RRS NEWSLETTER

THE OFFICIAL JOURNAL OF THE REACTION RESEARCH SOCIETY INC.

Volume 51 - July 1994 - No. 3 **For the Advancement of Rocketry and Astronautics**

Published by the Reaction Research Society
P.O. Box 90306 Los Angeles, Ca 90009

"Well, if it doesn't work this time we'll just drink the fuel!"

by Brian Wherley and David Crisalli

Way back in 1990, Brian Wherley and I were standing by the blockhouse at the Mojave Test Area during a firing and hit upon the idea of building a really inexpensive, easy to assemble, high performance amateur experimental rocket. What could be easier, we postulated, than a hybrid. Just squirt a little LOX down the middle of a pipe thrust chamber full of a wooden fuel grain and plunk a graphite nozzle in the end. Any idiot could do it. Both PRS and RRS members had done it back in the 1950's using Presto logs (compressed wax and sawdust) for fuel. Then we went one better and decided that instead of a metal thrust chamber, we would make the fuel grain out of laminated particle board (good high density stuff) and wrap a composite motor case around the outside (no welding - no machining. We were on a roll now!) The injector was a simple brass pipe nipple with a cap on the end drilled with the injection orifices. This was going to be great.

It did not take us long and by March of 1991 we tested it. It was the last of four engines tested in a 24 hour period, and the test was conducted on an abysmally cold night at 11:45 PM. Dave Matthews, our "LOX man" for this series of tests, had just had a serious, gastrointestinal reaction to a can of something he had eaten at the Jawbone Canyon general store. (Not exactly what you would call a four star eating establishment). Let it suffice to say that he was incapacitated for a time behind the LOX truck before we could get him up and about again. But I digress. This whole story will be the subject of an article about things that should not be attempted by the sane (or even insane). To make a long story even longer, the engine did not work well. In fact, you might say it didn't work at all.

So after this really cleaver, laborious, scientifically meaningless, and less than impressive lox / particle board engine test, we learned a valuable lesson. Apparently, particle board will not burn even in a liquid oxygen environment. At least not the lox environment we provided. Armed with that tidbit of information, we set about the task of whipping up a real rocket engine burning real propellants that could be pushed through pipes with some modicum of control. We debated all the various propellant combinations and decided, for purely engineering reasons of course, that an engine burning "151 Rum" and liquid oxygen was the best possible choice.

You see, after the hybrid test, we had this large, unburned, composite engine that was not useful for anything, but that neither one of us had the heart to throw away. We had labored long and hard on this marvel of modern rocket science, and, even though it had disappointed us, we could not part with it. To this day it takes up valuable floor space in the crowded garage workshop. But, we thought, if we designed an engine burning 75.5% ethyl alcohol and water, we could use 151 Rum for the fuel. Then, if for any reason the engine did not work, we would know exactly what to do with the fuel and it would not require floor space. As it turned out, the rum was too expensive to burn so we saved it for other things and used denatured ethanol.

In keeping with our original intent with the hybrid, we were looking to design an extremely simple, very low cost liquid rocket that would use

Inside this issue

contined on next page

The Reaction Research Society, one of the two oldest Amateur Rocket Societies in the nation was organized in 1943 as a non profit civilian organization whose purpose is to aid in the development of reaction propulsion and its applications, and to promote interest in this science. The Society owns the Mojave Test Area, referred to as the MTA, a 40 acre site located two and one half hours north of Los Angeles. It is at this location were several hundred rockets, using both solid and liquid propellants, have been static tested and launched. Currently there are over 120 members.

This newsletter is a more or less bi-monthly publication by the RRS and is intended to provide communication between members, and other societies.

Information regarding the Society and Membership can be obtained by writing to:

Reaction Research Society Inc.
P.O. Box 90306 World Way Postal Center
Los Angeles, California 90009

as many commonly available components as possible. This would minimize construction time and the use of specialized equipment for fabricating parts. While this is not entirely possible to do because of the nature of liquid rockets, the eventual design of the propulsion system was about as simple as you can make it without sacrificing safety or operability.

The engine was the first major effort undertaken. The motor was designed to produce 620 pounds of thrust at a chamber pressure of 250 psia for a duration of about 10 seconds. Based on the work that Scott Claflin had done with his larger lox / alcohol engine, we decided on a flight weight ablative and graphite thrust chamber. The injector would be fabricated from 6061 - T6 aluminum using an injection pattern called a split triplet. A drawing of the engine is shown in figure 1 and the injector is shown in figure 2.

