SAWE Paper No. 3481
Category No. 18

FLIGHT TEST ANALYSIS OF LOX/PROPYLENE UPPER STAGE ENGINE

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For Presentation at the
68th Annual Conference
of
Society of Allied Weight Engineers, Inc.
Wichita, Kansas, 18-20 May, 2009

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Abstract

The objective of this paper is to present testing and analysis of an early prototype upper stage engine which could be optimized and evolved into a second stage engine for a Nanosat Launch Vehicle (NLV). The NLV is designed to deliver a nominal 10 kg payload to LEO and is being developed by the California Launch Vehicle Education Initiative (CALVEIN), a partnership program between Garvey Spacecraft Corporation (GSC) and California State University, Long Beach (CSULB) [3].

The engine is pressure-fed and uses LOX/propylene as propellants. It is designed to operate at a chamber pressure of 1 MPa and provide a vacuum thrust of 2000 N. Propylene was chosen as a propellant because it provides a higher specific impulse than RP-1 with comparable density at cryogenic temperatures [16].

This paper presents a first iteration of the preliminary design intended for space operations with an expansion ratio of 70 as well as the testing of its sea level version with an expansion ratio of 4. The space engine is designed with targeted combustion efficiency of 95% and nozzle efficiency of 98%, corresponding to a specific impulse of 347 s.

A static fire test of the engine, shown in Figure 9, with expansion ratio of 4 has been conducted twice at sea level with a burn time of 15 seconds and 5 seconds, respectively. A flight test, shown in Figure 11, has also been conducted to test the capabilities of the engine. Recorded data will be used to assess previous assumed efficiencies and refine, if necessary, the shape of the nozzle and configuration of the injector.

The next steps include implementing necessary changes to the engine to achieve better performance for future testing. This paper will also address ways to manage and reduce overall engine weight to improve performance.

I. Introduction

The initial Nanosat Launch Vehicle concept, shown in Figure 1 [3][4][5], was first proposed by GSC and CSULB in 2003 using LOX/ethanol as propellants [2]. The configuration underwent a series of trade studies resulting in a 2-stage pressure-fed vehicle, using LOX and densified propylene as propellants, and capable of placing a 10 kg (22 lbm) payload into a nominal 250 km altitude polar orbit. The nominal chamber pressure is 2 MPa (300 psi) for the first stage engine and 1 MPa (150 psi) for the upper stage engine. Other vehicle characteristics are listed in Table 1. Some of the technologically advanced characteristics of the NLV are; composite propellant tanks, densified propylene as fuel [6], hot gaseous helium as pressurant, as well as the potential use of carbon/silicon carbide (C/SiC) [7] for the combustion chamber and nozzle. A series of developmental static fire tests and fight tests have been conducted, ranging from full scale first stage to a full scale NLV fight [8][9][10][11].

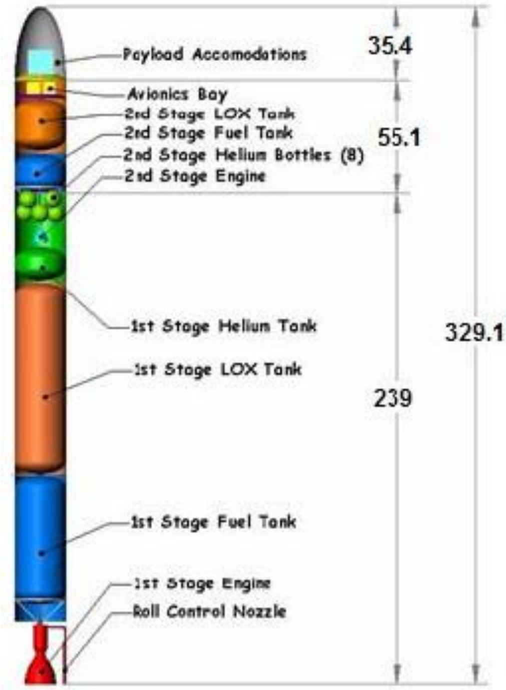

Figure 1- Baseline NLV Design

An evolutionary succession was the Prospector-9, which featured a pair of large integral composite tanks and a 20,000 N (4500 lbf) engine that is representative of the NLV first stage engine [18]. Development efforts to date have focused primarily on the first stage and its 20,000 N (4,500 lbf) thrust engine. Some initial static fire tests with room-temperature propylene have been conducted [6], but none were designed specifically for use on NLV upper stage. A concept for the upper stage engine development is presented in this paper at preliminary design stage. The next section describes the development of the engine while meeting NLV design requirements.

