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Development and Short-Range Testing of a 100 kW Side-Illuminated Millimeter-Wave Thermal Rocket

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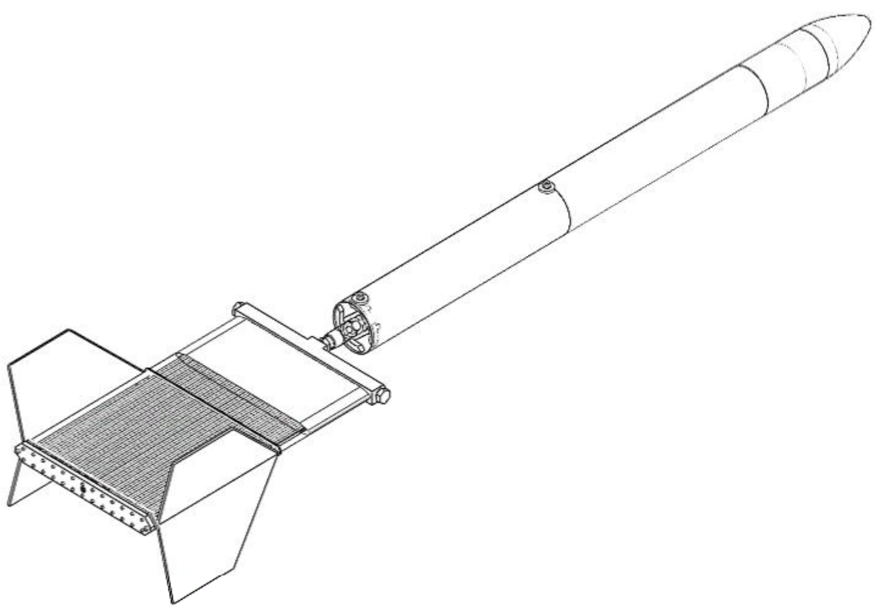

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Topic: Microwave Power Beaming and Propulsion

Abstract. The objective of the phase described here of the Millimeter-Wave Thermal Launch System (MTLS) Project was to launch a small thermal rocket into the air using millimeter waves. The preliminary results of the first MTLS flight vehicle launches are presented in this work. The design and construction of a small thermal rocket with a planar ceramic heat exchanger mounted along the axis of the rocket is described. The heat exchanger was illuminated from the side by a millimeter-wave beam and fed propellant from above via a small tank containing high pressure argon or nitrogen. Short-range tests where the rocket was launched, tracked, and heated with the beam are described. The rockets were approximately 1.5 meters in length and 65 millimeters in diameter, with a liftoff mass of 1.8 kilograms. The rocket airframes were coated in aluminum and had a parachute recovery system activated via a timer and Pyrodex. At the rocket heat exchanger, the beam distance was 40 meters with a peak power intensity of 77 watts per square centimeter. and a total power of 32 kilowatts in a 30 centimeter diameter circle. An altitude of approximately 10 meters was achieved. Recommendations for improvements are discussed.

Keywords: Beamed Energy, Propulsion, Millimeter-Waves, Rocket

INTRODUCTION

Millimeter-wave thermal rockets are a class of beamed energy rocket that utilize millimeter waves to heat a ceramic heat exchanger, which then convectively heats a working fluid. The hot working fluid is then expanded out a nozzle generating thrust similar to a conventional chemical rocket. This system decouples the energy source from the propellant enabling high heat capacity propellants to be used. For example, pure hydrogen propellant could achieve a specific impulse of 700 s with a modest temperature of 1800 K [1]. This differs from the first detailed beamed energy propulsion concept by Kantrowitz in 1972, which used lasers and ablation [2].

The first laboratory scale microwave thermal heat exchanger was developed and demonstrated by Parkin, which used a resonant cavity to focus and heat a single heat exchanger tube [1]. Further laboratory studies with a single tube and resonant cavity were conducted by

Bruccoleri et al. [3], which demonstrated microwave energy could be transferred to a working fluid. The work described here is the progression from laboratory scale studies to a subscale demonstrator, representing the first flight demonstration of a millimeter-wave thermal rocket.

The vehicle was built by the Millimeter-Wave Thermal Launch System (MTLS) project, a project funded by the Defense Advanced Research Projects Agency (DARPA) and the NASA Ames Research center. The goal of the this phase of the program was to advance the state-of-the-art in millimeter-wave thermal propulsion technology and demonstrate a flight test rocket. This work covers the design, construction and flight of the rocket. The millimeter-wave source was the Active Denial System 0 (ADS 0) provided via a partnership with the US Air Force Research Laboratory (AFRL) and the US Army High Energy Laser Test Facility (HELSTF). ADS 0 has a 95 GHz gyrotron with a total power of approximately 100 kW. The beam was focused and directed via an array of millimeter-wave optics to a target of 30 cm diameter. It was designed and built by Science Applications International Corporation (SAIC) and Leidos.

