

A Custom Launch System for Satellites Smaller than 1 kg

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ABSTRACT: This paper presents the concept and design of a custom launch system for satellites with a mass equal or less than 1 kg. Current launch opportunities carry these satellites as secondary payloads to orbit. To date, no launch system has targeted a satellite of less than 1 kg as a primary payload. The case is made for an airborne, balloon based rocket launch. A four stage rocket using a hybrid propellant combination is proposed. With an assumed specific impulse of 235 seconds, four stages and a redundant structure to propellant mass ratio of 1/10, a burnout velocity of 9,200 m/sec can be achieved. Subtracting the velocity losses due to drag in the upper atmosphere, gravitation and earth rotation, a real burnout velocity of more than 7,600 m/sec can be achieved, which is sufficient to achieve Low Earth Orbit. The initial mass of the rocket and payload would be below 200 kg. For currently available zero-pressure balloons, this is an average payload which can be lifted up to 30 km in about an hour.

INTRODUCTION

The educational value of engaging students in the design and operation of satellites with a weight of 1 kg has been demonstrated by the ongoing success and interest in the development of Cubesats¹. Their scientific and technological value is steadily increasing due to miniaturization of sensors, actuators and electronics. Although their design and fabrication is accessible to universities, their launch into orbit remains the biggest hurdle in completing a full mission cycle: design – fabrication – launch – operation, within a reasonable budget and time.

Access to space has concerned small satellite developers since OSCAR I, the first Orbiting Satellite Carrying Amateur Radio, which was launched into orbit on December 12th 1961. The launch continues to be the most expensive and highest risk phase of any mission, even for medium and large payloads. The most recent attempts to provide solutions to the launch of small satellites have been presented

in^{2, 3, 4, 5, 6, 7}. The estimated launch costs range from a few \$100 K for Cubesats to \$10M for microsats. Notable is the fact that² and³ propose launch systems in which small satellites would be primary payloads. However, all currently available launch opportunities carry these satellites as secondary payloads to orbit. This means that the launch schedule and orbit parameters are dictated by the primary payload, i.e. the small satellite mission has to adapt to these constraints.

The paper presents first the launch system concept, making the case for a balloon based rocket launch and for a hybrid, four stage launch vehicle. Then it shows the subsystems that have been developed so far, including experimental results and test flights. Finally, it concludes by iterating the benefits of the proposed launch system.

LAUNCH SYSTEM CONCEPT

Balloon Based Rocket Launch

Airborne rocket launch is not a new concept. Under the name Rockoon, it was first used by Dr. van Allen's group from University of Iowa in the 1950's for high altitude