Table 1- Initial Parameters of NLV

	First Stage	Second Stage
Dry mass	171 kg	30 kg
Stage inert mass fraction	0.131	0.137
Chamber Pressure	2 MPa	1 MPa
Sea-Level Thrust	20000 N	N/A
Sea-Level ISP	212 s	N/A
Vacuum Thrust	29600 N	2000 N
Vacuum ISP	314 s	347 s
Seperation/burnout altitude	54 km	250 km

II. Upper Stage Engine Design

1. Engine Overview

The engine is compromised of three major subassemblies; the injector, the combustion chamber assembly, and the ignition subsystem. The pressure-fed engine uses LOX/propylene, operates at a chamber pressure of 1 MPa (~150 psi) and provides a vacuum thrust of 2000 N (~450 lbf). Propylene is chosen as the propellant because it provides a higher specific impulse than RP-1 with comparable density at cryogenic temperatures [16] and offers very close performance to methane [19]. A direct spark or a spark torch are likely candidates for the ignition subsystem along with pyrophoric using TEA/TEB. The combustion chamber assembly is built using an ablative liner and a carbon fiber overwrap, while a flat head injector provides additional film cooling to minimize ablation rates. Although the engine is initially designed to use an ablative engine, the configuration can be modified such that the ablative combustion chamber assembly can be replaced with a ceramic matrix composite (CMC). Figure 2 shows the engine with an ablative chamber. The thrust to weight ratio of the engine is required to be above 50. This value is on the lower end and can be significantly increased. The injector weight of the current design can be significantly reduced, however ease of manufacturing is determined to be the primary focus for the prototype. Also, excessive ablative thickness in the nozzle and some other less critical areas can be removed. Some characteristics are summarized in Table 2 along with design variables.

