz and 64 channels of data at 10 Hz. The Masscomp data system recorded 50 channels of data at a recording rate of 1000 Hz. The spark plug gap voltage and the combustion chamber pressure were recorded at 1000 Hz.

Test Procedure

The operating procedures for hot firing an RL10B-2 rocket engine are fairly complex and involve many detailed steps. The intent of this section is to give an overview of the significant procedural steps used for this testing and the order in which the steps occurred. The two sets of test procedures are for the densified hydrogen test and the nominal test. The only major difference between the two tests is the density of the hydrogen.

Densified Hydrogen Test

The first step of the test procedure was to evacuate the vacuum chamber to ~2 torr to minimize the heat leak into the test tanks using the mechanical vacuum system. The spray chamber was evacuated to ~45 torr using both auxiliary steam ejector trains and then maintained at this pressure with only one train. The propellants were then loaded into the test tanks. The liquid hydrogen test tank was filled with ~239 gal of liquid hydrogen and the liquid oxygen test tank was filled with 35 gal of LOX. Both test tank shutoff valves (F1 and OX3) where open during the fill and the inlet shutoff valves (FIV, OIV) were closed thus allowing propellants to fill the feed ducts.

The propellants were then conditioned to the appropriate saturated starting conditions as required to conduct the test. The LOX tank pressure was controlled to 22.9 psia and the propellant was allowed to warm to the saturation temperature based on tank pressure. The liquid hydrogen was conditioned by vacuum pumping on the LH_2 test tank using the other train of the auxiliary steam ejectors. Reducing the vapor pressure caused evaporative cooling of the liquid hydrogen and lowered the temperature of the liquid hydrogen. The final vapor pressure of the liquid hydrogen was 1.94 psia.

Once the propellants were conditioned, video recording of the exit plane of the engine bell was initiated. The vacuum chamber pressure was then raised to 45 torr to equalize with the spray chamber pressure. The 11 ft valve was then opened. The propellant tanks were then pressurized to give the proper pump inlet conditions prior to engine pre-start. The hydrogen tank was ramped to 24.4 psia and the LOX tank was ramped to 43.8 psia. Helium pressurant gas was used for both propellants. ESCORT data recording was initiated.

The sequencer was then activated and the countdown started at T-60.0 sec. The sequencer automatically controlled all engine valves during the ignition test per a pre-programmed set of sequential instructions. These instructions included the following steps. At T-50.0 sec the fuel pump pre-start signal was initiated. At T-45.0 sec the LOX pump pre-start signal was initiated. The engine pre-start procedures involve flowing propellants through their respective pumps to chilldown the pumps and prevent cavitation during startup. The fuel pre-start flow was directed out of the engine to the low-pressure vent. The LOX pre-start flow was dumped directly into the vacuum chamber.

At T-2.0 sec the Masscomp data acquisition system was activated. At T-0 sec the engine start signal was given. The ignitor signal was activated at T+0.082 sec and de-activated at T+0.550 sec. Engine shutdown occurred at T+1.0 sec. At engine shutdown propellant flow was shutoff from the engine.

Nominal Test

For the nominal ignition test the liquid hydrogen test tank was filled with approximately 135 gal of LH_2 and 34 gal of LOX were loaded into the LOX test tank. The propellants were then conditioned to the appropriate saturated starting conditions. For the nominal ignition test the LH_2 tank pressure was controlled to 17.5 psia and the LOX tank pressure to 28.0 psia while the propellants saturated at these pressures.

At this point similar procedures between the nominal test and the densified test were conducted. Minor differences between the two tests include pressurizing the LH_2

tank to 29.8 psia and the LOX tank to 45.5 psia for the nominal test. Also, the ignitor was activated at T+0.270 sec and de-activated at T+0.870 sec for the nominal test.

The minor differences in LH_2 tank pressurization levels and the ignition times between the two tests were a result of increasing the probability of ignition which was the ultimate goal of the test. Since there was only one opportunity to conduct the densified hydrogen test it was important to generate the best possible conditions to ignite. The reduced LH_2 tank pressure was predicted to cause the mixture ratio of oxygen to hydrogen to increase slightly thus increasing the amount of oxygen available for ignition. The ignitor was activated sooner in the densified hydrogen test to increase the number of sparks early in the sequence, thus also increasing the possibility of ignition, in case the ignition window was earlier in the start sequence as a result of increased hydrogen mass in the combustion chamber due to densifying the hydrogen.