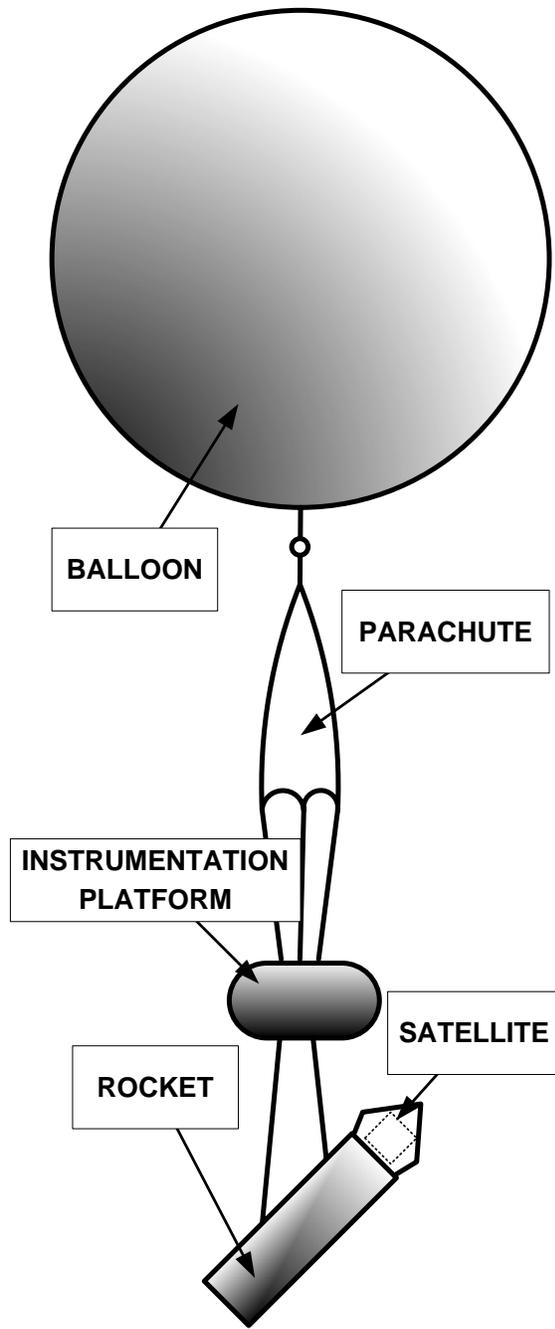


Figure 1. Launch System Assembly launched from the ground.

atmospheric research. In the Rockoon System, a sounding rocket was launched from a balloon at 20 km, reaching final altitudes of more than 100 km. In the past 15 years, Orbital Sciences Corporation has been successful in offering commercial launches with its Pegasus rocket launched from an aircraft⁷.

Current plastic thin film balloons are routinely used to lift payloads of more than 200 kg above 30 km⁸. Balloons are of two types: burst and zero-pressure. The so called zero-pressure balloons can float for extended periods of time once they reach the peak altitude. These are the obvious choice for this application because the ignition of the rocket will not be tied to the unpredictable moment of a balloon burst.

The System that is launched from the ground is shown, not to scale, in Figure 1. Under normal conditions, the helium filled balloon lifts its payload to an altitude of approximately 30 km. At this altitude the entire system floats until the ignition command is given from the ground launch control. In the case of an accidental tear of the balloon, its entire payload returns to ground using the parachute, which deploys automatically. The instrumentation platform contains: temperature and pressure sensors, an electronic compass, accelerometers, a Global Positioning System, video cameras, a flight computer and radios. These acquire and send real time telemetry, location information and live video to the ground control.

A typical launch profile is shown in Figure 2. After approximately 60 minutes, the balloon, the instrumentation platform and the rocket with the satellite reach the ~30 km floating altitude. This is followed by a period of passive stabilization. Then the rocket Inertial Navigation System is activated and location