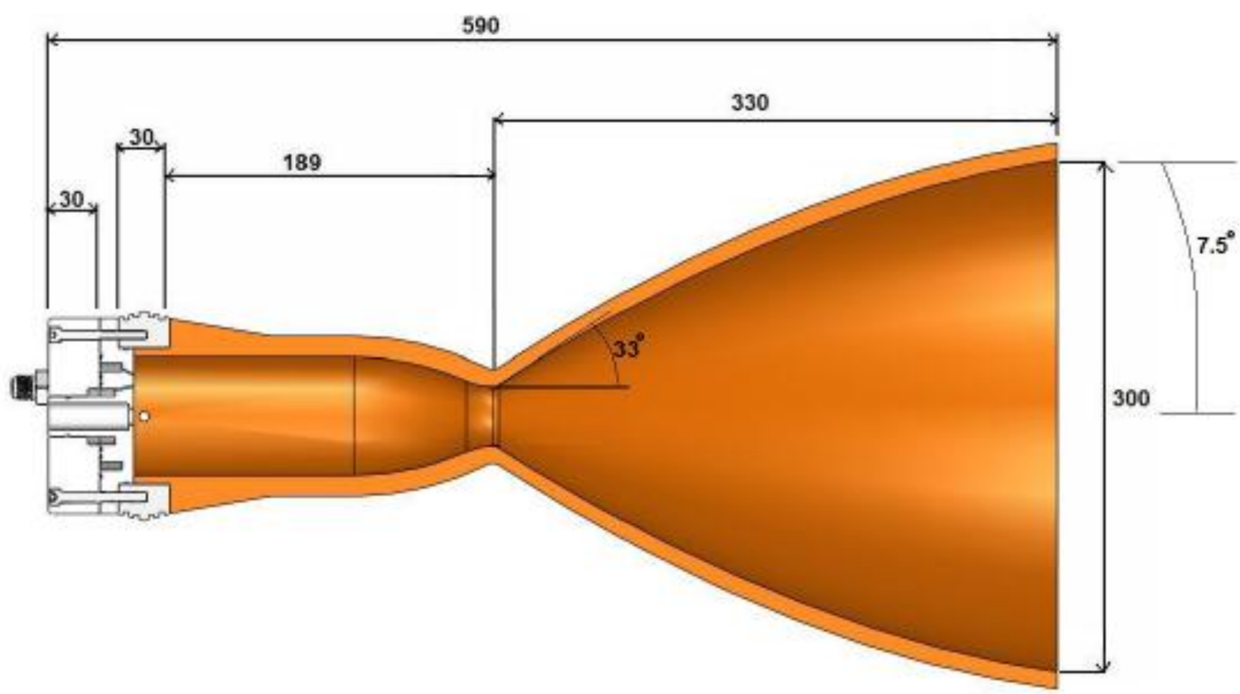

Figure 2- Upper Stage Engine Cross-Sectional View. Dimensions in mm

Table 2- Upper Stage Engine Characteristics

	Space Engine
minimum T/W requirement	50
ε	70
O/F	2.6
L*	0.8
M_c	0.16

III. Prototype Engine for Sea Level Testing

The development of a prototype version of the upper stage engine tested near sea level conditions is discussed here. The prototype engine is truncated for sea level testing, which took place at Mojave Desert. Due to possible high temperatures at the test site, the engine chamber pressure is adjusted to avoid cavitations in the feed lines and injector.

1. Engine Operating Conditions

The static fire test was conducted at an ambient temperature of about 25˚ C. Figure 3 displays physical characteristics of propylene evaluated for different temperatures at various chamber pressures. In order to avoid cavitations, the nominal chamber pressure was adjusted [17]. This ensured that the fuel stays liquid in the feed system and injector. Note that for cryogenic temperatures, this test problem can be neglected.