DESIGN CONSIDERATIONS

Based on initial assumptions of the beam transmission and absorption efficiency, Lambot et al. [4] set the basic design for the size and operating characteristics of the millimeter-wave absorbent heat exchanger to be mounted at the aft end of the rocket. The beam was focused down onto the heat exchanger at the launch site at a distance of 40 meters with a peak power density of 77 W/cm^2 with a total power of 32 kW in a 30 cm-diameter circle. The heat exchanger mass was estimated to be 0.5 kg. Compressed Argon gas was used as the propellant for simplicity. Argon is also dense and has a low specific impulse. The low specific impulse was desirable since it maximized thrust for a fixed power. This can be observed from simplified models of thrust and power. The power in the rocket exhaust jet P_j is;

$$P_j = \frac{1}{2}\dot{m}u_e^2 \tag{1)}$$

where $\dot{m}$ is the mass flow rate and u_e is the velocity of the jet which is the specific impulse multiplied by the gravitational acceleration ($u_e = I_{SP}g$). The thrust T, ignoring pressure effects is;

$$T = \dot{m}u_e \tag{2)}$$

Equation 2 can be substituted into equation 1 and rearranged to yield;

$$T = \frac{2P_j}{u_e} \tag{3)}$$

which shows that for a fixed power, thrust is maximized by minimizing the exhaust velocity or specific impulse. To overcome gravity for a small demo flight, maximizing thrust is important but it does not use propellant efficiently. For an actual mission to orbit, much higher-power beams will be used, and the specific impulse will be maximized.

The rockets needed to be pressurized remotely from the launch pad and then transported, which meant the tanks and associated high pressure components needed to have large safety margins for human handling. An aluminum tank, Catalina 9009, was chosen since it is a DOT-certified pressure vessel, making it a good choice for durability. The tank has an internal volume of 361 cm^3 and a mass of 0.47 kg empty. The Catalina tank can hold 130 grams of Argon at 3,000 psi. The targeted rocket mass at liftoff was 1.5 kg due to the limited power of the beam on the vehicle, which meant the airframe, recovery system and propellant feed system had to be approximately 0.5 kg.

VEHICLE DESIGN

The vehicle was designed to have the propellant tank in the airframe with an adapter, valve, and pressure regulation system upstream of the heat exchanger. Ahead of the tank was a bulkhead and a bay for the parachute. The timer, batteries, and switches were mounted in a small bay within the nose cone. This design was modular such that different tanks, airframes and electronics bays could be swapped and replaced. See Fig. 1 for a computer aided design (CAD) depiction of the rocket.