Results and Discussion

Table I shows the test conditions at engine start for both the densified hydrogen propellant ignition test and the nominal ignition test.

The liquid hydrogen density at the inlet of the fuel pump for the densified ignition test was 4.738 lbm/ft^3 and for the nominal test the density was 4.317 lbm/ft^3. This is a 9.8 percent increase in liquid hydrogen density over the nominal run.

Once again, the purpose of the testing was to demonstrate the ignition of densified hydrogen in the RL10B-2. The primary indicator of a successful ignition is given by a rapid increase in combustion chamber pressure. The secondary indicator is visual observation of the flame exiting from the bell of the engine. Figures 4 and 5 show the combustion chamber pressure and the spark gap discharge voltage for the densified hydrogen ignition test and the nominal ignition test, respectively. Time zero is when the engine start signal is initiated.

In Fig. 4, the densified test, the combustion chamber pressure starts at 0.9 psia which is the B2 facility vacuum chamber pressure into which the engine ignites. At ~75 msec propellant begins to flow into the combustion chamber causing an increase in combustion chamber pressure to about 5 psia. A rapid increase in combustion chamber pressure occurs at 244 msec indicating a successful ignition. The combustion chamber pressure at ignition peaks at 33 psia in about 4 msec and then decays down to about 15 psia for the remainder of the test.

Figure 4 also shows the spark gap voltage being discharged by the spark plug ignitor. The ignitor was programmed to begin discharging a spark at 82 msec instead of the nominal 280 msec to increase the probability of ignition. Each 25 msec decay of the spark gap voltage indicates that a spark has been generated by the ignitor. The spark gap voltage shown in Fig. 4 indicates that combustion began on the 7th spark.

The nominal ignition test combustion chamber pressure plotted in Fig. 5 shows a rapid pressure increase at 281 msec. This indicates a successful ignition on the first spark 1 msec after the ignitor was discharged. The combustion chamber pressure peaked at 14 psia in about 3 msec then steadied out at ~13 psia for about 300 msec before climbing slightly to 15 psia at the end of the test.

Visual observations of the exit plane of the rocket engine bell with a video camera verified successful engine ignition in both tests. The video showed an exhaust plume from the nozzle exit in both the densified hydrogen test and the nominal test. If the engine had not ignited, a foggy plume of propellants would have been seen flowing from the bell instead of the flame.

Engine Performance Analysis

An analysis was conducted to illustrate that the potential effects of densified propellants on the specific impulse of the RL10B-2 engine are minimal, and to provide input for the vehicle analysis. The calculations were theoretical

TABLE I.—IGNITION TEST START CONDITIONS

Parameter description	Parameter name	Densified hydrogen ignition test	Nominal ignition test
Fuel pump housing temperature	FPHT1R	58.4 R	62.8 R
Oxidizer pump housing temperature	OPHT1R	176.0 R	184.2 R
Fuel pump inlet temperature	FPIT2R	27.5 R	39.1 R
Fuel pump inlet pressure	FPIP11	24.1 psia	29.3 psia
Oxidizer pump inlet temperature	OPIT2R	171.8 R	177.4 R
Oxidizer pump inlet pressure	OPIP11	47.5 psia	46.8 psia
External throat tube temperature	MAEW1C	387 F	422 R
Fuel turbine inlet temperature	FTIT2R	376 R	493 R