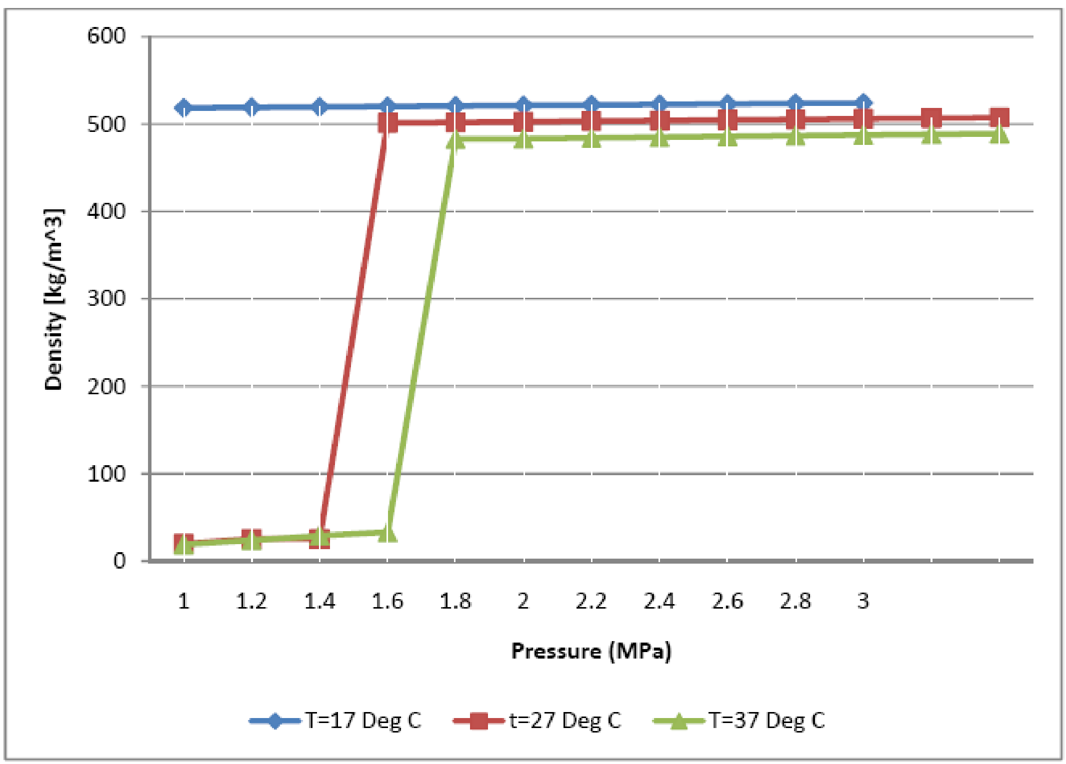

Figure 3- Pressure vs. Density at different temperatures

2. Prototype Overview

The prototype has a similar configuration to the upper stage engine. The configuration was kept similar to help make future upper stage engines easy to machine with interchangeable components, if necessary. This made the engine design compatible with spark torch ignition components as well as a CMC chamber. A pyrotechnic torch igniter was used instead of the baseline spark torch igniter for the prototype engine because of previous pyrotechnic torch ignition experience at CALVEIN, which lowered system risk. The igniter is still center mounted, which is ignited using an electric match. Figure 4 shows a cross section of the prototype engine along with the injector, center mounted igniter, combustion chamber, and the truncated nozzle.

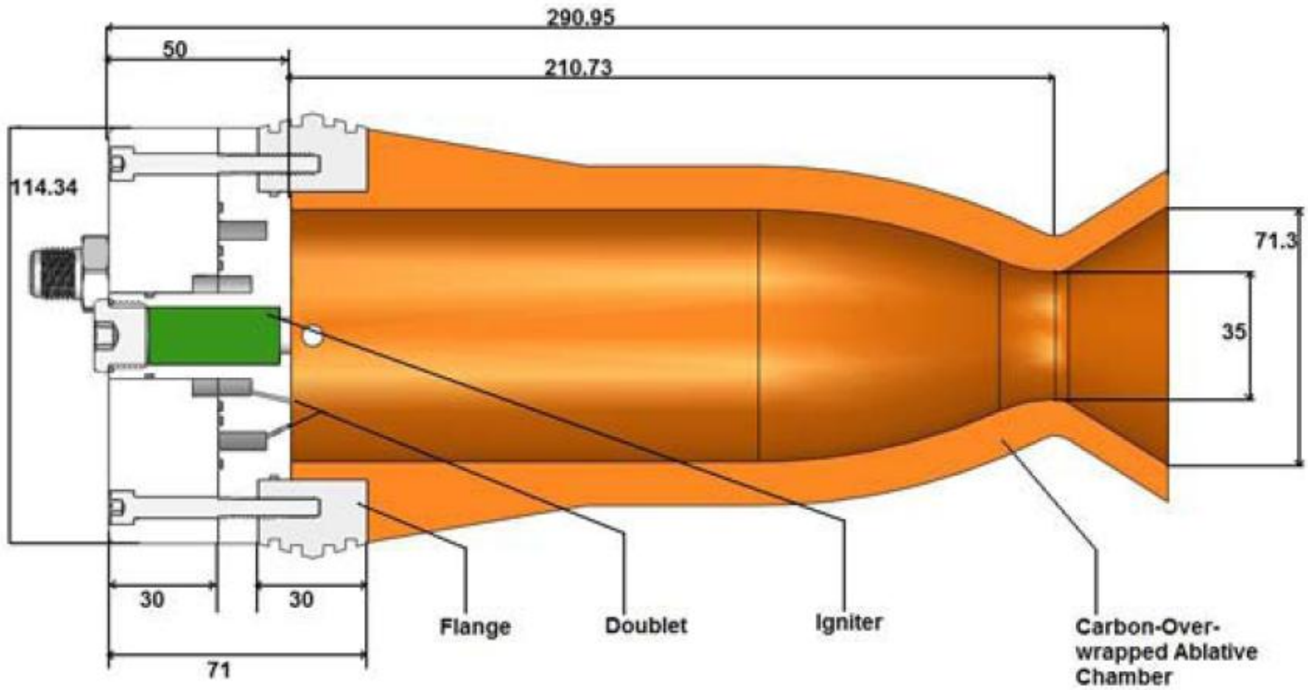

Figure 4- Prototype Sea Level Engine Cross-Sectional View. Dimensions in mm

Table 3 compares characteristics for both the prototype and the upper stage engine.