The thrust chamber was made from a 4 inch diameter mild steel tube with a 0.125 inch wall. A nozzle retaining ring was machined from 0.25 inch steel plate and welded into the lower end of the thrust chamber housing. A second steel ring was made and welded to the upper end of the thrust chamber. Drilled for an eight bolt pattern, this ring provided the attachment point for the injector. A stainless steel eighth inch AN fitting was also welded to the outer wall as a chamber pressure tap. This would provide a measurement

contined on page 4

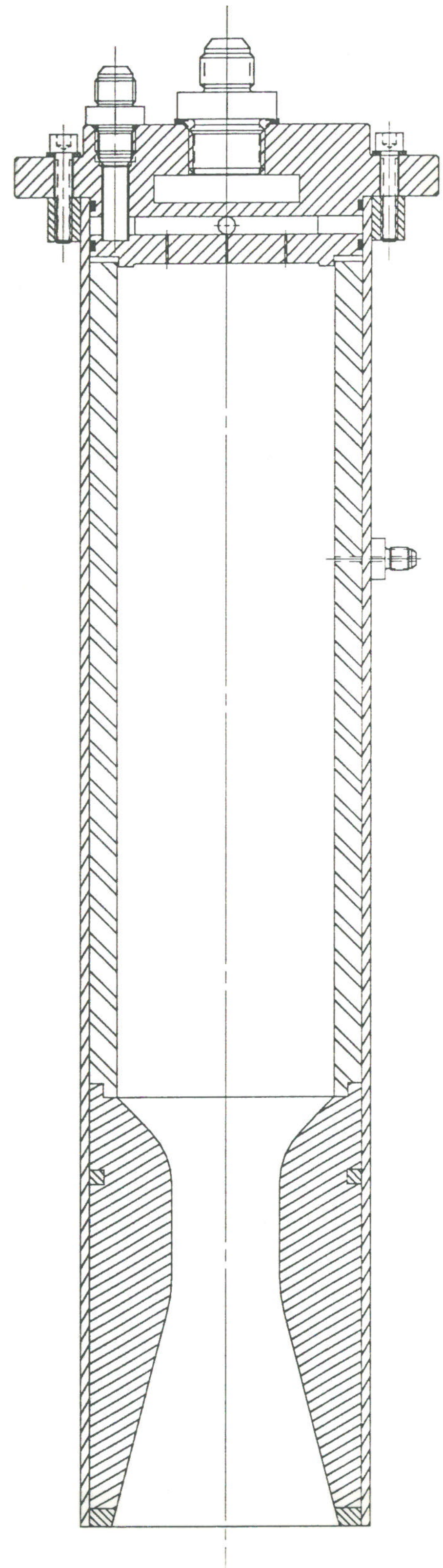


Figure 1. Engine assembly.