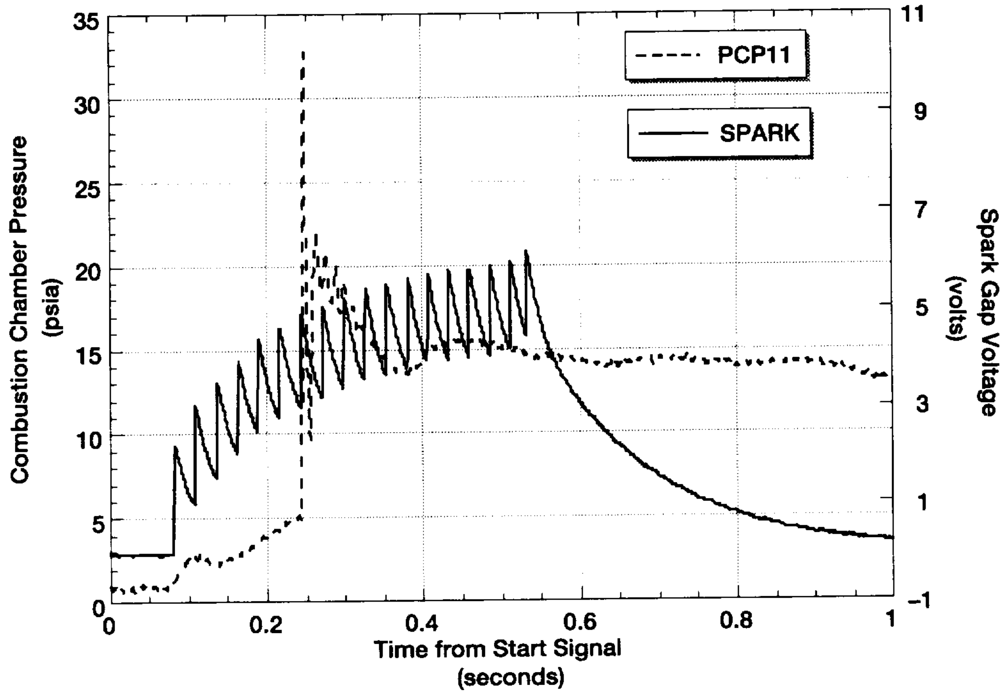


Figure 4 - Combustion Chamber Pressure and Spark Gap Voltage
Densified Hydrogen RL10B Ignition Test

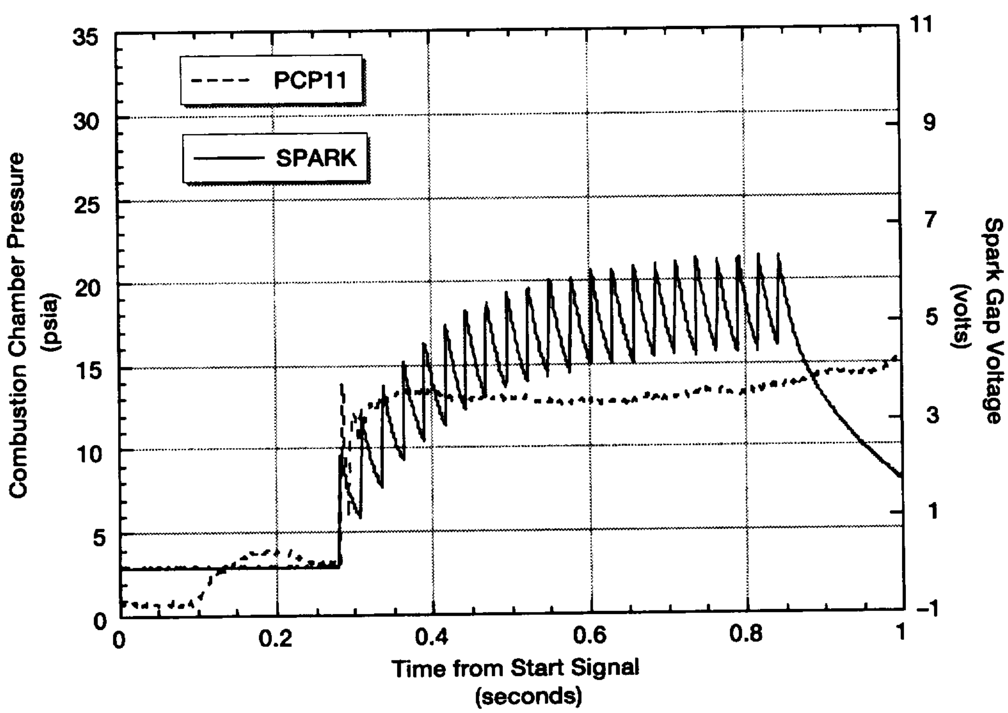


Figure 5 - Combustion Chamber Pressure and Spark Gap Voltage
Nominal RL10B Ignition Test