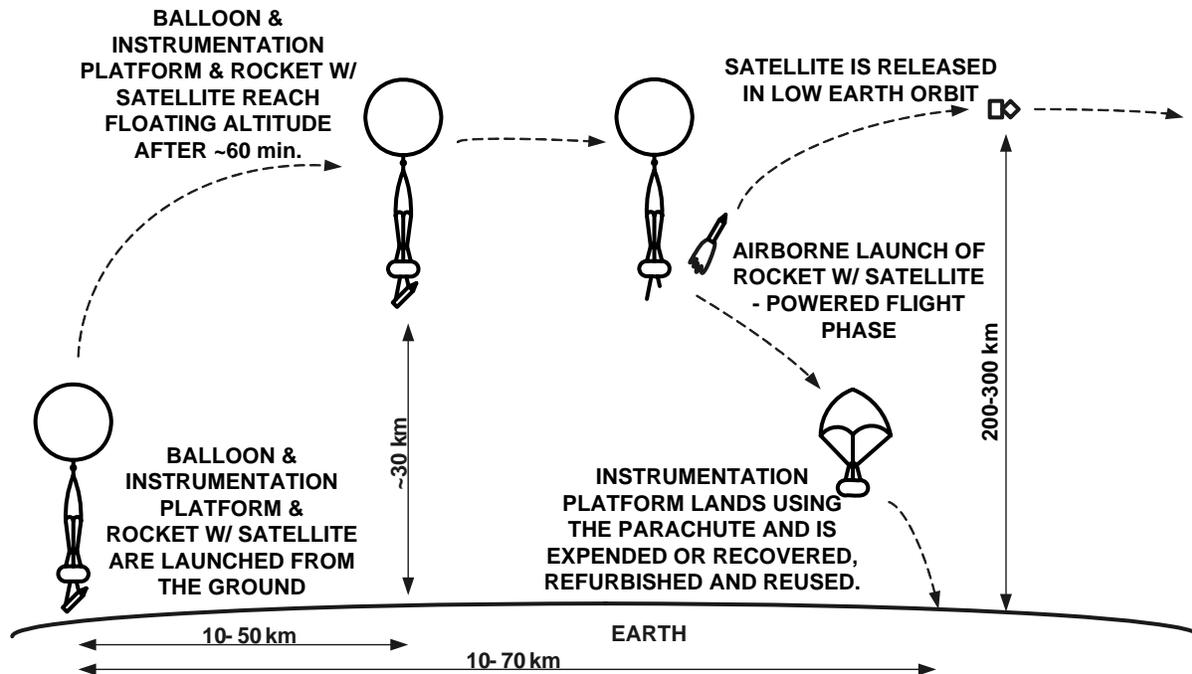


Figure 2. Typical Launch Profile.

and attitude information are uploaded. Next the ignition command is given from the ground, and the rocket starts the powered flight phase at an angle of $\sim 60^\circ$. This inclination is necessary to clear the very large balloon envelope. A few seconds into the flight, the rocket turns to a vertical attitude, from which point on the rocket follows a trajectory under the control of the Inertial Navigation and Guidance System. Under ground control command, the rip-off panel tears the balloon, releasing the instrumentation platform with the parachute. These are expended or recovered, refurbished and reused.

The balloon based rocket launch requires a mobile launch control, which can be located in a Van or SUV. It is estimated that a crew of seven can set up in about three hours and complete the launch with orbital insertion in less than six hours. The recovery of the instrumentation platform would require extra time. This mobility allows launches from virtually any latitude, restricted only by safety range considerations. As in the Rockoon

Program, the balloon and assembly can be launched even from a small ship at sea.

Obviously weather conditions affect the release of the balloon and assembly from the ground and the un-powered phase of the flight. Of particular concern are wind speed and direction. However, these vary greatly with altitude, and past experience has shown that with an ascent rate of ~ 500 m/min. the assembly would reach the floating altitude without getting out of radio communication range.

The Launch Vehicle

Safety is our utmost concern, as this is a student oriented, student driven university project. Therefore, it has been decided that safety considerations overwrite performance characteristics in the selection of propellant(s). Solid propellants are easy to store and relatively safe to handle and transport, but most of these could ignite upon impact, if for any reason the rocket would return to ground. Cryogenic liquid propellants like liquid

oxygen and liquid hydrogen can achieve specific impulses above 400 sec, but cryogenic storage would create a difficult problem both on the ground and during the un-powered phase of the flight. Non-cryogenic liquid propellants like rocket-grade ethanol and hydrogen peroxide would require another gas, helium, for pressurization.

Therefore, we have turned our attention to hybrid propellants. The fuel selected is R45M HTPB – Hydroxyl Terminated Poly Butadiene. The oxidizer is NOX, i.e. liquid Nitrous Oxide (N₂O) or laughing gas. This propellant combination has been already successfully used in human rated rocket engines^{9, 10}. The fuel is solid and non-explosive. The NOX can be stored and transported in liquid form at room temperature. It is liquefied at -88.5 C, but once loaded into the tank, it self-pressurizes at room temperature to about 5 MPa or 750 psi. This excludes the need for an additional gas for pressurization.

HTPB and NOX are not hypergolic, i.e. they do not ignite on impact. There is in fact a need for atomized NOX and atomized HTPB to start and sustain a high temperature burn. The initial atomization of HTPB is achieved through the burn of an igniter at more than 300 °C. Once ignited, the thrust can be throttled by controlling the amount of NOX, making it extremely versatile. In the case the rocket returns to ground without having been fired, the HTPB and NOX would not ignite and not explode on impact. The NOX could even be vented during the descent, without any adverse environmental impact. Storage, handling and transport of HTPB are very safe, as it is in fact a form of rubber. NOX can be purchased on site, safely stored for a few days, and safely handled and transported. Due to its intrinsic safety properties, this is the propellant combination we have chosen.