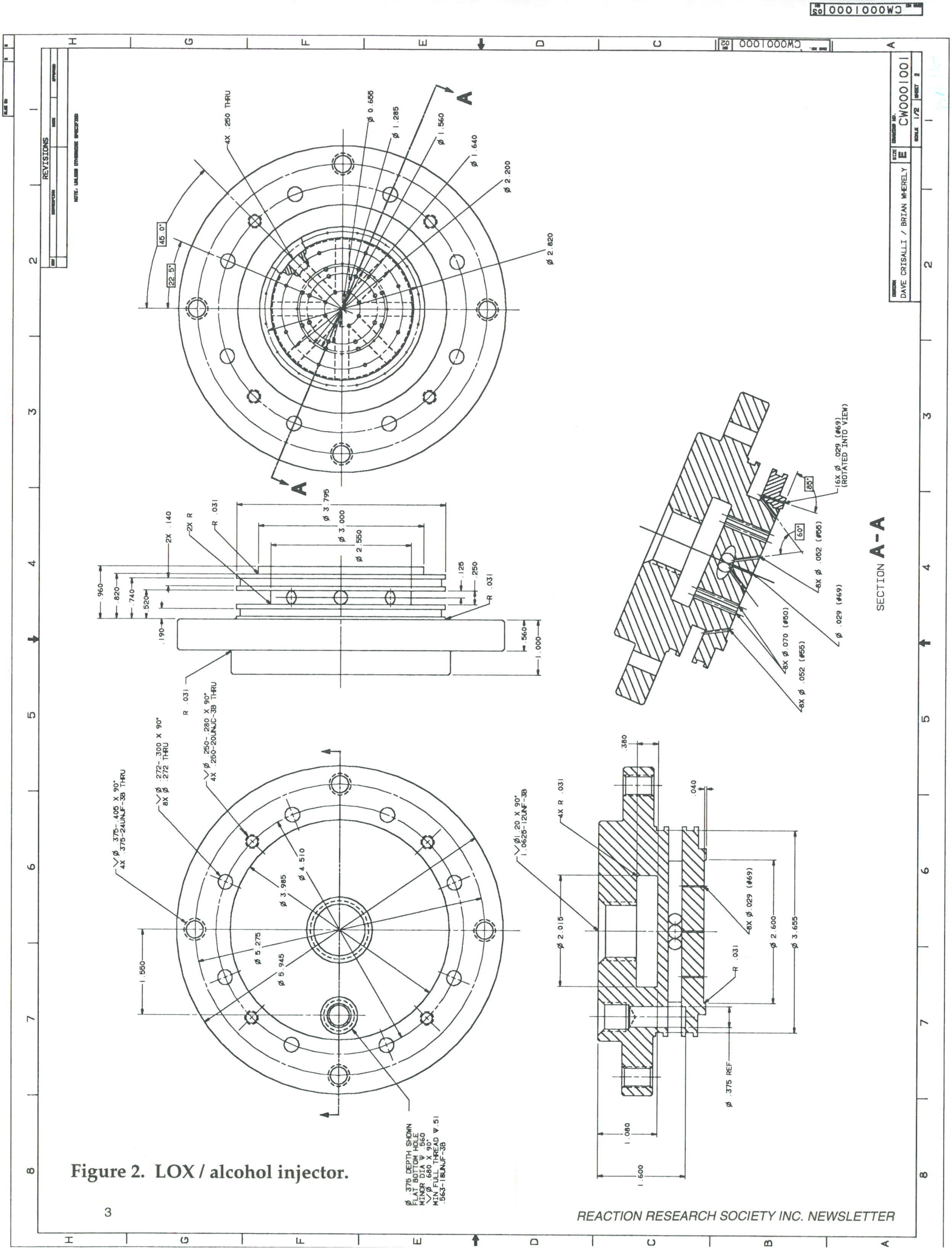


Figure 2. LOX / alcohol injector.

point during static testing.

Due to the demonstrated success of Scott's engine, we elected to use the same type of nozzle and a slightly modified ablative chamber liner. The nozzle was machined from the original graphite nozzle made for the first composite hybrid. The material was a low grade, coarse grained graphite bar available from many sources such as Graphite Machining in Los Angeles. This particular piece was found by luck and accident at Norton Sales. After machining, the nozzle was coated with a mixture of silica micro balloons (available from places like Aircraft Spruce in Fullerton, CA.) and a powdered urea - formaldehyde wood glue available at most hardware stores (Weldwood Plastic Resin Glue). This coating would serve to protect the graphite from oxidation for some part of the burn.

The ablative liner started with a 3.0 inch inside diameter cardboard tube with about a 0.125 inch wall. This tubing was reclaimed from discarded rolls of expended CAD drawing paper. The tube was cut to near final length and soaked with a polyurethane wood finish (Deft or Minwax). This provided some additional resinous filler to act as an ablative. We were not sure if this would help, but, what the heck, it couldn't hurt. The tube was then wrapped with a fine weave "S" glass cloth (also available at many places such as Aircraft Spruce) and the same urea wood glue used to coat the nozzle. This wrap was built up several layers thick until the outside diameter was just slightly smaller than the ID of the chamber housing. After the liner had cured completely (approximately 24 hours), a layer of Bondo (available at auto parts stores) was applied to the outside diameter. The liner was then turned on a lathe to bring the final diameter to a sliding fit in the chamber housing. This last layer of Bondo was added to prevent having to machine the fiberglass / urea material. The abrasive nature of the glass fibers destroys the edges of cutting tools very rapidly, while the Bondo machines easily. Although not as strong as the fiberglass portion of the liner, the Bondo layer is sufficiently strong to provide a good filler in the thermally benign area near the steel chamber wall.