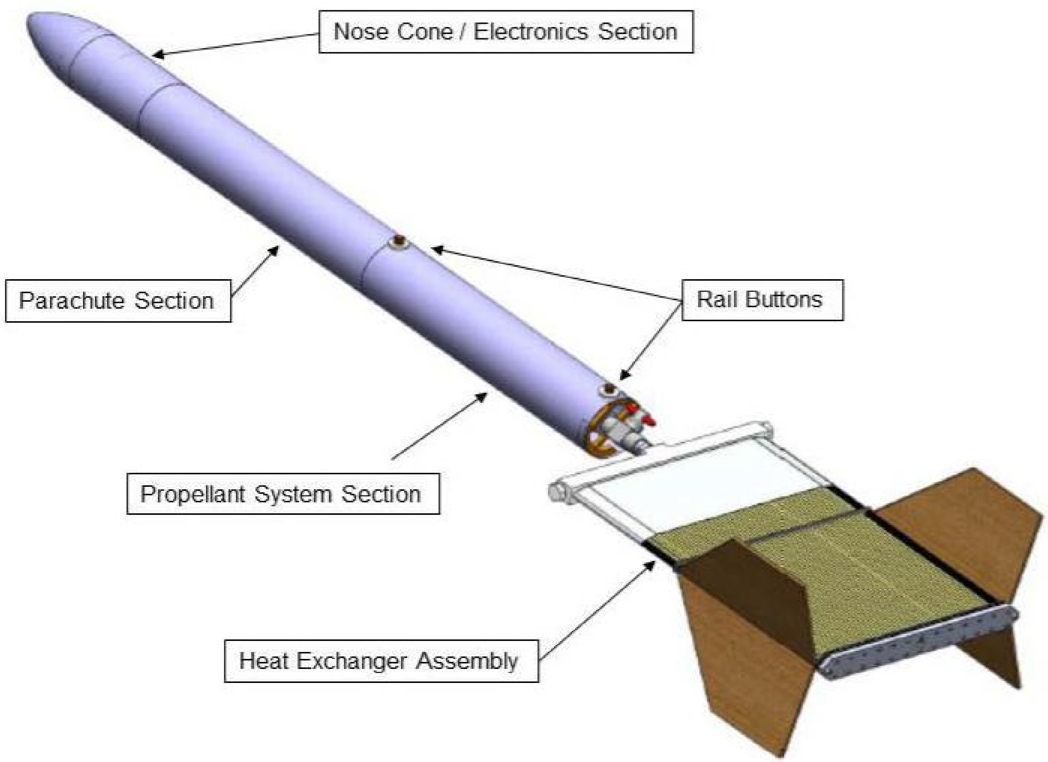


Figure 1. CAD drawing of Rocket.

PROPELLANT FEED SYSTEM

The propellant feed system started with the main propellant tank, which had an adapter machined from aluminum to mate with a ball valve. The adapter had a quick disconnect as a fill valve and a 5,000 psi burst disk for safety. The ball valve was the main release valve for the propellant and was turned externally by a pneumatic actuator. See Fig. 7 for a picture of the actuator. Following the ball valve was a sonic orifice to reduce the pressure in the heat exchanger. See Fig. 2 & 3 for a drawing and picture of the propellant feed system.

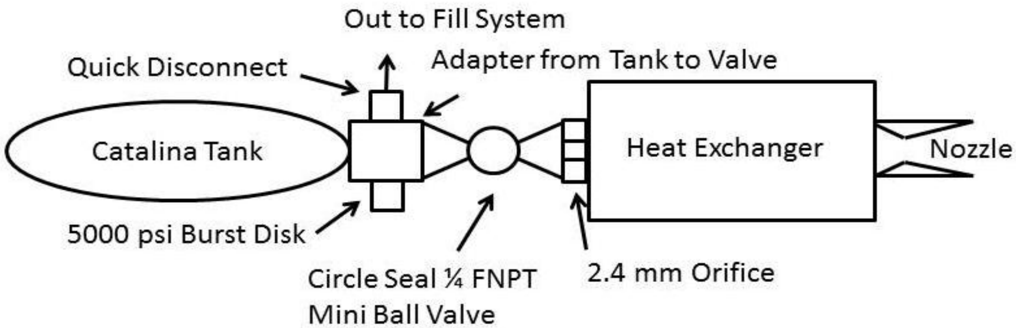


Figure 2. Drawing of propellant feed system.

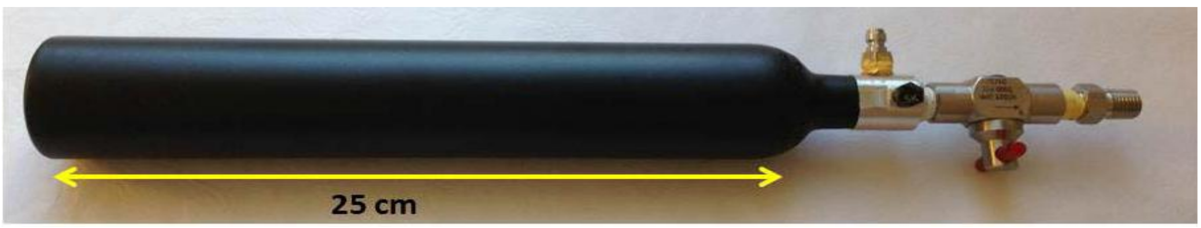


Figure 3. Photograph image of propellant tank, adapter, valve and orifice. Note: The orifice shown is stainless steel and was used for static testing. An aluminum orifice was used for the flight vehicle.

