Development of Miniature Hybrid Rockets for Orbital Upkeep and Transfer Applications in Nano/Pico-Satellites
DEVELOPMENT OF MINIATURE HYBRID ROCKETS FOR ORBITAL
UPKEEP AND TRANSFER APPLICATIONS IN NANO/PICO-SATELLITES

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By

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Abstract

This research has developed and constructed a successful miniature hybrid rocket test assembly for research into hybrid rocket scaling issues. It is hoped that such systems will find eventual integration into small satellites. The test assembly houses 3” fuel grains and provides an inlet for gaseous oxidizer. Although far from complete, preliminary data were taken with PMMA and GOx and some useful conclusions were drawn that will benefit future research into this topic. Fuel grains with combustion chambers whose diameters range from 1/16” to 3/8” were tested at different flow rates and thrust data gathered. Maximum flow information was gathered for the 3/8”, 5/16”, ¼”, and 1/16” fuel grains. Supersonic exhaust was found to exist when chamber diameters were at or below 3/16”. Thrust scaling trends were obtained for the 3/16” and 1/8” grains. No quenching limit was found to exist in these grain sizes.
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Introduction

The trend of today’s satellites is one of diminishing size. These small satellites bear names such as “nanosat”, or “picosat”. A nanosat is defined as a small satellite with a mass that ranges from 1 kg to 10 kg and a picosat has a mass that ranges from 0.1 kg to 1 kg. Figure 1 shows three picosats used by Tethers Unlimited Inc.’s (TUI) MAST (Multi-Application Survivable Tether) experiment early in 2007. In this experiment, the tether inspector traversed the length of the tether to inspect, “how quickly the tether experiences damage due to impacts by micrometeoroids and orbital debris, as well as due to erosion by atomic oxygen and UV light” [10]. Size is demonstrated relative to the size of a quarter in the bottom of the figure.

This sizing trend of satellites is justified by their low costs relative to that of their full-sized cousins. Small statures yield low materials cost. Complexity and versatility are somewhat diminished with this size of satellite, but reduced complexity offers the
benefit of reduced manufacturing costs. Additionally, such low masses require little
thrust to propel them into low earth orbit (LEO), so launch costs are diminishingly small.
With size proportional to cost, scale becomes one of the biggest advantages of these next
generation satellites.

Small satellites have seen some space applications after piggybacking on a launch
vehicle and being placed into LEO. However, these are all short term missions,
ultimately limited in scope by propulsive restrictions. Small thrusters have been
researched for small satellites, providing attitude control and orbital upkeep abilities. An
example of a cold gas thruster can be seen in Figure 2, developed by The Aerospace
Corporation for applications in picosats [4]. All small satellites to date, however, have
been lacking in the ability to perform the orbital insertion, orbital transfer, and de-
orbiting maneuvers necessary for effective long-term space missions. That is, they lack
the versatility to perform a range of tasks for an extended period of time under their own
internal propulsive power.

Aerospace COSA (365 gram, 10cm diameter)
Figure 2: A cold gas propulsion module designed by The Aerospace Corporation for future
picosat orbital upkeep capability.
This research aims to design and develop a miniaturized hybrid rocket test assembly able to measure thrust, flow rate, and pressure. While systematically scaling down the size of the rockets and adjusting oxidizer mass flow rates, some preliminary data will be taken and analyzed. Additionally, combustion intensity will be observed and regression rate will be calculated. This data will be used in an attempt to establish a design protocol for future experiments into the scaling issues associated with miniaturized hybrid rocket systems.