Table 3- Characteristics of Prototype and Upper Stage Engine

	Prototype (SFT)	Upper Stage Engine
Fuel	room temperature	densified (cryogenic)
Nominal P_c	1.03 MPa	1.03 MPa
Isp	195 s (SFT)	347 s (Vacuum)
P_c range	1.03-2.76 MPa	0.5-1.03 MPa
Thrust	1220-4000 N	1000-2000 N
ε	4	70
λ	0.95	0.98
η_{c*}	0.90	0.95
c*	1616 m/s	1616 m/s
T_c	3470 K	3470 K
P_e	56,500 Pa	1507 Pa
P_a	90,000 Pa	10 Pa
γ	1.21	1.12

a. Injector

A flat head injector was chosen as the type of injector since it has a potential of providing better performance than other injectors types. It is designed with one set of unlike doublet and 16 % of the fuel is allocated to film cooling. The discharge coefficient was assumed to be 0.8, and was adjusted to 0.668 after several water-flow tests. To reduce pressure coupling between the combustion chamber and the feed system and prevent chugging, the assumed pressure drop between propellant feed system and chamber pressure was assumed to be 30% [12][13] of the chamber pressure. The drop is on the higher end since the upper stage engine will operate in blow down mode in its final phase of operation.

One of the key design features of the injector is its ability to get disassembled from the chamber without completely disassembling the two plates. This greatly reduces engine assembly time and eases integration. The igniter is center mounted. The injector is manufactured to have a pressure drop of close to 100 psi at nominal chamber pressure of 180 psi. The actual O/F ratio is determined to be 2.5. Figure 5 shows the side view and the bottom view of the manufactured injector, displaying o-ring grooves and the orifices, respectively. Figure 6 shows the dimensions of the orifices in mm.

Figure 5- Machined injector bottom plate showing o-ring grooves (left) & orifices on the face of plate (right)

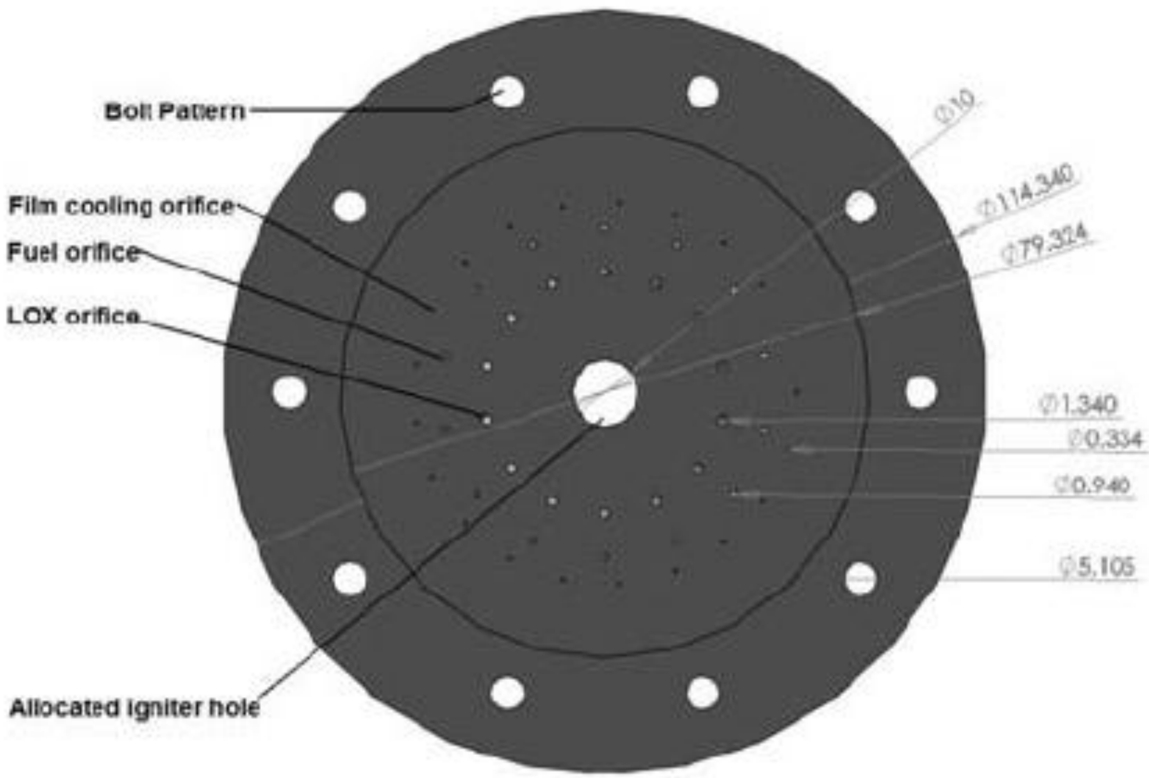

Figure 6- Injector Bottom plate showing orifices. Dimensions in mm

b. Combustion Chamber Assembly (CCA)

