# Flight Testing of a Prototype LOX/propylene Upper Stage Engine[1]

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**The paper discusses the development and testing of a 2000 N thrust liquid oxygen/propylene rocket engine designed to power the upper stage of a Nanosat Launch Vehicle (NLV). The preliminary design is intended for space operations with an expansion ratio of 70. The targeted combustion efficiency is 95% and nozzle efficiency is 98%, corresponding to a specific impulse of 347 s.**

**Propellants are introduced and mixed in the combustion chamber utilizing an unlike doublet injector element. In addition, film cooling is provided in order to extend the life of the ablative chamber. Ignition is accomplished with a single igniter located in the center of the face of the injector.**

**Consistent with the employed incremental approach, these requirements are relaxed for the first prototype version of the engine: the expansion ratio of the nozzle is reduced to 4 by truncating the nozzle, the targeted combustion efficiency is reduced to 90% and the nozzle efficiency to 95%. Two static fire tests (SFT) have been conducted. The recorded chamber pressure of the second test was nominal 202 psig along with a thrust of about 1712 N. Recorded data is presented within this report.**

**On Feb. 21, 2009, a team at California State University, Long Beach was able to conduct a successful flight test of this LOX/propylene engine. It is believed to be the first time a LOX/propylene engine has been used in flight. Performance data of this flight test is presented within this report.**

## Nomenclature

| | |
|---|---|
| $\eta_{c^*}$ | Combustion Efficiency |
| $\gamma$ | Ratio of Specific Heats |
| $\lambda$ | Nozzle Efficiency |
| $\varepsilon$ | Nozzle Expansion Ratio |
| $c^*$ | Effective Characteristic Exhaust Velocity |
| $L^*$ | Characteristic Length |
| LOX | Liquid Oxygen |
| $M_c$ | Combustion Chamber Mach Number |
| O/F | Oxidizer to Fuel Ratio |
| $P_a$ | Ambient Pressure |
| $P_c$ | Combustion Chamber Pressure |
| $P_e$ | Nozzle Exit Presure |
| $T_c$ | Chamber Temperature |
| T/W | Thrust to Weight Ratio |
| TEA/TEB | Triethylaluminum and Triethylborane |

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# I. Introduction

The initial Nanosat Launch Vehicle concept was first proposed by Garvey Spacecraft Corporation (GSC) and California State University, Long Beach (CSULB) in 2003 using LOX/ethanol as propellants[2].

The configuration underwent a series of trade studies resulting in a 2-stage pressure-fed LOX/densified propylene vehicle, shown in figure 1[3,4,5], 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 and 1 MPa (150 psi) for the upper stage. Other vehicle characteristics are listed in table 1. A schematic of the vehicle is presented in figure 1.

Some of the technological 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]. A series of developmental static fire tests and flight tests have been conducted, ranging from a low fidelity but full scale first stage to a full scale NLV flight[8,9,10,11]. An evolutionary succession was the Prospector-9, which featured a pair of large integral composite tanks instead of the cluster of small 2.85 US Gal tanks used in all earlier vehicles and a 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]. 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 meeting NLV requirements.

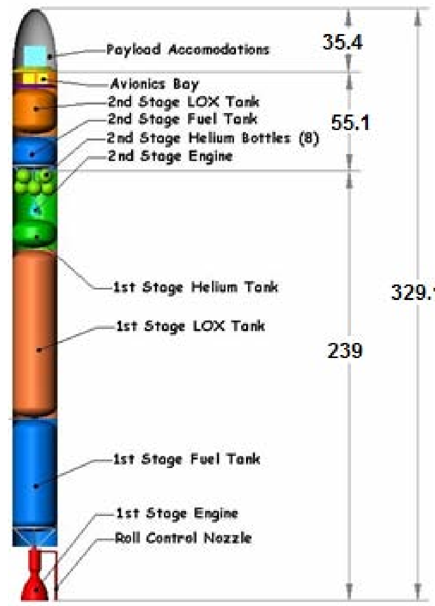



Figure 1: NLV Baseline. Dimensions in cm

Table 1: NLV Characteristics

| | 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 development

### A. Engine overview

The engine is compromised of three major subassemblies; the injector, ignition system and the combustion chamber assembly. The engine uses LOX/propylene, is pressure fed, operates at a chamber pressure of 1 MPa (approx. 150 psi) and provides a vacuum thrust of 2000 N (approx. 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 system 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.