Background

A. Propulsion considerations

In searching for an appropriate propulsion choice for nano and picosats, a number of chemical rocket systems are readily available and attainable. Table 1 displays the propulsion systems potentially suitable for small satellite systems. Monopropellant systems such as cold gas thrusters offer a unique degree of simplicity, especially pertaining to propellant storage issues. These systems produce thrust by releasing pressurized propellant (in the form of a single chemical species). Unfortunately, values of specific impulse (thrust per unit mass of propellant) for this type of system are not great enough for orbital maneuvering.
Table 1: Comparison of propulsion system types (adapted from [4])

<table>
<thead>
<tr>
<th>Type</th>
<th>Propellants</th>
<th>Thrust (N)</th>
<th>Isp (s)</th>
<th>Propellant mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thermodynamic</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Cold Gas</td>
<td>$N_2$, $He$, $H_2$</td>
<td>0.2 to 2700</td>
<td>60 to 225</td>
<td>195</td>
</tr>
<tr>
<td>Liquid</td>
<td>$H_2/LOX$, $RP-1/LOX$, $N_2H_4$, $N_2H_4/N_2O_4$</td>
<td>10 to $6 \times 10^6$</td>
<td>270 - 530</td>
<td>100</td>
</tr>
<tr>
<td>Solid</td>
<td>$Al/\text{NH}_4\text{ClO}_4$, Asphalt/$\text{NH}_4\text{ClO}_4$</td>
<td>0.1 to $1.2 \times 10^7$</td>
<td>250 to 300</td>
<td>104</td>
</tr>
<tr>
<td>Hybrid</td>
<td>$\text{HTPB}/\text{LOX}$, $\text{HTPB}/\text{PE}$</td>
<td>10 to $3 \times 10^7$</td>
<td>250 to 350</td>
<td>101</td>
</tr>
</tbody>
</table>

Liquid systems on the other hand, boast the highest performance values of any chemical rocket. Liquid rockets, as the name suggests, utilize the combustion of liquid fuel and liquid oxidizer to produce thrust. Despite the performance benefits of liquid systems, the technology required to effectively control two liquid phase propellants, which often require being stored cryogenically, is quite impressive. The use of liquid systems in small satellites would necessitate the miniaturization and installation of the same amount of hardware found in larger systems. This considered, liquid rockets are not cost effective for miniature satellites.

In contrast to liquid rockets, solid rocket systems are substantially simpler yet yield a somewhat poorer performance. Solid rocket motors combine solid phase fuel and solid phase oxidizer to produce thrust. Propellants are mixed during manufacturing into a solid fuel grain, prior to actual combustion. This results in some safety issues concerning solid rockets. Solid fuel grains are volatile, as any small spark, flame, or even sufficient thermal conduction may act as a catalyst for grain combustion. Care must be taken during manufacturing and transport in order to prevent the creation and propagation of
cracks in the grain. These can be very dangerous, as an undetected crack prior to combustion is likely to cause premature ignition of excess fuel and oxidizer within the crack, the result of which would be devastating. Additionally, once combustion in a solid rocket is started, it is continued through to completion, unable to be adjusted to fit mission parameters. For these reasons, the solid rocket motor was removed from consideration for applications in small satellites.

**B. Hybrid Rocket Motors**

Hybrid rockets combine the performance of liquid rockets with the simplicity of solid rockets. In actuality, the simplicity of a hybrid falls between solid and liquids rockets, and performance values of hybrids are between those for liquids and solids. In many ways, hybrids utilize the advantages of both systems. Strictly speaking, a hybrid rocket is one in which the fuel and oxidizer are found stored as different states of matter. For example, a hybrid may use a liquid fuel and a solid oxidizer. The vast majority, however, use a solid fuel with a liquid or gaseous oxidizer. Similar to the solid rocket, the solid phase is called a grain, through which a bore is machined. This bore serves as the rocket’s combustion chamber. The liquid phase is channeled through the combustion chamber via a pressurized source or pump. Once combustion is initiated in the grain’s combustion chamber (via an ignition mechanism or a hypergolic fuel/oxidizer combination), the exhaust is passed through a nozzle to increase the flow velocity and produce thrust.

Hybrid rockets boast a number of exploitable advantages which make them a prime candidate for miniaturized propulsion systems. Compared to liquid propulsion
systems, hybrids have a reduced level of complexity. When it is considered that effectively dealing with liquid propellants requires the use of pumps, injectors, valves, regulators, and other such hardware elements for each liquid phase propellant, not to mention redundancies for these systems, it quickly becomes apparent why the elimination of one liquid phase propellant greatly reduces the system’s complexity and cost.