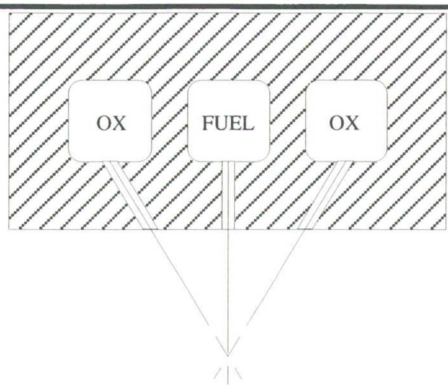
The injector was the next major component to be designed. Here again the intent was to design the simplest possible injector yet still have it deliver respectable performance. The first task in injector design is to select an impingement pattern. Triplet injectors are usually high performers and customarily have two angled oxidizer holes on either side of a single fuel orifice which emerges normal to the injector face. This division of the oxidizer into two streams impinging on a single stream of fuel keeps the hole diameters approximately equal and promotes better mixing. The drawback for home made rocket engines is that any minor mistakes in drilling the orifices will result in misimpingement and will cause oxidizer rich areas near the thrust chamber wall. This would cause a localized and accelerated degradation of the wall and a greatly reduced chamber life. For this reason it is desirable to make the oxidizer orifices normal to the face and angle the fuel holes. Unfortunately, this puts the oxidizer in the center and splits the fuel into two streams. This, in turn, makes the orifices very different in diameter and reduces performance.

One injector element type that improves on this situation is a split triplet, also known as a FOOF (fuel-ox-ox-fuel). In this element, the oxidizer flow is split into two streams emerging normal to the injector face. The two fuel streams are angled in from either side of the two oxidizer holes, and all four holes are in the same plane. Basically, this element is two back to back unlike doublets. Figure 3 shows a comparison between a conventional and a split triplet element. The injector was designed using eight of these elements. Additional showerhead fuel holes were later added between these elements to adjust the fuel side pressure drop and to kill any unmixed oxidizer recirculating between elements toward the chamber wall.

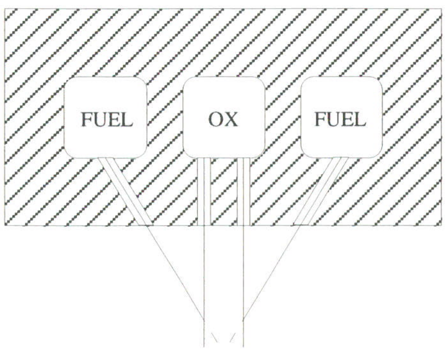
The injector was machined from a billet of 6061 T-6 aluminum bar. The oxidizer manifold was machined by using a specially ground tool. This design simplified the number of seals and fittings required in the liquid oxygen system. The injector took one evening to machine and one additional evening to drill. The total cost for the engine was less than $150. Photo 1 shows the engine components.

With the engine completed, the remainder of the propulsion system was designed and built. It was intended that the tankage and plumbing built for the static test would be in exactly the same configuration as it would be in the flight system with the exception of the propellant lines to the engine. These would need to be lengthened to accommodate the test stand thrust measurement system. Two D-2 aircraft breathing oxygen tanks were acquired for use as the fuel and oxidizer tanks. Each tank had a volume of 500 cubic inches and

continued next page



CONVENTIONAL TRIPLET - Holes are all approximately the same diameter (@ 2:1 MR) and oxidizer holes are angled toward a single, normal emerging fuel hole.



SPLIT TRIPLET - Oxidizer holes are slightly larger than fuel holes, but splitting the oxidizer stream helps. Oxidizer holes emerge normal to injector face, while the fuel holes are angled

Figure 3. Conventional and Split Triplet Comparison

they were rated for a service pressure of 450 psi. A lightweight 300 cubic inch aluminum pressure bottle was also found at a surplus yard and would hold sufficient helium at 2000 psi to feed propellants to the chamber. A ten foot long, one foot wide support skid was built from EMT tubing and unistrut. The pressure bottle, LOX tank, and alcohol tank were unistrut clamped to the skid in their approximate flight locations. Some decisions were then made on the plumbing and fluid control required.

The original system used an orifice to control the flow of pressurant gas into the propellant tanks in the same fashion as the system used on the 1950 hydrogen peroxide rocket. This eliminated the expense and weight of a regulator. An orifice was sized and machined. It was installed just downstream of the quarter inch Marotta solenoid valve used to start the flow of helium from the high pressure bottle to the propellant tanks. Fuel loading was to be accomplished through a small brass pet cock, and liquid oxygen would be loaded through a quarter inch LOX compatible check valve installed at the outlet fitting on the LOX tank. The main propellants would be controlled through linked, quarter inch commercial brass ball valves with teflon seats. The two valves were mounted back to back with their handles joined with screws. They were actuated to the open position by a heavy spring. The valve handles were pulled closed and tied to a block on the test stand with a loop of stainless steel safety wire. A pneumatic pin puller was used to release the tie wire and the spring would open the main valves. Simple, but only one way. This left no possibility of a shut down if one was required. A schematic of this original configuration is shown in figure 4.