The ablative cooled combustion chamber and nozzle assembly, shown in Figure 7, is created using silica tape and high temperature epoxy. For longer duration burns, phenolic material can be used as an ablative. The ablative liner is over-wrapped with carbon fiber to provide structural rigidity to the assembly. The combustion chamber is cylindrical in shape, having a diameter of 69 mm and a length of 174 mm with a 3 degree draft to allow for easy release from the mold used to lay-up the ablative chamber. The throat diameter is 35 mm and the exit diameter is 71.3 mm, which results in an expansion ratio of 4.

Figure 7- CCA showing Silica Fiber Ablative & Carbon overwrap

Figure 8- Ignition Test

c. Ignition Subsystem

Ignition for the later NLV first stage engines have been accomplished using several small pyrotechnic lances inserted through the nozzle and ignited using electric matches. Similarly, like the larger first stage engines, [18] a pyrotechnic device mounted on the engine was used for ignition. In this case, however, instead of being mounted radially to the combustion chamber and firing inward like on previous large CALVEIN engines, the design is such that one igniter is mounted in the center of the injector. This configuration was validated at CSULB with a simple ignition test shown in Figure 8.

3. Static Fire Testing

The engine was static fire tested twice on a vertical test stand in order to validate the ignition algorithm, verify the engine basic integrity at startup, and characterize the engine performance over a broad range of operating conditions. Once the tanks have been pressurized, the ignition sequence begins at T-10. At T-3, the command is sent to turn on the igniter. Once the thermocouple confirms a significant change in temperature, the main valves are opened. At the end of the predetermined burn time, command is sent to close the main valves. The system behavior during start-up and test is shown in Figure 10, which depicts the variation of pressures in the propellant tanks, in the feed lines right upstream of the injector, and in the combustion chamber. In this configuration, the LOX and fuel valve have separate pneumatic actuation systems. The LOX valve starts opening at 75 ms while the fuel valve opens about 160 ms after the command is issued, leading to a smooth ramp up of the combustion chamber pressure. For the first test, tank pressures of nominal 275 psig resulted in a chamber pressure of nominal 130 psig. The test lasted for about 10 seconds. The second test, shown in Figure 9, was determined to be 5.5 seconds. Fuel and LOX tank pressures were at nominal 390 psig, resulting in a nominal chamber pressure of 202 psig and a thrust of approximately 385 lbf.