HTPB and NOX can be mixed in different amounts to achieve different specific impulses, I_{sp}. Of course, the higher the I_{sp} the better, but that usually implies a very high chamber temperature and implicitly expensive materials. One oxidizer / fuel ratio cited in the literature^{11, 12}, is NOX / HTPB = 6, with which a specific impulse of I_{sp} = 235 sec can be achieved. Assuming an I_{sp} = 235 sec, the exhaust velocity is then v_e = I_{sp}*g = 2300 m/s. The LEO orbital velocity, i.e. 200-300 km, is v_{LEO} = 7600 m/s. A total Δv_{LOSS} of 1600 m/s is assumed, due to: (1) thrust-atmospheric loss, (2) drag loss, (3) gravity loss, and (4) maneuvering and launch window allowance. The thrust-atmospheric and drag losses will be much smaller compared to a sea-level launch, due to the very high altitude of the entire powered flight. With these assumptions the necessary burnout velocity becomes v_{bo} = 9200 m/s. Assuming a four stage launch vehicle, with each stage contributing equally to the velocity increment, the burnout velocity after each stage would need to be v_{bostg} = 2300 m/s. Now with v_e = 2300 m/s and v_{bostg} = 2300 m/s, the mass ratio of each stage is:

$$\Delta m/m_0 = 1 - 1/\exp(v_{bostg}/v_e) = 1 - 1/\exp = 0.632$$

Assuming a 10% redundant structure, each stage will be able to lift ~0.267 of its initial weight. The table below shows the per-stage mass breakdown for different satellite masses, all being expressed in kg.

Table 1. Per Stage Mass Breakdown.

Masses in kg		Satellite Mass		
		1	0.75	0.5
Stage Mass	4 th	3.7	2.8	1.87
	3 rd	14	11	7
	2 nd	53	40	26
	1 st	196	148	98

Because the first stage fires at ~30 km, where atmospheric density and pressure are 1% relative to sea-level, the rocket engine nozzle geometry can be optimized for high efficiency. Furthermore, the value of the

maximum dynamic pressure or max-q, will be very low resulting in a more relaxed structural design. The latter is further supported by lower launch loads, e.g., vibrations, compared to a launch on a conventional vehicle.

LAUNCH SYSTEM DEVELOPMENT

The Instrumentation Platform

Three generations of students^{13, 14, 15}, have worked on the design and development of the instrumentation platform. This contains sensors, actuators and electronics, which allow the mobile ground control crew to monitor and control the launch system assembly. It was flight tested two times during spring 2005. Figure 3 shows the instrumentation platform being launched using a burst balloon.



Figure 3. The Instrumentation Platform is being launched using a burst balloon, during one of its test flights in Upstate

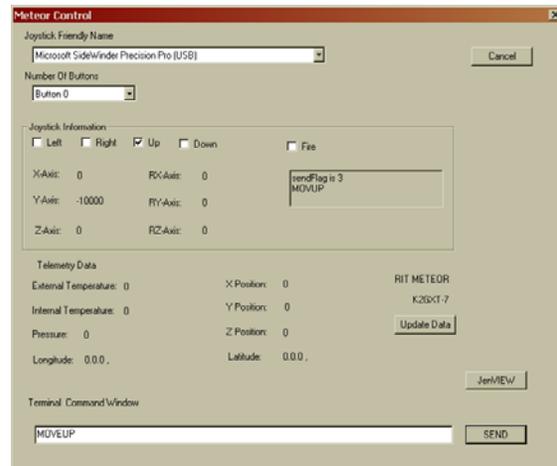


Figure 4. Ground Control Software developed by students.

Two of the authors and most of the students are licensed Amateur Radio Operators, and therefore radio communications occur in the allocated Amateur Radio Bands. Currently the 2 m band is used for telemetry download and commands upload, and the 70 cm band for the live video feeds. On the ground, the telemetry is processed and displayed using software developed by the students, as can be seen in Figure 4. The position is continuously updated in the Map Window.

Currently four video cameras are used to visually monitor the launch system assembly. For redundancy, the telemetry is overlaid on the video, as shown in Figure 5.

A fourth senior design team will complete the instrumentation platform implementation

during the academic year 2006/2007. The goal is to further decrease its weight through the use of almost exclusively surface mount components.