Compared to solid propulsion systems, hybrids have an increased level of safety. The separate states of fuel and oxidizer mean that the fuel (or oxidizer) grain is itself inert and unreactive unless in the presence of the oxidizer (or fuel). This allows for safer manufacturing and transport. Cracks in the fuel grain are also less dangerous for this reason, as cracks do not contain any pre-mixed oxidizer.

By controlling oxidizer mass flow rates, the user is able to influence the fuel/oxidizer ratio within the combustion chamber, which affects combustion intensity and thus the amount of thrust produced. Throttling of a rocket in this manner is a highly valued advantage for performing space missions. A throttleable rocket is significantly more versatile than an unthrottleable one. This control prevents the hybrid rocket from necessarily utilizing all of its fuel at once, as is the case for the solid rocket, allowing fuel to be conserved for later use.

Fuel and oxidizer combinations are nearly endless. The selection of fuel and oxidizer lies in the hands of the designer, who may consider factors such as specific impulse ($I_{sp}$), combustion temperature, or reaction products. Fuel and oxidizer may be chosen such that the combustion reaction results in environmentally friendly exhaust products. Although this is not a concern in space applications, as there is no environment to affect, it is an advantage for researchers who wish to study the effects of these rockets.
C. Scaling issues

Though much attention has been given to the operational characteristics and performance advantages of hybrid rocket systems in general, this research primarily focuses on the scaling issues associated with miniaturized hybrid rockets. The focus on small scale satellite applications demands understanding of how performance characteristics scale with size. When systems are scaled to significantly different scales, there exist a set of observations which describe the way scaling occurs when an individual dimension is altered. These scaling laws are easily understood with the help of an example. Most easily considered is a simple three-dimensional cube with each side having a unit length, as seen in Figure 3. Increasing the unit length by a factor of two increases the area by a factor of four and increases the volume by a factor of eight. Similarly, reducing the unit length by a factor of two reduces the area by a fourth and reduces the volume by an eighth. This is because area scales with the square of the length, while volume scales with the cube of the length.

![Figure 3: Visual description of classic scaling laws. Area and volume are shown to scale differently as the length of a side is adjusted.](image)

Where rockets are concerned, parameters such as thrust, $I_{sp}$, and many others must be considered in the design process. A chart displaying some of these relevant equations
is given in Figure 4 [1]. Obviously, these equations contain relationships which are much more complicated than the simple area and volume relationships previously discussed. Each variable scales differently, making the net effect of reducing combustion chamber diameter on that variable very difficult, perhaps even impossible, to predict. It is for this reason that scaling relationships for miniature hybrids are better suited to be empirically determined.

Another concern when scaling is the limit of that scaling, especially concerning combustion. Seen in Figure 5 is a schematic depicting the combustion process within a hybrid rocket. Combustion provides sufficient heat to vaporize the exposed surface of the fuel grain, which is mixed with the oxidizer flowing closer to the chamber’s center. Near the grain’s surface, this mixture is too fuel rich for combustion to occur, and nearer

\[
\begin{align*}
\text{Known:} & \\
\quad P_t &= \text{Total Pressure} & \quad R &= \text{Specific Heat Ratio} \\
\quad T_t &= \text{Total Temperature} & \quad \gamma &= \text{Gas Constant} \\
\quad p_0 &= \text{Free Stream Pressure} & \quad A_e &= \text{Area} \\

\text{Mass Flow Rate:} & \\
\dot{m} &= \frac{A_e P_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} \left( \frac{\gamma + 1}{2} \right) \left( \frac{\gamma - 1}{2} - \frac{M_e^2}{M_e} \right) \\

\text{Exit Mach:} & \\
\frac{A_e}{A_e} &= \left( \frac{\gamma + 1}{2} \right) \left( \frac{1 + \frac{\gamma - 1}{2} M_e^2}{M_e} \right)^{\frac{\gamma + 1}{2(y-1)}} \\