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 can be removed. Characteristics are summarized in table 2 along with design variables.

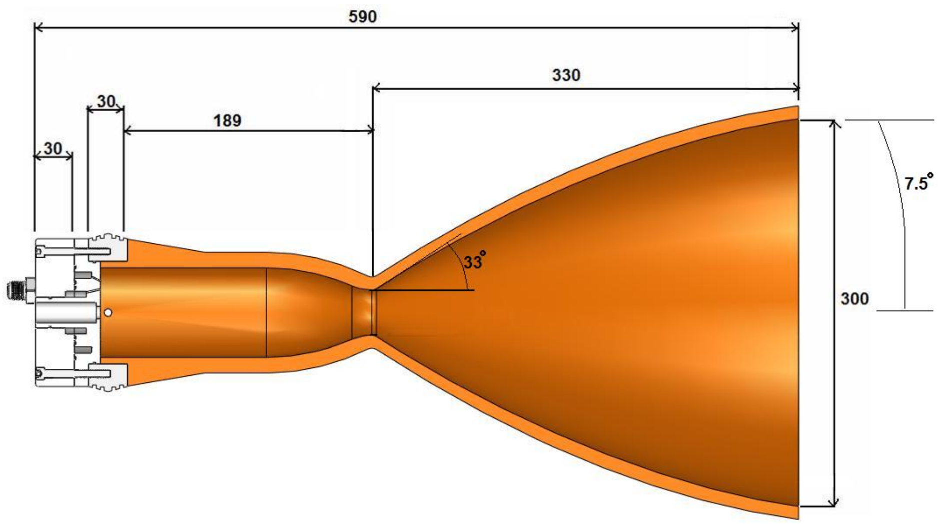



Figure 2: LOX/propylene ablative engine concept. Dimensions: mm

Table 2: Upper Stage Engine Characteristics

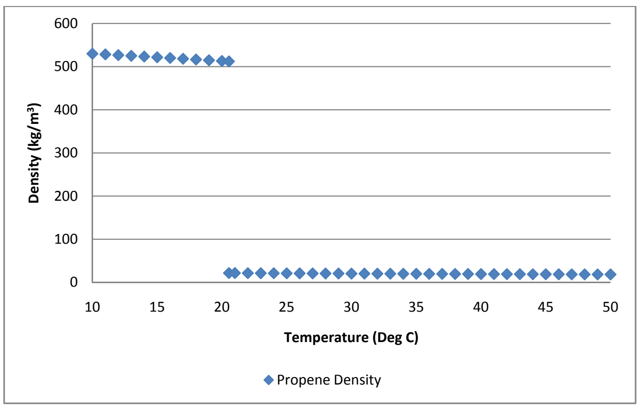
| | Space Engine |
|---|---|
| minimum T/W requirement | 50 |
| $\varepsilon$ | 70 |
| O/F | 2.6 |
| L* | 0.8 |
| $M_c$ | 0.16 |

# III. Prototype Engine for Sea Level Testing

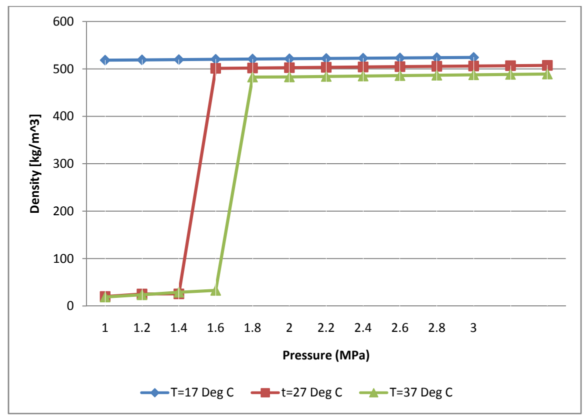
The development of a prototype version of this engine tested near sea-level conditions is discussed here. The prototype engine is truncated for sea level testing, which took place at Mojave Desert. Also, due to possible high temperatures at the test site, the engine chamber pressure is adjusted to avoid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