The Launch Vehicle

During the academic year 2005/2006, a team of eight senior students have begun the development of the fourth stage of the launch vehicle, as their senior capstone project. Without any prior experience in rocket design at the Rochester Institute of Technology, this student team and their advisors have designed and built a small rocket engine test stand and a 200 N hybrid rocket engine in just 20 weeks. The rocket engine was successfully tested twice, and more tests are scheduled for summer and fall 2006.

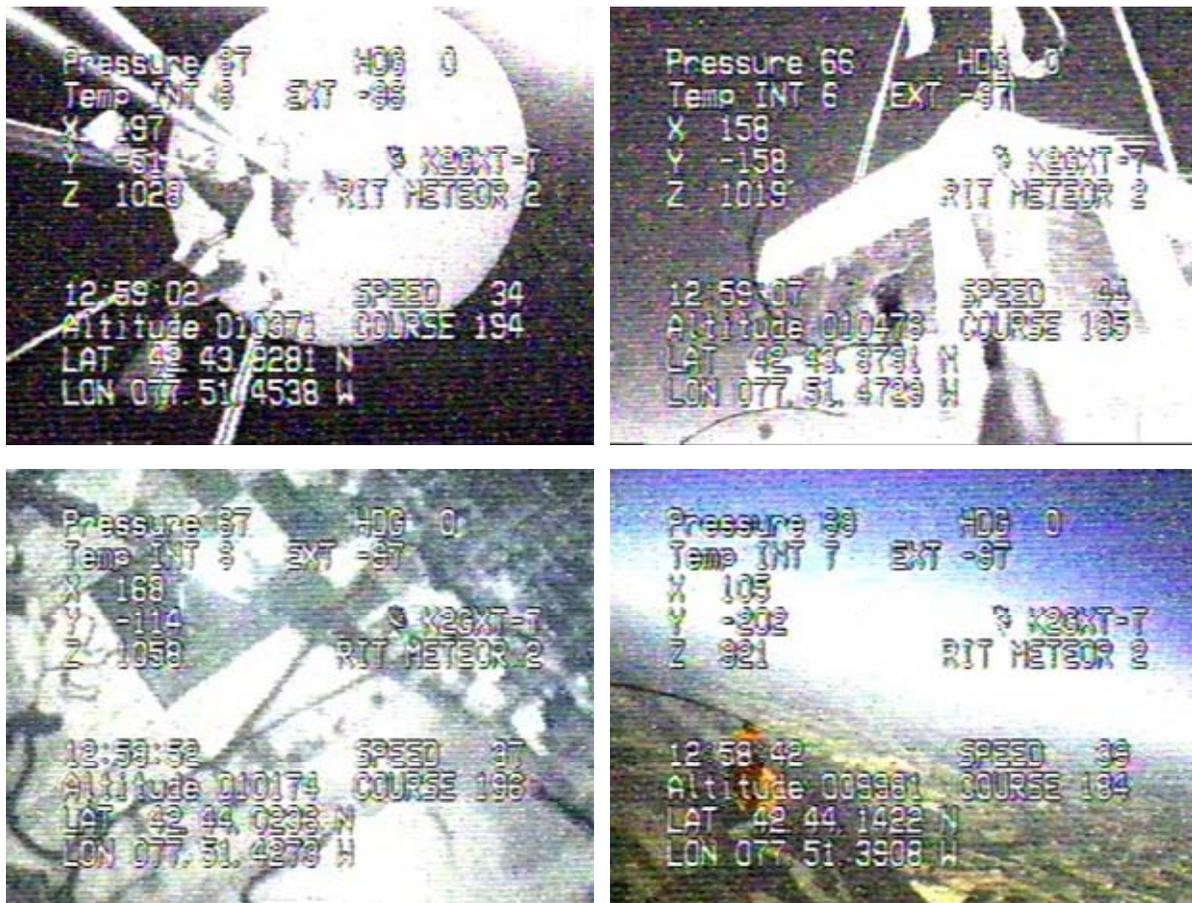
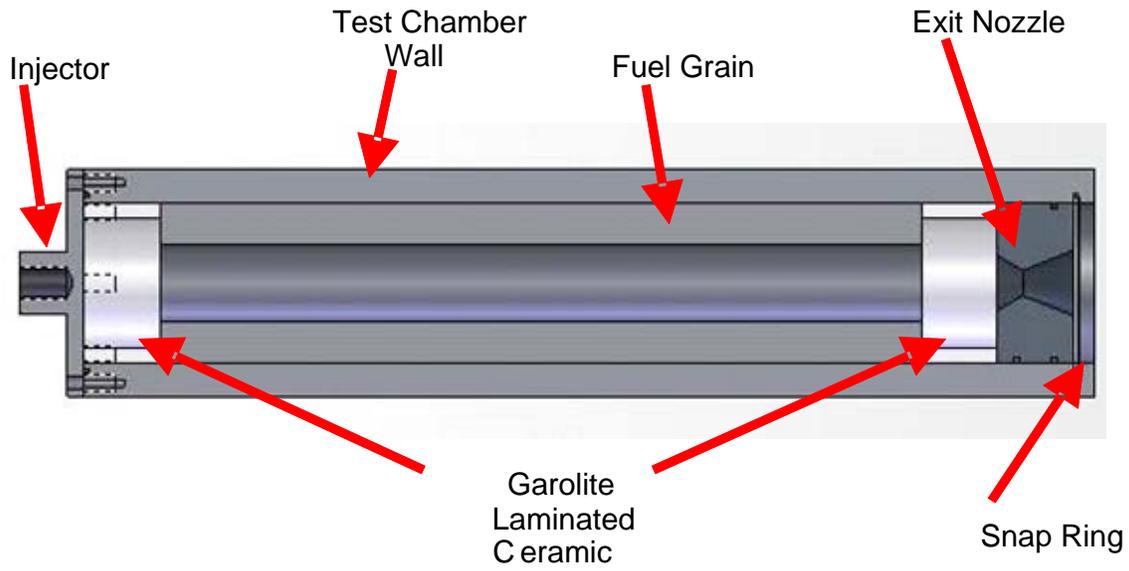