The drawback of using compressed gas as the propellant was a rapid reduction in tank pressure and hence mass flow rate with time. Qualitatively, the tank is losing propellant mass and the gas also cools as it does work during the expansion, which leads to a non-linear function of tank pressure with time. The pressure in the tank can be derived analytically assuming an isentropic process. This modeling provides good insight into the behavior of the gas; however, a significant amount of heat will transfer into the gas since the tank is approximately three times as massive as the propellant, which starts at ambient temperature. The following derivation shows the basics of the pressure model [5].[1] To start, the change in density ρ with respect to time is;

$$\frac{d\rho}{dt} = \frac{\dot{m}}{V_{tank}} \qquad 4)$$

where $\dot{m}$ is the mass flow rate and V_{tank} is the tank volume. The mass flow rate is the product of the density ρ^*, orifice area A^*, and gas velocity u^*;

$$\dot{m} = \rho^* A^* u^* \qquad 5)$$

[1] The derivation follows from the Iowa State University, class Ae311L homepage. "Blowdown of a Pressurized Tank".

where the superscript * represents sonic, or critical, conditions at the orifice. The tank pressure starts at approximately 200 atm and flow can be assumed sonic through the orifice for the majority of the tank blowdown. The gas velocity at Mach 1 is;

$$u^* = \sqrt{\gamma R T_g^*(t)} \tag{6}$$

where γ is the ratio of the specific heat at constant pressure to the specific heat at constant volume, R is the specific gas constant and $T_g^*(t)$ is the gas temperature at the sonic orifice. Observing the relations of density and temperature for an isentropic process;

$$T_g(t) = T_g(0)\left(\frac{\rho_g(t)}{\rho_g(0)}\right)^{\gamma-1} \tag{7}$$

where $T_g(t)$ and $T_g(0)$ are the gas temperature with time and initial gas temperature respectively, and $\rho_g(t)$ and $\rho_g(0)$ are the gas density with time and initial gas density respectively. The temperature and density at the sonic throat are also functions of the tank stagnation conditions and γ;

$$T_g^*(t) = T_g(t)\frac{2}{\gamma+1} \tag{8}$$

$$\rho_g^*(t) = \rho_g(t)\left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}} \tag{9}$$

Equation 9 can substituted into equation 5, and equation 7 into 8 and then 6 and back to 5 to have a function for the mass flow rate with density and the initial tank temperature. A differential equation is found when substituting the mass flow rate into equation 4 and this can be solved and substituted into the ideal gas law;

$$P = \rho R T \tag{10}$$

where P is the pressure. The analytical solution of tank gauge pressure with respect to time is;

$$P_g(t) = \left(P_g(0) + P_{atm}\right)\left[\frac{(1-\gamma)\sqrt{\gamma R}\,A^*}{2V_{\text{tank}}}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\sqrt{T_g(0)}\,t + 1\right]^{\frac{2\gamma}{1-\gamma}} - P_{atm} \tag{11}$$

where $P_g(0)$ is the initial tank gauge pressure and P_{atm} is the ambient atmospheric pressure. This function drops quickly with time. To reduce the rapid drop in pressure, the tank volume can be increased or the sonic orifice diameter can be reduced.

The sonic orifice was used to reduce pressure since it was lightweight and simple to implement. A proper pressure regulator for approximately 50 g/s argon would have weighed on the order of 1 kg, far too much to fit into the 0.5 kg mass budget of the airframe, recovery system and feed system. Assuming an isentropic process, the mass flow into the heat exchanger is proportional to the tank absolute pressure $P_g(t)_{abs}$ and sonic orifice area;

$$\dot{m} \propto P_g(t)_{abs} * A^* \tag{12}$$

The mass flow leaving the heat exchanger is the same as entering during steady state and is proportional to the heat exchanger absolute pressure $P_{HX}(t)_{abs}$, and the nozzle throat area A_t;

$$\dot{m} \propto P_{HX}(t)_{abs} * A_t \qquad 13)$$

The pressure in the heat exchanger is proportional to the tank pressure times the area ratio of the orifice to the nozzle throat;

$$P_{HX}(t)_{abs} \propto \frac{P_g(t)_{abs} * A^*}{A_t} \qquad 14)$$

This allows the pressure in the heat exchanger tubes to be reduced by using an orifice smaller than the throat area. This solution was simple, though it still meant the mass flow rate was changing with time, which was not suitable for efficient heat exchanger operation. An orifice diameter of approximately 2.4 mm was used in the final rocket configuration.

AIRFRAME AND RECOVERY SYSTEM

The main airframe was made from ≈ 65 mm diameter wound paper tubing and a plastic nose cone from Estes, as they were lightweight, easy to work with and readily available. The propellant tank was held in via plywood centering rings and a plywood bulkhead separated the propellant tank section from the parachute. For stabilization, 4 fins were mounted on the side of the heat exchanger with a 15 degree dihedral. The fins were made from balsawood laminated with fiberglass to make a stiff, lightweight structure.