## A. Engine operating conditions

The static fire test is conducted at an ambient temperature of about 25 degree Celsius. In order to avoid cavitations, the nominal chamber pressure needed was adjusted[17] to ensure that the fuel stays liquid in the feed system and injector. Figure 3 displays physical characteristics of propylene evaluated at nominal chamber pressure. The corresponding temperature at which propylene vaporizes is determined to be close to 21 degree Celsius. Note that for cryogenic temperatures, this test problem can be neglected.




(a) Vaporization Properties of Propylene evaluated at 1 MPa




(b) Vaporization Properties of Propylene

Figure 3: Propylene physical characteristics

## B. Prototype Overview

The prototype has a similar configuration to the upper stage engine, however some modifications were necessary. The spark ignition system has been replaced by a pyrotechnic torch which is still center mounted but the electric match used to ignite the pyrotechnic device is fed through the nozzle. Figure 4 shows a cross section of the prototype engine along with the injector and center mounted igniter.

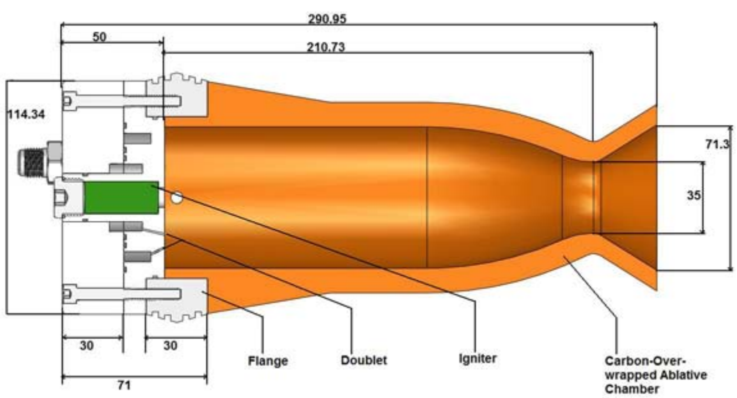



Figure 4: Prototype Cross Section. Dimensions: mm

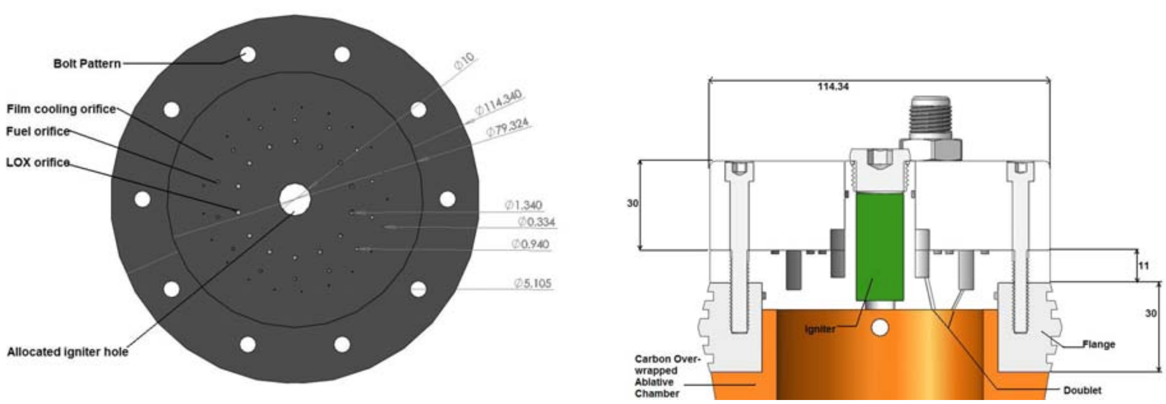
Table 3 compares characteristics for both the prototype and the upper stage engine.