and were made using CET89, a rocket engine performance prediction code described in Ref. 2. The two main parameters that were varied in this analysis were the enthalpy of the propellants at the inlet to the injector and the oxidizer to fuel mixture ratio (O/F). For the densified cases the propellant enthalpy at the injector inlet was assumed to differ from the nominal case by the same difference calculated between the storage enthalpies of the densified cases and the nominal case. The propellant densification options considered were triple point liquid hydrogen (TP LH_2B), 50 percent solid slush hydrogen (SLH_2), and oxygen at 140 °R. Because the oxidizer and fuel tank volumes were fixed for the vehicle analysis, it was also assumed that the O/F for the densified cases were adjusted from the nominal 6.0 by an amount proportional to the change in density of the densified propellants from the nominal case.

The results of the analysis are given in Table II. Case 1 is the nominal RL10B-2 engine configuration with an O/F of 6.0. For this case the theoretical specific impulse was 488.4 sec. The nominal delivered specific impulse for the RL10B-2 is 466.5.[1] The ratio of delivered to theoretical specific impulse for the nominal case was used to calculate the delivered specific impulse for the densified cases. The results of the densified cases show that the effect on engine performance is small. In fact, by densifying both the hydrogen and the oxygen, the engine performance essentially remains the same.

Vehicle Performance Analysis

The benefit of using densified propellants on a launch vehicle is to increase the propellant mass fraction of the vehicle which translates into an increase in payload mass to orbit. This increase in payload mass is calculated here with a simplified launch vehicle performance analysis using the two-stage-to-orbit rocket equation. The performance will be measured in terms of additional pounds of payload to low earth orbit (LEO) that can be obtained with densified propellants as compared to a baseline vehicle which uses normal boiling point propellants. The equations used to calculate the increase in payload mass are taken from Ref. 3 and are shown in Appendix A.

The baseline vehicle used in this analysis is a two stage rocket. The first stage uses the RS-27 RP-1/liquid oxygen engine with an average Isp of 264 sec. The second stage is a liquid hydrogen/liquid oxygen rocket powered by the RL10B-2 engine which has a delivered Isp of 466.5 sec at a mixture ratio of 6.0. This baseline vehicle is designed such that the total vehicle weight, propellant masses, and payload mass are averages of the Atlas/Centaur and Delta III launch vehicles which were obtained from Ref. 4. The baseline does not include any increase in vehicle weight for hardware, such as a recirculation manifold, required to integrate the vehicle with the GSE densification unit.

Table III shows the results of the launch vehicle performance calculations. The following six cases are analyzed; (1) baseline, (2) triple point liquid hydrogen (TP LH_2), (3) 50 percent solid slush hydrogen, (4) densified oxygen at 140 °R, (5) triple point liquid hydrogen and densified liquid oxygen at 140 °R, and (6) 50 percent solid slush hydrogen and densified liquid oxygen at 140 °R. The table gives the mass breakdown of the vehicle for each case. For the densified propellant cases, the vehicle tanks were fixed at the baseline tank volume but loaded with additional propellants. The dead weight mass for both the first and second stage were estimated. The final delta velocity for each case was held constant at 30 882 ft/sec which is approximately 20 percent greater than the orbital velocity required to get to LEO. The higher final orbital velocity used in this simplified analysis is an attempt to compensate for the affects of gravity and drag forces which are not explicitly entered into the rocket equation calculations.

The results of the analysis show that the baseline vehicle can place 15 000 lb of payload into LEO using normal boiling point hydrogen and oxygen. When triple point liquid hydrogen is used the payload mass to orbit

TABLE II.—RL10B-2 ANALYTICAL ENGINE PERFORMANCE RESULTS WITH DENSIFIED PROPELLANTS

Parameter	Units	1 Nominal	2 TP LH2	3 Hydrogen 50 percent solids	4 Oxygen 140 R	5 TP LH2 140 R LOX	6 50 percent SLH2 140 R LOX
Hydrogen density	lbm/ft³	4.42	4.80	5.08	4.42	4.80	5.08
Oxygen density	lbm/ft³	71.2	71.2	71.2	75.0	75.0	75.0
O/F		6	5.5	5.2	6.3	5.8	5.5
Theoretical specific impulse (Ivac)	seconds	488.4	489.2	489.3	487.2	488.5	488.8
Delivered specific impulse (Isp)	seconds	466.5	467.3	467.4	465.4	466.6	466.9