A 36 inch 1.1 ounce nylon parachute made by Top Flight Recovery was used for recovery. It was deployed with ≈ 1.3 gram of Pyrodex “P” with a filament set off via an electric timer, a MicroTimer2 by PerfectFlite. A Nomex cloth along with Estes wadding protected the parachute from the heat of the Pyrodex. Range safety considerations required that a tether be attached to the rocket to prevent it from exceeding 200 meters in altitude. The tether was attached to a switch which fired the Pyrodex charge to deploy the parachute if the rocket trajectory exceeded the tether length. A switch was used to reduce the strength requirement of the tether and allowed for the use of Kevlar fishing line material. The canisters to hold the Pyrodex, made by Pratt Hobbies Inc., had two filaments such that the timer and tether could be wired independently. The timer and tether both had switches wired with their respective batteries to turn them on and off right before flight. The timer also used a break-wire to initiate the countdown. See Fig. 4 for a schematic of the timer and tether.

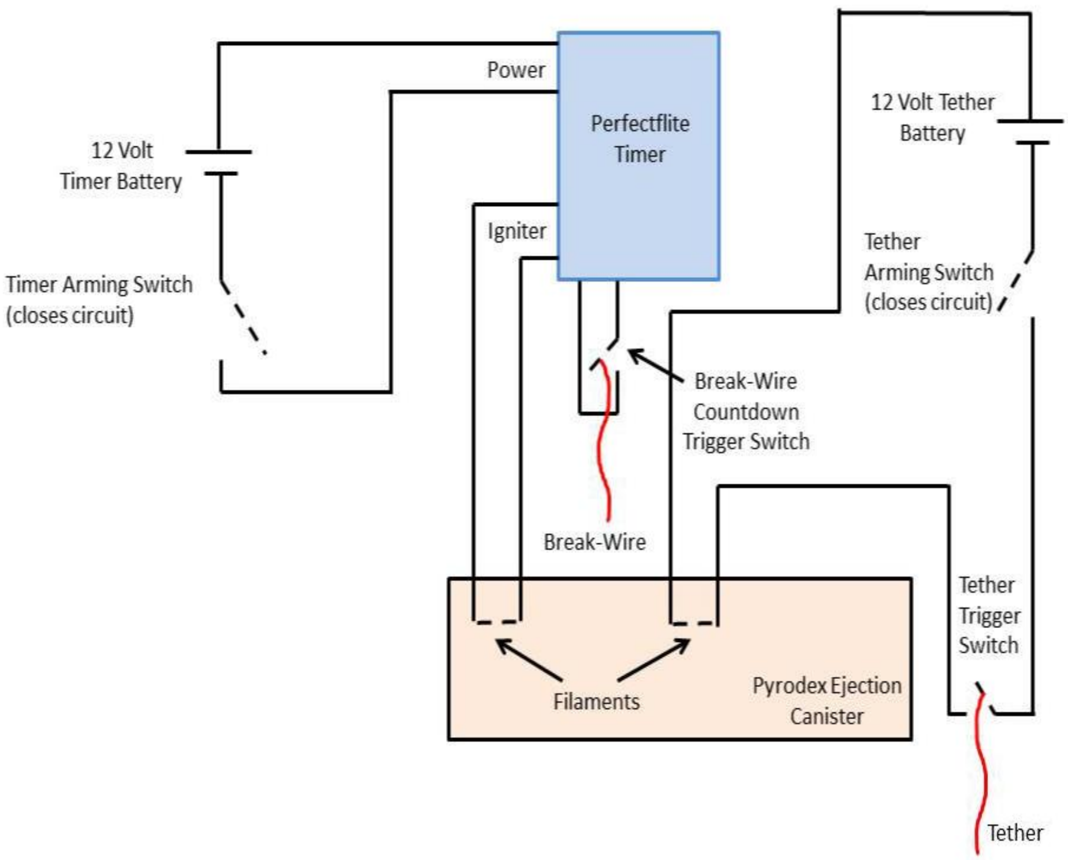


Figure 4. Wiring diagram for timer and tether.