Figure 9- Second Static Fire Test, November 2008

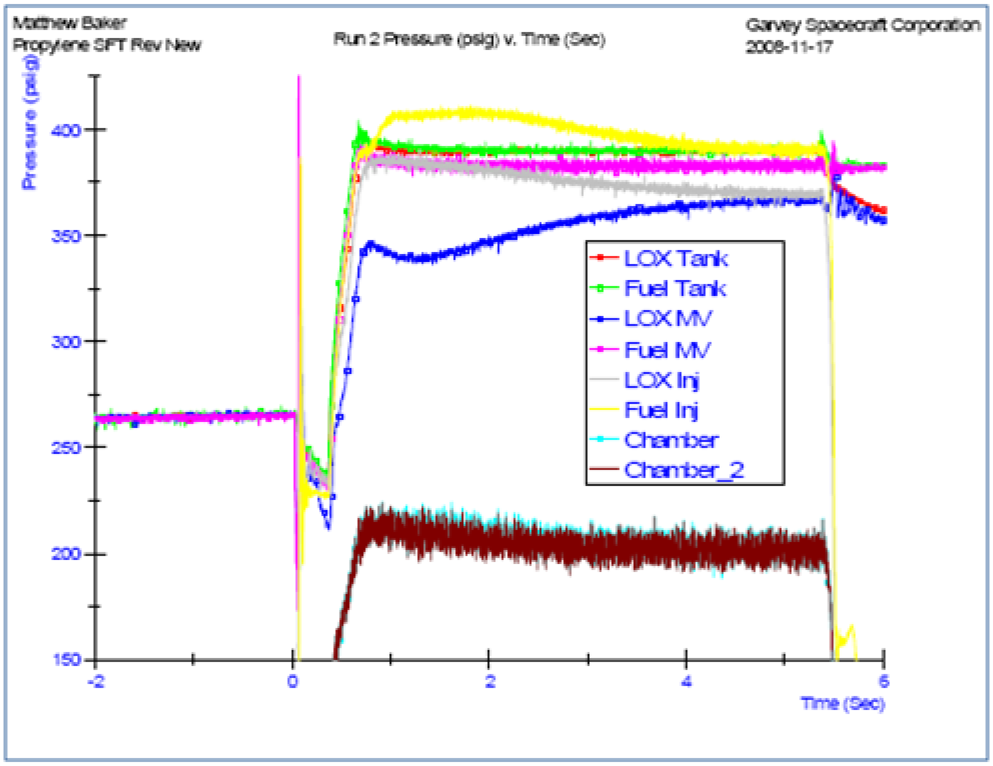

Figure 10- Second Static Fire Test Data showing Pressure in various components vs. Time

IV. Engine Mass Properties

The sea level engine used a Mass Equipment List (MEL) to keep track of the masses of all components as well as the mass of the system. An example of the MEL used to calculate the mass of the system is shown in Table 4. During the design phase of the engine, each component was designed in SolidWorks and assigned a density depending on the material used. The mass of each component was then calculated using mass properties function in SolidWorks to achieve a calculated value. Being at its final stage, the MEL shown below has all actual masses, and are indicated at the component level. The thrust to weight ratio was 62.94 for the engine. The goal is to get this value close to a 100. A similar MEL (not shown) was also updated for the upper stage engine concept which has a lower thrust to weight ratio because of its heavy nozzle with an expansion ration of 70.

Many solutions have been placed to reduce the mass of the system. The injector machined out of aluminum does not need to be as large. It was designed that way to ease the process of machining for this prototype. The ablative material used for the combustion chamber assembly was also thicker than necessary. The next generation CCA will have variable thickness depending on the how critical the area is and the amount of pressure it will need to handle. The thickness of carbon overwrap on CCA will also be reduced with a smaller factor of safety.

Table 4- Mass Equipment List of Sea Level Engine

			Total Predicted Mass [lb]	Basic Mass			Estimated Growth							Thrust to Weight Ratio	62.94	
SS-ARES Summary			7.2	7.2			0.0									
			lead	Basic Mass [lb]			Growth Allowance [lb]	Growth Category			Growth Factor [%]			Maker	Vendor / Distributor	Ref. #
1	2	3		1	2	3		Actual	Calculated	Estimated	Actual	Calculated	Estimated			
								19	0	0	0%	15%	30%			
								100%	0%	0%						
Engine				7.2												
	Injector		Deepak		3.7											
		Top Plate				1.6	0.0	1			0%	15%	30%	CSULB-Manufactured	IMS	IN_TPT-1
		Bottom Plate				0.8	0.0	1			0%	15%	30%	CSULB-Manufactured	IMS	IN_BPT-1
		Flange				1.0	0.0	1			0%	15%	30%	CSULB-Manufactured	IMS	IN_F-1
		O-Ring 1				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_OR-1
		O-Ring 2				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_OR-2
		O-Ring 3				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_OR-3
		O-Ring 4				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_OR-4
		O-Ring 5				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_OR-5
		Fasteners				0.3	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IN_FST
		LOX Fitting				0.1	0.0	1			0%	15%	30%	Deering Industries	Deering Industries	IN_LF-1
		Fuel Fitting				0.1	0.0	1			0%	15%	30%	Deering Industries	Deering Industries	IN_FF-1
	CCA		Kai		3.3											
		Ablative Liner				2.8	0.0	1			0%	15%	30%	CSULB-Manufactured	cotronics.com	CCA_AL-1
		Carbon Fiber Overwrap				0.3	0.0	1			0%	15%	30%	CSULB-Manufactured	avtcomposites.com	CCA_CFO-1
		PT Fitting				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	CCA_PTF-1
		PT Coupler				0.0	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	CCA_PTC-1
		PT Elbow				0.1	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	CCA_PTE-1
		Thermocouple				0.1	0.0	1			0%	15%	30%	Omega	omega.com	CCA_TC-1
	Ignition System		Mehrdad		0.1											
		Plug				0.1	0.0	1			0%	15%	30%	McMaster Carr	Mcmaster.com	IG_P-1
		Solid Motor				0.0	0.0	1			0%	15%	30%	Aerotech-Rocketry.com	Aerotech-Rocketry.com	IG_SM-1