Table 3: Upper Stage Engine Characteristics

| | 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 |
| $\varepsilon$ | 4 | 70 |
| $\lambda$ | 0.95 | 0.98 |
| $\eta_{c^*}$ | 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 |
| $\gamma$ | 1.21 | 1.12 |

### *1. Injector*

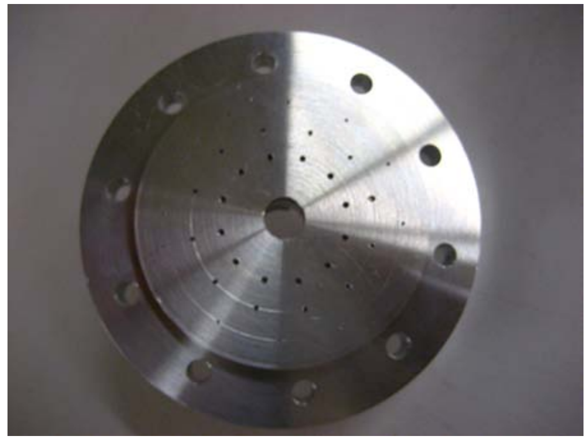
A flat head injector was chosen as the type of injector since it provides better performance than other injectors. It is designed with one set of unlike doublets and 16 % of the fuel is allocated to film cooling. The discharge coefficient was assumed to be 0.8, adjusted to 0.668 after several water-flow 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 is 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 (throttled) in its final phase of operation. One of the key design features of the injector is that it can be removed from the chamber without completely disassembling the 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*a* shows the bottom view and figure 5*b* the side view of the injector. Dimensions for both figures are in mm. The manufactured bottom plate is shown in figure 6.




(a) Bottom View (b) Cross Section View

Figure 5: Injector Design Figures

(a) Bottom Plate, showing O-ring Grooves



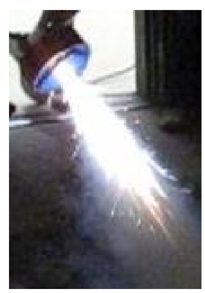
(b) Bottom Plate, showing Orifices

Figure 6: Manufactured Injector

### 2. *Combustion Chamber Assembly*

The combustion chamber and nozzle assembly is an ablative 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 add extra strength to the chamber and nozzle. 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 (Figure 7*a*). The throat diameter is 35 mm and the exit diameter is 71.3 mm, which results in an expansion ratio of 4.

(a) Combustion Chamber



(b) Ignition System Test

Figure 7: Combustion Chamber and Ignition System Test

### 3. *Igniter*

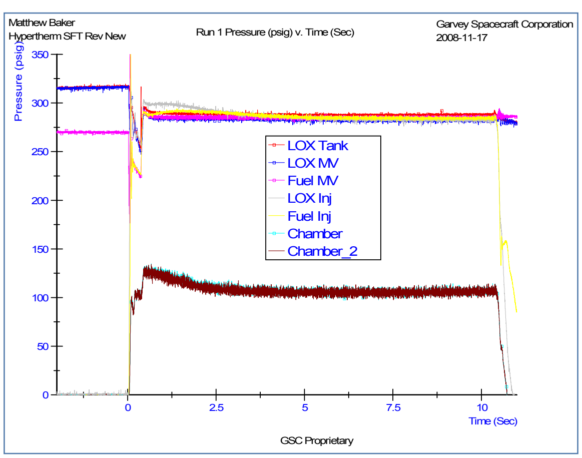
Ignition system for the later NLV first stage engines are accomplished by using several small pyrotechnic lances which were inserted through the nozzle. This represented a scaling up from the ignition configuration employed on smaller engines. Similarly to the larger first stage engine prototype,[18] a pyrotechnic device mounted on the engine is used for ignition. In this case, however, instead of being mounted radially to the combustion chamber and firing inward, the design is such that one igniter is mounted in the center of the injector. Figure 5*b* shows the igniter in green. The configuration itself was validated at CSULB with a simple ignition test shown in figure 7b.