MILIMETER-WAVE RADIATION SHIELDING

The rockets needed a reflective mm-wave radiation protection system to shield against excessive heating in the regions outside of the heat exchanger. Aluminum foil (0.018 mm-thick), and aluminum tape were used for this purpose. The foil was glued to the fin material during the fiberglass lamination of the balsawood and glued to the body tubes via fiberglass resin or epoxy. The tape was used to cover complex contours or holes in the foil. Aluminum's low electrical resistivity enables thin films to be used to reflect almost all of the millimeter waves. A simple model is presented by Buffler for the power reflected from a microwave susceptor film [6]. The power reflected P_r is;

$$P_r = \frac{1}{(1+2r)^2} \tag{15}$$

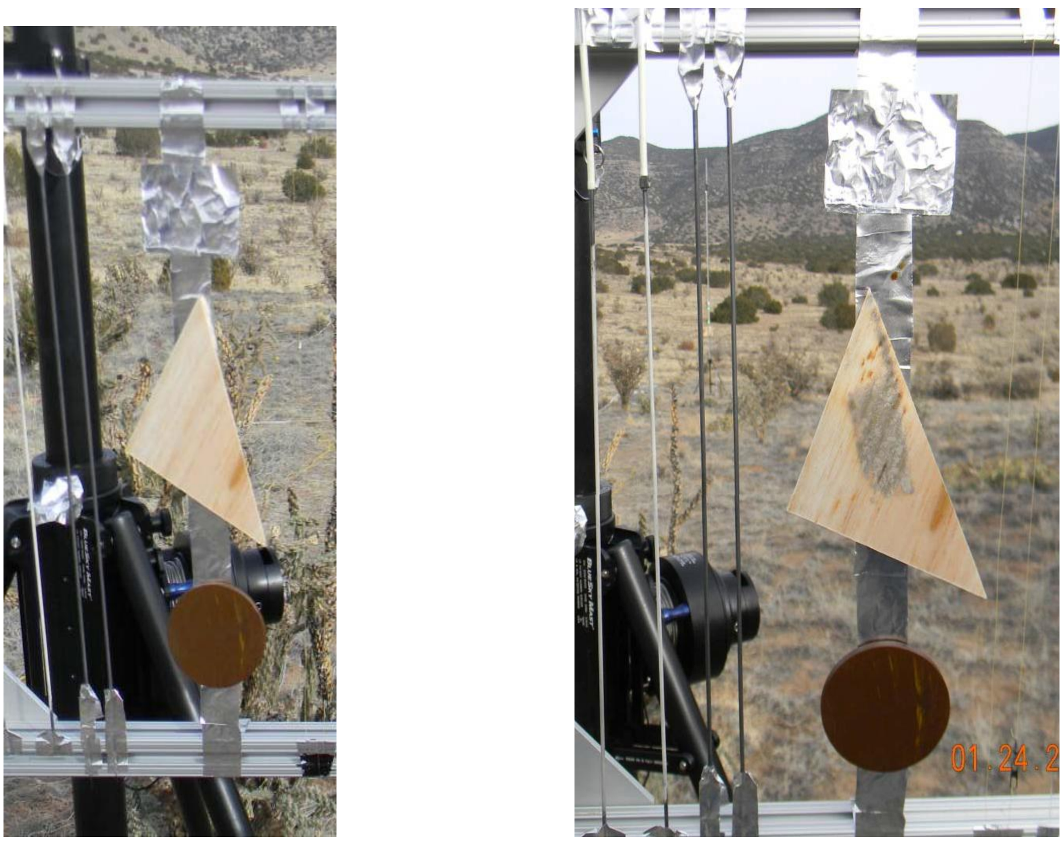
where r is the ratio of the film resistance to the free space impedance, Z_0 (377 ohms). As an example, a 500 nm-thick aluminum film, with a resistivity of 2.82E-8 Ω*m [7], would reflect 99.94% of the beam.

A test was done to test the efficacy of the protective shielding. Both bare and protected paper tubing and fin material were exposed to the millimeter wave beam. The beam was still undergoing characterization at the time; however, the intensity was still significant and yielded good qualitative results. See Table 1 for the beam parameters.

Parameter	**Value**
Beam size	14 cm horizontal, 18 cm vertical
Peak power density	45 W/cm^2
Average power density over 20 cm diameter circle	26 W/cm^2,
Distance from mirror	24 m
Shot duration	0.67 s

Table 1. Beam parameters for radiation tests.

The bare tubing and fin material both showed considerable burn marks from the beam, while the foil protected material showed no signs of heating or damage. This built confidence that the thin aluminum foil and tape were sufficient for shielding. See Fig. 5 for pictures of the fin absorption test.



(a) (b)

Figure 5. (a) Bare fin material before radiation test. (b) Bare fin material after radiation test with observable burn mark in the center.