\text{Exit Temperature:} & \\
\frac{T_e}{T_t} &= \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right)^{\frac{1}{\gamma - 1}} \\

\text{Exit Pressure:} & \\
\frac{p_e}{p_t} &= \left( 1 + \frac{\gamma - 1}{2} M_e^2 \right)^{-\frac{\gamma}{\gamma - 1}} \\

\text{Exit Velocity:} & \\
V_e &= M_e \sqrt{\gamma RT_e} \\

\text{Thrust:} & \\
F &= \dot{m} V_e + (p_e - p_0) A_e
\end{align*}
\]

Figure 4: Several equations used in the design of rockets. Each variable obeys different scaling laws.
the center of the combustion chamber, the mixture is too fuel lean for combustion to occur. There exists a region between these two extremes in which the fuel/oxidizer ratio can and does support combustion. This region, called the flame zone, lies within the flow’s boundary layer, near the surface of the fuel grain.

![Figure 5: Combustion diagram for hybrid rocket motors.](image)

At very small scales, combustion is unable to be established and maintained. This phenomenon, called quenching, is due to scaling laws very similar to those in the example presented by Figure 3. As the diameter of the combustion chamber is reduced, grain surface area and flame volume also are decreased, but at different rates. These rates are such that the ratio of surface area to volume increases with decreasing diameter. A given flame produces a set amount of heat per unit volume, and a given fuel absorbs a set amount of heat per unit of surface area. The disproportionate scaling of these geometries may result in a flame which is unable to provide enough heat to vaporize the surrounding fuel, thus quenching the flame. There exists some ratio of surface area to volume (and some chamber diameter) which results in a flame able to produce just enough heat to vaporize the surrounding fuel. This interface between quenching and combustion is
called the quenching limit. Determining this lower limit of combustion will provide sizing restrictions for small scale hybrid rocket motors.

**Experiment**

A photograph depicting actual experimental hardware is shown in Figure 6, below. Equipment was secured to the composite board surface of a lab cart, allowing transportation of the entire system. This allowed tests to be conducted outdoors in safety despite being constructed in the lab.

![Figure 6: Complete miniature hybrid rocket test stand](image)
A. Propellants

To study the scaling issues associated with small rocket systems, polymethyl methacrylic (PMMA), or plexiglas®, was used for fuel and gaseous oxygen (GOx) was used as oxidizer. Little consideration was given to other fuels and oxidizers, as the emphasis of this study lies in scaling issues and not general performance or efficiency. The fuel and oxidizer combination was chosen based on ease of obtaining experimental completion. PMMA and GOx were chosen for their low cost and commercial availability, as well as having environmentally friendly exhaust products. These two propellants ignite at to produce only water and carbon dioxide. Additionally, PMMA is easily machinable, a preferred fuel characteristic. Eighteen fuel grains were cut from a 1” diameter, 8’ rod, each to a length of 3”. Axial bores were drilled into each of the grains, ranging from diameters of 1/16” to 3/8” in 1/16” increments. This yielded 3 grains of each of 6 bore sizes. PMMA fuel grains can be seen in Figure 7, arranged from smallest chamber diameter (left) to largest chamber diameter (right). Also included is an un-drilled sample fuel grain.
B. Set-up

Shown in Figure 8 is a simple schematic depicting the flow system to be used for testing. In this system, pressurized propane and GOx flow from storage tanks into the premix chamber of the rocket test stand. Omega® mass flow controllers, such as the one shown in Figure 9, were installed in each flow line, enabling mass flow variability. The propane mass flow controller limits the propane flow to 10 SLPM (standard liters per minute) while the GOx mass flow controller limits GOx mass flow to 20 SLPM. Each flow controller houses an electronic valve, which is used to determine mass flow. Flow lines include flashback arrestors which prevent flame propagation into the tanks. The GOx line includes an added safety valve in the form of a check valve, which prevents any backwards flow in the line.
C. Test Stand