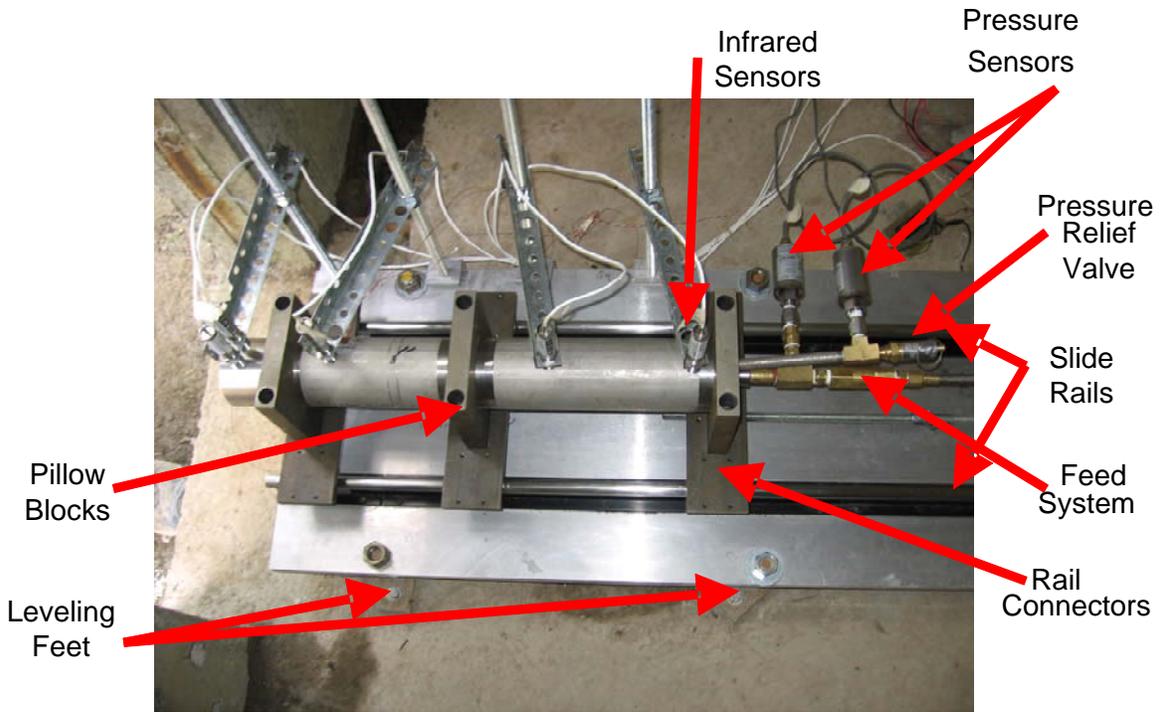
Figure 5. Life video feeds looking up, at the instrumentation platform, down and at the horizon. Telemetry is overlaid on the video stream for redundancy.