TABLE III.—LAUNCH VEHICLE PERFORMANCE CALCULATIONS FOR A HYDROGEN/OXYGEN FUELED UPPER STAGE USING DENSIFIED PROPELLANTS

Parameter	Symbol	Units	1 Baseline	2 TP LH2	3 Hydrogen 50 percent solids	4 Oxygen 140 R	5 TP LH2 140 R LOX	6 50 percent SLH2 140 R LOX
First stage								
Initial mass	m_{01}	lbm	554166	554858	555335	556472	557189	557676
Propellant mass	m_{p1}		450000	450000	450000	450000	450000	450000
Dead weight mass (estimated)	m_{d1}		45000	45000	45000	45000	45000	45000
Payload mass	m_{L1}		59166	59858	60335	61472	62189	62676
Engine performance	Isp_1	seconds	264	264	264	264	264	264
Second stage								
Initial mass	m_{02}	lbm	59166	59858	60335	61472	62189	62676
Propellant mass hydrogen			5666	6153	6510	5666	6153	6510
Propellant mass oxygen			34000	34000	34000	35802	35802	35802
Propellant mass total	m_{p2}		39666	40153	40510	41468	41955	42312
Dead weight mass (estimated)	m_{d2}		4500	4500	4500	4500	4500	4500
Payload mass	m_{L2}		15000	15205	15325	15504	15734	15864
Engine performance	Isp_2		466.5	467.3	467.4	465.4	466.6	466.9
Engine mixture ratio	O/F	seconds	6.0	5.5	5.2	6.3	5.8	5.5
Delta velocity	dV	ft/sec	30882	30882	30882	30882	30882	30882
Increase in payload mass		lbm	0	205	325	504	734	864

increases by 205 lb of payload. When 50 percent slush hydrogen is used the payload mass increases by 325 lb. When densified oxygen at 140 °R is used the payload mass increases to 504 lb. When both triple point liquid hydrogen and 140 °R liquid oxygen are used together an additional 734 lb of payload can be obtained over the baseline vehicle. Finally, when 50 percent slush hydrogen and 140 °R liquid oxygen are used the payload gain is 864 lb.

It is important to point out that in case 4, 140 °R liquid oxygen, the O/F climbs to 6.3. The increased mixture ratio raises concerns about higher combustion temperatures and excess oxygen near the wall which can reduce the lifetime of the combustion chamber. The analysis presented here shows that if hydrogen, densified to the triple point, is used in conjunction with densified oxygen the O/F stays below the nominal and thermal damage to the engine is no longer an issue.

Concluding Remarks

Densified propellants offer vehicle manufacturers more payload flexibility and weight margin than other advanced technologies for the same amount of investment. By subcooling LH_2 and LOX to near triple point conditions, a substantial increase in vehicle performance can be realized without the 2 phase fluid complexities of a slush mixture. The vehicle performance analysis presented in this report indicates a payload gain of up to 5 percent (734 lb) if both densified LH_2 and LOX are used. While this number does not account for the weight penalty of incorporating a recirculation manifold and disconnect for the densification GSE, it still represents the potential for significant payload gains with only minor tank redesign and a nonrecurring investment in launch pad ground support equipment.

In addition the test results presented in this paper demonstrate that an aerospace industry standard-the RL10 rocket engine can be ignited with densified LH_2 with no hardware changes. Additional testing is required to optimize the ignition sequence for both densified LH_2 and LOX, but this successful ignition demonstrates a vital step in bringing densified propellants to a technology readiness level of 6 (system/subsystem model or prototype demonstration in a relevant environment).