FLIGHT VEHICLE AND LAUNCH

The complete flight vehicle had a liftoff mass of 1.81 kg and was approximately 1.4 meters in length. It had two sets of green LED's for utilization by the pointing and tracking system of the beam. Twenty feet of 1 inch square aluminum extrusion from 80/20 Inc. was used as the launch rail, and the rocket had two plastic buttons to slide along the rail. The rail was supported with a mast system from BlueSky Mast Inc. A rod to grip the main ball valve on the rocket was attached to the stand and pneumatically actuated to turn 90 degrees, and weighted to fall away as the rocket moved upward. The actuator was based on similar design by McCormick et al. [8]. See Figs. 6 and 7 for photographic images of the rocket on the pad with the pneumatic actuator installed.

Figure 6. Photograph of rocket on pad.

The heat exchanger for propulsion is described in detail by Lambot et al. [4], and static thrust data is presented here as well. The thrust of the heat exchanger was measured with and without the beam to give quantitative data for the thrust increase. See Fig. 8 for the thrust data. The heat exchanger operating purely with cold gas had approximately 72 N peak thrust and a total impulse of 36.3 N-s. The total impulse is calculated by integrating the thrust with time curve,

$$Total\ Impulse = \int_{t1}^{t2} T dt \qquad 16)$$

where T is the thrust and $t1$ and $t2$ are start and finish times of the thrust. The heat exchanger, with the beam on, had approximately 88 N peak thrust and 41.6 N-s total impulse, representing a

22% increase in peak thrust and 15% increase in total impulse. The thrust curves were used to estimate the altitude of the rocket. The acceleration a was calculated by;

$$a = \frac{T - mg}{m} \qquad 17)$$

where T is the thrust, m is the rocket mass, and g is the gravitational constant. The acceleration was integrated with time to find velocity, which was integrated to calculate altitude. Air drag, rail friction, actuator imperfections, and the change in rocket mass were all ignored in the calculation. See Fig. 9 for a plot of predicted altitude versus time. The rocket with the beam on has a predicted altitude of 15.1 m versus 10.8 meters without the beam.

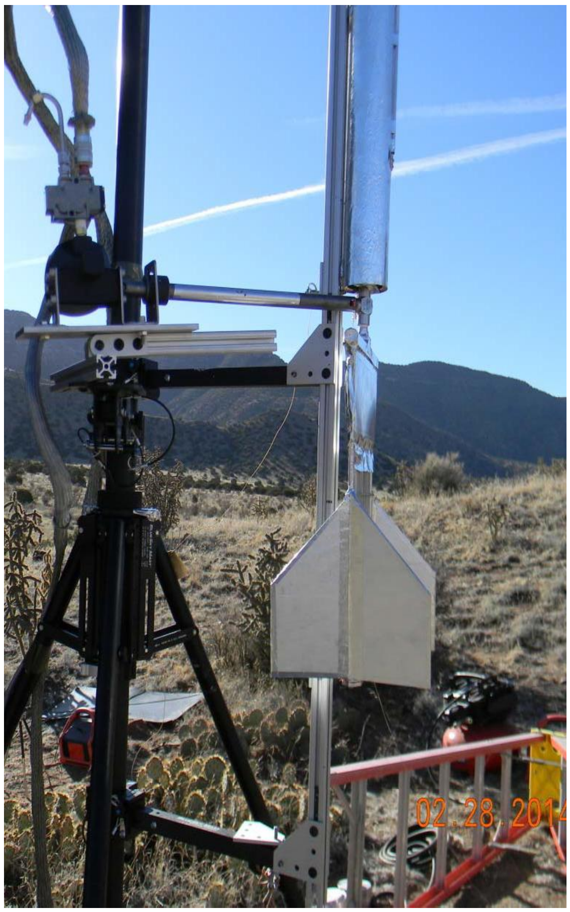


Figure 7. Photograph of rocket showing the actuator attached to the main ball valve.

The rocket flight of February 28, 2014 achieved an altitude of approximately 10 meters, determined by observations from the ground. The rocket configuration was the same as that used for the static thrust test data shown in Fig. 8. It successfully cleared the rail and was a free flying rocket albeit with a tether attached. The rocket impacted the top of the rail as it came down releasing the parachute, which became tangled on the launch rail's guy wires.