A solid steel test stand was designed and machined to hold the PMMA fuel grains during ignition and burn. Figure 10 shows the design and some preliminary dimensioning of the stand. Labels for various parts are given in Figure 11. Four main parts compose the test stand: the premix chamber, inlet block, joining rods, and nozzle
block. Both blocks were shaped to a 3” x 3” x ½” size and 5 holes were drilled as shown in Figure 10. Additionally, 1” diameter recesses were cut in the center of each block for the fuel rods to fit into. Using a lathe, a nozzle was machined into the nozzle block with a half-angle of 5°. Inlet and nozzle blocks are held together by ¼” diameter steel rods fed through corner holes. The premix chamber was machined to a 2” diameter, 3” long steel cylinder, hollowed out for propane and oxygen mixing. Eight screws hold together the pre-mix chamber and inlet block.

Figure 10: Preliminary design and partial dimensioning of stand assembly
The test assembly is secured with four aluminum ell brackets. Figure 12 shows the finished product.
The measurement of thrust required the machining of two 3/4” aluminum plates, by which the test stand was mounted on a 6-axis force balance. One plate was bolted to the stand’s ell brackets as a base. The other plate was bolted perpendicular to the first as well as to the force balance, giving the test stand an “upside down” orientation. Figure 13 shows the test stand mounted on the force balance.

![Figure 13: Test stand mounted on 6-axis force balance.](image)

**D. Procedure**

Initial plans for ignition involved filling the premix chamber with a mixture of propane and oxygen, then igniting the mixture via resistive heating of a glow plug. When testing this ignition process, the method was found to be more dangerous and more difficult than expected. For these reasons, ignition was obtained via a simpler and safer, yet cruder, method. After mounting the fuel grain, a strip of paper was placed in the
combustion chamber, spanning its length. A low GOx flow rate was allowed to ensure flame retention. Once the flame neared the chamber inlet, GOx flow was increased to the desired value, igniting the rocket. Thrust as well as upstream pressure and flow data were taken using a LabVIEW data acquisition system and a computer. An upstream pressure sensor is integrated into the mass flow controller. Additionally, fuel grains were weighed before and after burn so that average regression rate could be calculated.

Preliminary Data

Data gathered for a 3/8” fuel grain is shown in Figure 14. Thrust data did not seem particularly useful for this size grain, due to large ignition thrust fluctuations and the observation that the exhaust flow was not accelerated to Mach 1 due to over-sizing of the nozzle throat. Unless the exhaust becomes sonic at the throat, the diverging nozzle will serve only to slow the flow and reduce thrust. At a chamber of 3/8” diameter, a flow rate of 20 SLPM was insufficient to create a supersonic exhaust. An example of this subsonic exhaust is shown in Figure 15. For useful thrust to be obtained at this grain size, a higher flow rate and pressure would be required. The 5/16” grain displays the same problem in Figure 16. Only one 3/8” and 5/16” grain was tested. Without the ability to increase flow rate above 20 SLPM, every test would yield the same results.
Figure 14: 3/8" grain thrust and pressure data.
Figure 15: Subsonic combustion exhaust.
When a 20 SLPM flow rate was attempted with a $\frac{1}{4}$” grain a flame bleed was experienced, in which a section was burned at the inlet surface of the grain, allowing flow to escape and pressure to drop. To assure this wasn’t a mistake, another grain was tested at 20 SLPM with the same results. This confirmed the bleed was due to the excessive oxidizer pressure. When the flow was dropped to 17 SLPM, there was no bleed. This can be seen in Figure 17, asserting that the maximum flow rate for a $\frac{1}{4}$” grain lies
between 17 and 20 SLPM. Also notable, is the fact that the ¼” grain remained subsonic and thus produced low thrust.