V. Flight Demonstration

After completion of the static fire tests, a fight test was conducted in February, 2009 (Figure 11). The LOX/propylene engine was mounted on the Kimbo V vehicle, which was previously launched in 2001 by CALVEIN. This vehicle was refurbished to accommodate the design of this engine. A new recovery system was added to the forward end of the vehicle with a parachute for nose cone and another one for the vehicle. Feed lines were changed on the aft end to adapt to the fitting sizes on the engine. An adapter plate was fabricated to be able to mount the engine on the vehicle. Mass and center of gravity calculations were performed to achieve fin design and meet stability requirements.

The ignition system ignited the propellants instantly at T-0 and the vehicle took off with a thrust to weight ratio of approximately 4. The Gross Lift-Off Weight (GLOW) was determined to be approximately 55 kg (~122 lbs). The tank pressures were set to be nominal 425 psig, and both propellant tanks were loaded to about 50% of their capacity. Figure 11 shows Prospector-13 LP climbing the launch rail and in fight. The engine had a nominal burn time of about 20 s with an apogee of 1588 m (~5210 ft). The nose cone was forced off using pyrotechnic charge to make way for the main parachute on the vehicle to deploy. The main parachute was anchored on the forward end of the vehicle and was caught in the aft launch rail guide during the deployment stage and never fully opened its canopy. The crashed vehicle was recovered and further analysis is being performed to be able to rectify recovery and any other issues for future launches.

Figure 11: P-13 LP climbing the launch rail (left) & first ever flight using LOX/propylene (right)

VI. Conclusion

The approach presented here follows the incremental development approach employed by the team to leverage existing resources while making progress towards the development of an operational NLV. The paper presents a preliminary design of the NLV upper stage engine and expands on the development of an early prototype and testing near sea level conditions. The static fire test validated the design and the fight test showed that the engine is capable of at least a 20 second burn. A detailed analysis will be conducted on the data acquired from both the static fire and the flight test. The pressure drop within the injector is larger than anticipated and will need to be investigated. Testing will need to be performed on the spark torch igniter, which will be used on the space engine to replace the currently used pyrotechnic igniter. The engine configuration will also be optimized to achieve reduction in mass. Overall, the system was a successful design and created a baseline design to optimize for the final NLV upper stage engine.

Author Biography

Deepak Verma

Mr. Verma graduated from California State University, Long Beach (CSULB) with Bachelors in Aerospace Engineering and is currently pursuing Masters in Aerospace Engineering with emphasis on Space Systems Engineering at CSULB. While pursuing his undergraduate studies, he worked on CALVEIN program or better known as "Rocket Project". This program is made possible through joint collaboration of CSULB and Garvey Spacecraft Corporation (GSC). On this program, he managed and led various different projects from concepts to completion. He was the project lead on the Prospector-14 LM vehicle, which became the first ever flight of a vehicle using Liquid Oxygen and Methane. He also worked on the Liquid Propelled Annular Multi-Chamber Aerospike Engine through its design and development stage. Recently, he successfully launched the first ever vehicle using Liquid Oxygen and Propylene. This engine was designed with the help of his teammate Kay Gemba and tested with collaboration by Dr. Eric Besnard, the project director at CSULB, and John Garvey, CEO of GSC. Deepak recently started work at Northrop Grumman and is now working in the Lunar Lander Program as a Propulsion Engineer.

Kay Gemba

Mr. Gemba graduated from California State University, Long Beach (CSULB) with Bachelors in Aerospace Engineering and in Applied Mathematics. He is currently pursuing Masters in Chemical Engineering at CSULB. He is a PhD candidate in Ocean and Resource Engineering at University of Hawaii for the Fall 2009 semester. He has been an active member of CALVEIN and worked on various launch vehicles. He designed the LOX/propylene upper stage engine with his teammate Deepak Verma and created procedures for static fire and flight testing of the engine. His current research and interests include Liquid Propulsion, Neural Networks and Alternative Energies.

Acknowledgements

We would like to thank the CALVEIN team for their outstanding performance and help in conducting a static fire and fight test of the engine.

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