The beam tracking system did not have sufficient control authority to track the rocket. In lieu of accurate tracking, the heat exchanger was heated for 3 seconds prior to the start of gas

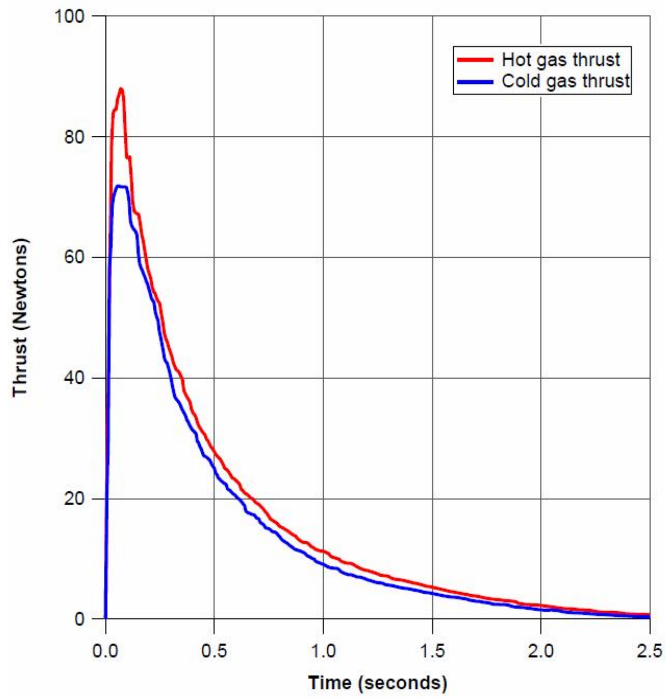


Figure 8. Static thrust vs time for the heat exchanger with and without the beam on. Load cell data taken February 26.

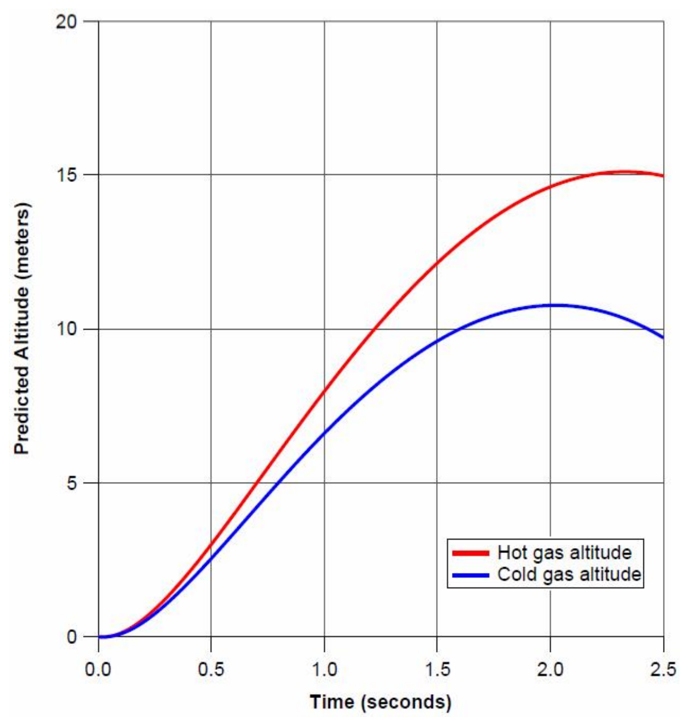


Figure 9. Predicted altitude versus time from thrust data. Rocket mass of 1.81 kg used and assumed constant. No effects for drag or friction were used in the calculation.

flow which still added a significant amount of heat to the fluid. Proper tracking with the beam would have added to the final altitude of the flight and is necessary for any beamed energy propulsion rocket system. Rail friction was also a likely cause of a loss in altitude since the plastic buttons that held the rocket to the launch rail had visible wear after the flight. The ball valve actuator also perturbed the rocket such that it swung side to side as it climbed up the rail, further adding to the friction.

IMPROVEMENTS AND CONCLUSION

The rockets were too heavy to fly in excess of 100 meters for the amount of beam power available, which is to say that too little power was absorbed by the rocket due to the compound effects of inefficiencies in the beam director and heat exchanger, all of which are correctable with further work. For the power actually absorbed, the entire rocket would need to be downscaled by at least a factor of two. For example, a rocket mass of 0.9 kg has a predicted altitude of 85 meters. The main propellant tank and high pressure components were all built to be safe and pressurized with people in close proximity. A remote system to fill the rockets would have allowed much lighter components. A burst-type valve would also have made the launch of the rocket smoother and reduced rail friction. A pointing and tracking system with better acceleration is also required since typically the rocket immediately flew out of the tracker field of view.

The rocket flights were a successful demonstrator of a beamed energy propulsion rocket with millimeter-wave power. Specifically the rockets successfully converted ground-based millimeter waves into heat within thin ceramic tubes. These tubes in turn heated a propellant, which measurably increased the thrust of the rocket, increasing its maximum altitude during a short range flight.

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