![Figure 17: Thrust vs. time data for ¼” grain size.](image)

Thrust is plotted against time for a 3/16” fuel grain in Figure 17. While the grain was not over-pressured even at 19 SLPM, there was a noticeable thrust increase at 17 SLPM compared to 18 and 19 SLPM. This was probably due to a more favorable fuel/oxidizer ratio at flow rates closer to 17 SLPM. Although this observation is not conclusive, it does provide a starting design point for future research using 3/16” fuel grains. The most useful observation obtained from this series of testing was that the exhaust flow from the 3/16” grain did exceed Mach 1, resulting in increased thrust values. An example of this supersonic exhaust is given in Figure 18. This asserts that the critical nozzle throat area lies between ¼” and 3/16”. This may be critical design information for future nozzle designs at these flow rates.
Figure 17: Thrust vs. time data for 3/16” grain size.

Figure 18: Supersonic combustion exhaust.

Thrust data is plotted against time for three 1/8” fuel grains burned at 15, 16, and 17 SLPM in Figure 19. Results for the 1/8” grain are similar to those for the 3/16” grain.
size. Thrust is increased with lower flow rates, likely due to more favorable fuel/oxidizer ratios. Any future design or testing with 1/8” fuel grains should begin with flow rates closer to 15 SLPM than 16 or 17 SLPM.

![Figure 19: Thrust vs. time data for 1/8” grain size.](image)

Due to time constraints from ignition difficulties with such a small fuel grain, only two 1/16” grains were tested. Results from the first test were discarded due to a premature ignition and subsequent abnormal burn in the grain. The second 1/16” grain test was run at 3 psia and was found to be very close to the maximum flow rate a 1/16” grain could withstand. Combustion at this pressure limit yielded some interesting observations. An initial thrust was produced from ignition, but the flame soon diminished, eventually failing to leave the combustion chamber altogether, as seen in Figure 20. The oxidizer flow rate was too high to for useful combustion over the entire grain length, but not too high to prevent combustion altogether. As a result, the flame produced at the inlet was stretched down the length of the chamber, able to vaporize fuel...
but unable to burn it. Once the combustion chamber diameter was wide enough, combustion resumed as normal.

![Figure 20: Flame stretching due to high pressure flow.](image)

Fuel regression rates were calculated and graphed in Figure 21 using measured masses before and after combustion and burn time. Regression rate was nearly constant across the different chamber diameters. The highest and lowest regression rates were calculated for the same size fuel grain, indicating that chamber size has little or no effect on regression rate.
These set of experiments did not yield any information on the quenching limit.

The preliminary data suggest that quenching is not a considerable limitation in the scaling of small hybrid rocket systems.

Figure 21: The effect of combustion chamber diameter on regression rate.
Conclusions

This research does not claim to have sufficiently explored all of the scaling issues discussed. However, despite generally unusable data, some important conclusions could be reached which may be of use to future designers of miniature hybrid rocket systems. This test assembly and preliminary data provide good baseline performance characteristics and some initial design considerations. For example, efficient combustion with 3/8” or 5/16” fuel grains will require flows greater than 20 SLPM and the maximum flow for a ¼” fuel grain lies between 17 and 20 SLPM. Greater thrust values for a 3/16” fuel grain are obtained for flow rates at or below 17 SLPM and greater thrust values for a 1/8” fuel grain are obtained for flow rates at or below 15 SLPM. Quenching was not found to be a problem with small fuel grains.

One of the most significant results obtained from this research was how flow velocity was affected by chamber diameter. At a diameter of 3/16”, the flow was found to be choked, and the exhaust supersonic. Despite an oversized nozzle throat, it was found that the desirable nozzle throat diameter lies between ¼” and 3/16”. Future researchers will want to take this into consideration in their design to ensure supersonic flow independent of chamber diameter. For further study in this area, it would be desirable to have a variable geometry nozzle so that different flow rates could be studied for a given chamber size. Nozzles could be machined independently of the nozzle block with ability to be detached and reattached, allowing nozzles to be switched out for different experiments.
Ignition remains an area in which much improvement could be made. Propane and GOx ignition as originally intended was found to be dangerous and difficult to implement. The paper method used for ignition was crude and subject to human error. Hypergolic fuels and oxidizers would greatly ease ignition troubles, as ignition would be as simple as turning on the oxidizer flow.
References