### C. Ignition tests

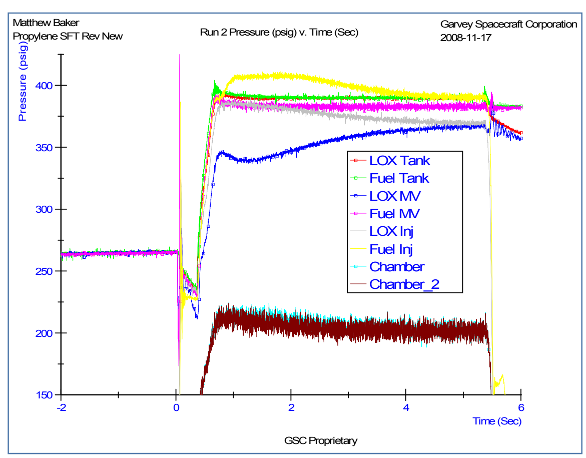
The engine was first tested on a vertical test stand (Figure 8) in order to validate the ignition algorithm, verify the engine basic integrity at startup, and characterize the engine performance over a broad range of operating conditions. Once the tanks have been pressurized, the ignition sequence begins 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 the command is sent to close the main valves. The system behavior during start-up and test is shown in Figure 9 which depicts the variation of pressures in the propellant tanks, in the feed lines right upstream of the injector and in the 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

Figure 8: First Static Fire Test

resulted in a chamber pressure of nominal 130 psig. The test lasted for about 10 seconds. The duration of the second test was determined to be 5.5 seconds. Fuel and LOX tank pressures are nominal 390 psig, resulting in a nominal chamber of 202 psig and a thrust close to 385 lbf.




(a) Static Fire Test No. 1




(b) Static Fire Test No. 2

Figure 9: Recorded Data of LOX/propylene SFT

## IV. Flight Test Demonstration

With the completion of the static tests, a flight test was conducted in Feb. 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 itself. 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 (Figure 10).

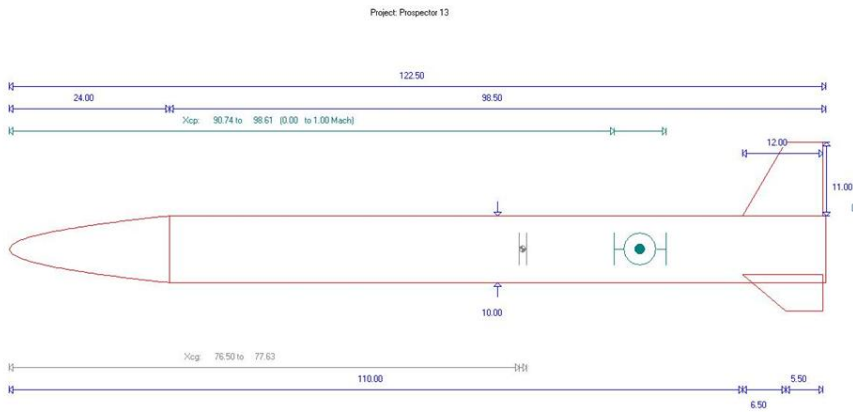



Figure 10: CG and CP locations of P13. Dimensions: inch

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 liftoff weight (GLOW) was determined to be close to 122 lbs. The tank pressure was set to be nominal 425 psig, both propellant tanks were loaded to about 50% of their capacity. Figure 11 shows the P13 climbing the launch rail and in flight. The engine had a nominal burn


(a) P13 on rail

(b) P13 in flight

Figure 11: First Flight of a LOX/propylene engine

time of about 20 s with an apogee of 5211 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.

## V. 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 flight test shows 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 and will be presented in the near future. The pressure drop within the injector is larger than anticipated and will need to be investigated. The configuration of the thermocouple will be changed. Testing will need to be performed on the spark igniter which will be used on the space engine to replace the currently used pyrotechnic igniter.

## VI. Acknowledgments

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

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