Acknowledgments

The authors would like to thank Mr. Michael L. Meyer of NASA Lewis for helping to conduct the engine analysis. The authors would also like to acknowledge and thank Mr. Jim Brown of Pratt & Whitney and Mr. Mark Berger of McDonnell Douglas Aerospace for allowing us to utilize the RL10B-2 rocket for our hot fire ignition tests. We also want to thank Ms. Mary Wadel, and Mr. Mike Binder of the NASA Lewis Research Center, Mr. Bob Grabowski and Mr. Richard Patz of Pratt & Whitney Government Engines & Space Propulsion, and Mr. Javier Vasquez of McDonnell Douglas for sharing their rocket engine test experience in helping to pick favorable ignition conditions. We also want to acknowledge the engineers and technicians of Sierra Lobo, Inc. Engineering and Technical Services for their dedication to advancing U.S. aerospace technology through testing.

Appendix A

Vehicle Performance Calculations

The vehicle performance calculations are made using the two stage rocket shown in Eq. (1). The rocket equation is derived from Newtons second law of motion F = ma. The form that is used is taken from Ref. 3 and does not take into account drag force and gravity force.

$$\Delta v = (\mathrm{Isp}\, g)_1 \ln\left(\frac{1}{\delta_1 + \lambda_1}\right) + (\mathrm{Isp}\, g)_2 \ln\left(\frac{1}{\delta_2 + \lambda_2}\right) \quad (1)$$

The ΔV is the change in velocity that is required to reach and maintain a circular orbit at a given altitude. The initial velocity is assumed zero at the launch site. A typical value of ΔV required to maintain LEO is around 25 000 ft/sec. The ΔV used in this analysis is 30 882 ft/sec. This higher value of ΔV is used in Eq. (1) to compensate for gravity force and drag force. The value of the gravitational constant used in the analysis was 32.2 ft/sec^2.

The dead weight ratio is calculated in Eq. (2) and the payload ratio is calculated in Eq. (3).

$$\delta_i = \frac{m_{d_i}}{m_{0_i}} \quad (2)$$

$$\lambda_i = \frac{m_{L_i}}{m_{0_i}} \quad (3)$$

The initial mass of the first stage is calculated in Eq. (4) and the initial mass of the second stage is given by Eq. (5).

$$m_{0_1} = m_{L_1} + m_{d_1} + m_{p_1} \quad \text{where} \quad m_{L_1} = m_{0_2} \quad (4)$$

$$m_{0_2} = m_{L_2} + m_{d_2} + m_{p_2} \quad (5)$$

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REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	July 1997	Technical Memorandum

4. TITLE AND SUBTITLE

Hot Fire Ignition Test With Densified Liquid Hydrogen Using a RL10B–2 Cryogenic H_2/O_2 Rocket Engine

5. FUNDING NUMBERS

WU–565–02–02

6. AUTHOR(S)

Nancy B. McNelis and Mark S. Haberbusch

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
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8. PERFORMING ORGANIZATION REPORT NUMBER

E–10758

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA TM–107470
AIAA–97–2688

11. SUPPLEMENTARY NOTES

Prepared for the 33rd Joint Propulsion Conference & Exhibit cosponsored by AIAA, ASME, SAE, and ASEE, Seattle, Washington, July 6–9, 1997. Nancy B. McNelis, NASA Lewis Research Center and Mark S. Haberbusch, Ohio Aerospace Institute, (presently at Sierra Lobo Inc., Propellant Densification Systems, 20525 Homestead Park Drive, Strongsville, Ohio 44136). Responsible person, Nancy B. McNelis, organization code 5870, (216) 977–7474.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Category 28

This publication is available from the NASA Center for AeroSpace Information, (301) 621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

Enhancements to propellants provide an opportunity to either increase performance of an existing vehicle, or reduce the size of a new vehicle. In the late 1980's the National AeroSpace Plane (NASP) reopened the technology chapter on densified propellants, in particular hydrogen. Since that point in time the NASA Lewis Research Center (LeRC) in Cleveland, Ohio has been leading the way to provide critical research on the production and transfer of densified propellants. On October 4, 1996 NASA LeRC provided another key demonstration towards the advancement of densified propellants as a viable fuel. Successful ignition of an RL10B-2 engine was achieved with near triple point liquid hydrogen.

14. SUBJECT TERMS

Densified propellants; Rocket ignition

15. NUMBER OF PAGES	16. PRICE CODE
12	A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102