The current rocket engine does not meet the 1/10 redundant structure to propellant ratio, required for flying. Its purpose is to acquire experimental data related to optimal fuel grain geometry, oxidizer flow rate, igniter and

nozzle geometry. Figure 6a shows the rocket engine cross section. Figure 6b shows the test stand with the rocket engine and sensors in place. Temperature was recorded in several locations along the body of the engine using



A)



B)

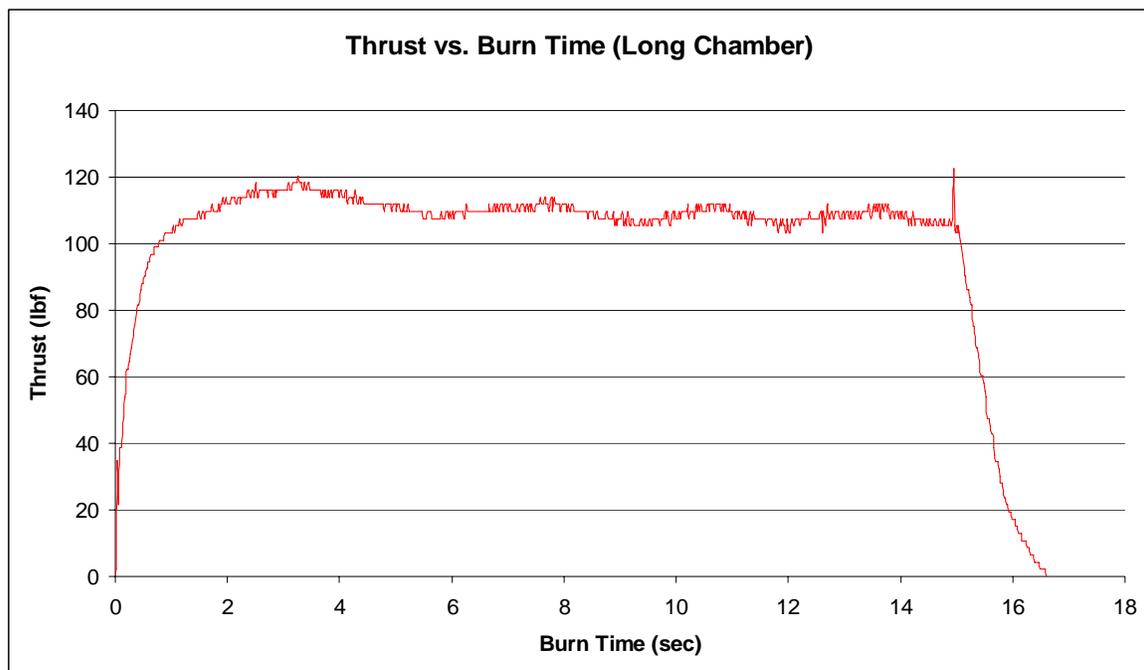
Figure 6. A) Rocket Engine Cross-Section. B) Test Stand.

non-contact infrared sensors. For these tests the NOX was fed in from an external tank. The thrust was measured using a compression load cell, not visible on the right. The load cell is calibrated using a system of pulleys and

weights. The entire test stand is placed inside a 14'x7'x7' concrete culvert, with 6'' thick walls. The only opening during testing is on the exhaust plume side. A snapshot of one of the successful firings and its associated thrust



A)



B)

Figure 7. A) Snapshot of test burn. B) Thrust curve of the same.

curve is shown in Figure 7. The burn has been intentionally limited to ~15 seconds to avoid the complete burn of the HTPB. The thrust is not as high as expected because the supersonic nozzle geometry has not been optimized.

During the academic year 2006/2007, a second senior design team will continue the ground tests, to achieve reliable ignition and optimal nozzle geometry. Time permitting the team will also start the structural design of the flying engine.

An important component of the launch vehicle is the Inertial Navigation and Guidance System. A graduate student is currently working on the Inertial Navigation System (INS), which will generate attitude and location information during the powered flight. An Inertial Measurement Unit ¹⁶, (IMU) weighing only 150 g, outputs 3D acceleration, 3D angular rate and 3D magnetic field data. This data is then processed by the INS. The INS implements a Kalman Filter, followed by a Direction Cosine Matrices block. The implementation of the INS is currently being investigated using three

approaches: (1) off-the-shelf MSP430 Microcontroller, (2) full custom hardware block using the Virtex-II-Pro Field Programmable Gate Array, and (3) a combination of off-the-shelf Microcontroller and custom hardware block. The winning solution will be the one with the best tradeoff between weight – power consumption and execution time.

As can be inferred from Table 1, the weight allowance for these systems and their associated power sources will have to be in the tens to hundreds of grams. We are confident that through the use of highly customized hardware and software, with which the authors have experience, these weight targets can be achieved. The Guidance System, (GS) and the Attitude Control System (ACS) will be developed by graduate students during the academic year 2006/2007.

The Satellite

The satellites that will be used during the first test flights will contain exclusively redundant

Table 2. Development Plan.

Phase	Activity	Objectives	Duration / Status	Students Involved
1	Instrumentation Platform design and testing.	Design, implementation and testing of a high altitude balloon tethered instrumentation platform for use in Phase 4.	Started in Fall 2003 – ongoing	Platform Team 1 – 7 Platform Team 2 – 2 Platform Team 3 – 5
2	Rocket design and testing.	Design, implementation and ground testing of a hybrid propellant rocket engine.	Started in Fall 2005 with the design of the 4 th stage, – on going	Rocket Team 1 – 8
3	Pico-Satellite design, construction and testing.	Design, implementation and ground testing of a Test-Satellite.	Started in Spring 2004 – on going	Satellite Team 1 – 4
4	Sub-orbital test flight of one rocket stage.	Test stage and guidance system	Summer 2007 – tentative date	
5	Complete Launch System testing.	The airborne testing of the complete launch system with the launch of an earth remote sensing Pico-Satellite.	Pending successful completion of previous stages	
6	Launch System improvements and upgrades / Pico-Satellite developments	Improve and upgrade the Launch System with state-of-the-art technologies / Develop Pico-Satellites tailored for scientific space experiments	Indefinitely	Platform Team ~ 6 Rocket Team ~ 6 Satellite Team ~6

navigation, telemetry and video monitoring equipment. Once the functionality of the main systems is proven, the first target interface will be for the Cubesats. A specific interface will then be developed, one that will take full advantage of the primary payload status of the small satellite. A senior design student team will start the development of the first test satellite during the academic year 2006/2007.

Development Plan

Table 2 outlines the current development plan for the custom launch system. Target dates are tentative because of the uncertainty of funds. All activities performed so far have been funded by the Rochester Institute of Technology – cash, and by more than ten companies – in-kind donations.

BENEFITS OF THE LAUNCH SYSTEM

We believe that the development of the proposed custom launch system will offer the following benefits:

- Regular launch opportunities for the small satellite developer's community at a cost of \$100-200K per launch.
- Domestic launches.
- Short integration timeline and responsive launch.
- The small satellite has primary payload status, i.e. it can choose its own orbit parameters.
- Hands-on education and multi-disciplinary experience of future aerospace engineers and scientists, during the development and future continuous improvement of the launch system and the small satellites.
- The launch system and satellites volume and mass constrains, will force the use of new technologies, becoming test beds for space qualification.

In light of the above, the authors hope that the small satellite community, the sponsoring agencies and aerospace companies will start and continue to support the development of this launch system.

CONCLUSION

A custom launch system for satellites with a mass of 1 kg or less has been proposed. The concept and its development to date have been presented. Once it will become operational, the launch system will offer an unmatched, unique service to this class of satellites. The intention is not to compete with any current commercial launcher, but rather to fill an unmet need, ultimately complementing existing launch services.

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