Half inch copper line was used to plumb the oxidizer to the engine and 3/8 inch copper for the fuel. All pressurization plumbing was fabricated of 1/4 inch copper tubing downstream of the orifice. Quarter inch diameter stainless steel lines were used on the high pressure side above the orifice.

The original static test for this system was set for July of 1991 at the same time as Mark Grant's launch. The static test was scrubbed when the flight did not go as planned and no one wanted to lower the launch tower to clear the test stand. If we lowered it we would have to put it back up again. It had been no easy feat the first time to stand up a 35 foot tall steel tower with only four guys and one rope! We rescheduled for March of 1992. This test was cancelled when electrical and mechanical malfunctions of support equipment made continuing the effort impossible. Since Brian was sent away again on business for almost 10 months, the next opportunity to test came a year later in March of 1993.

In the intervening months, the propulsion system was modified and improved. A very small and light weight, aluminum, Circle Seal aircraft breathing oxygen regulator was found in a scrap yard and was substituted for the orifice. This provided much better control of tank pressure and, therefore, chamber pressure and thrust. This regulator had a built in relief valve, so the facility relief valve plumbed onto the skid was removed. The main propellant valves were also modified to a more controllable configuration. The spring and pneumatic pin puller were replaced by a bracket surrounding both propellant lines, and a small double acting pneumatic cylinder. This cylinder was plumbed to two three way Marotta valves drawing pressure from the fuel tank pressuriza-

FIGURE 4 ORIGINAL CONFIGURATION

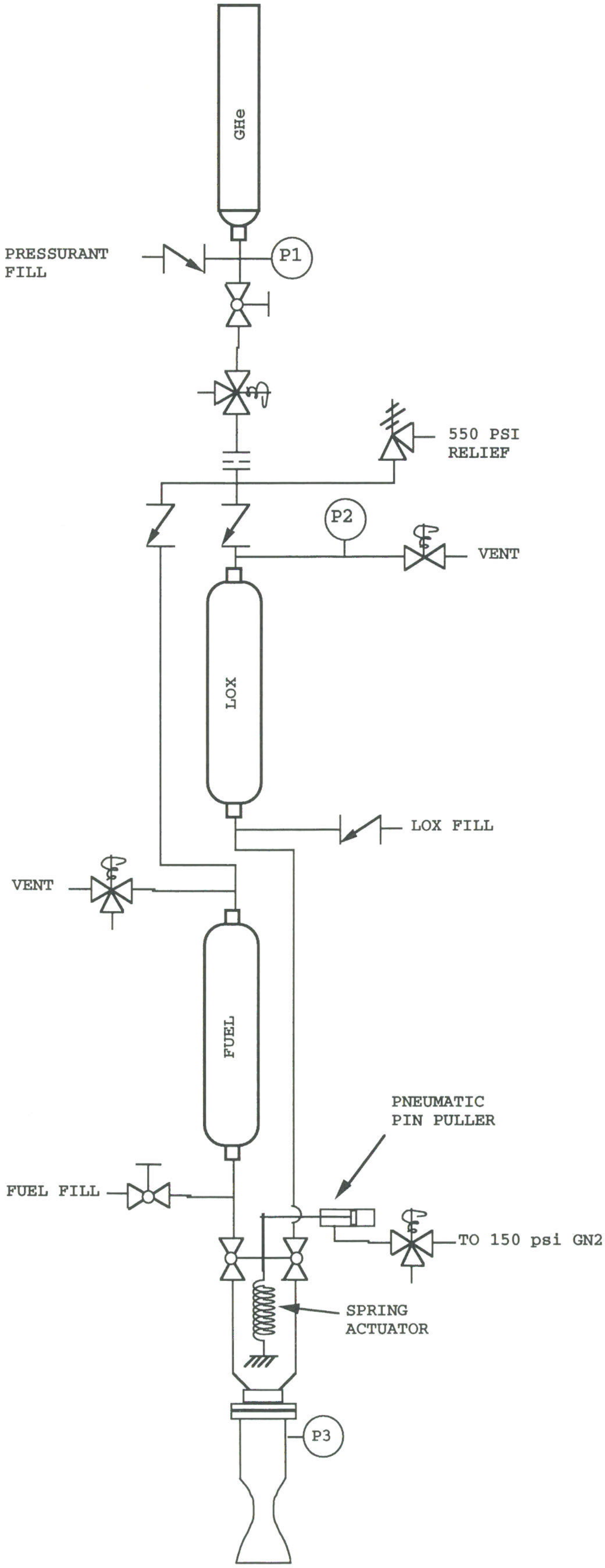
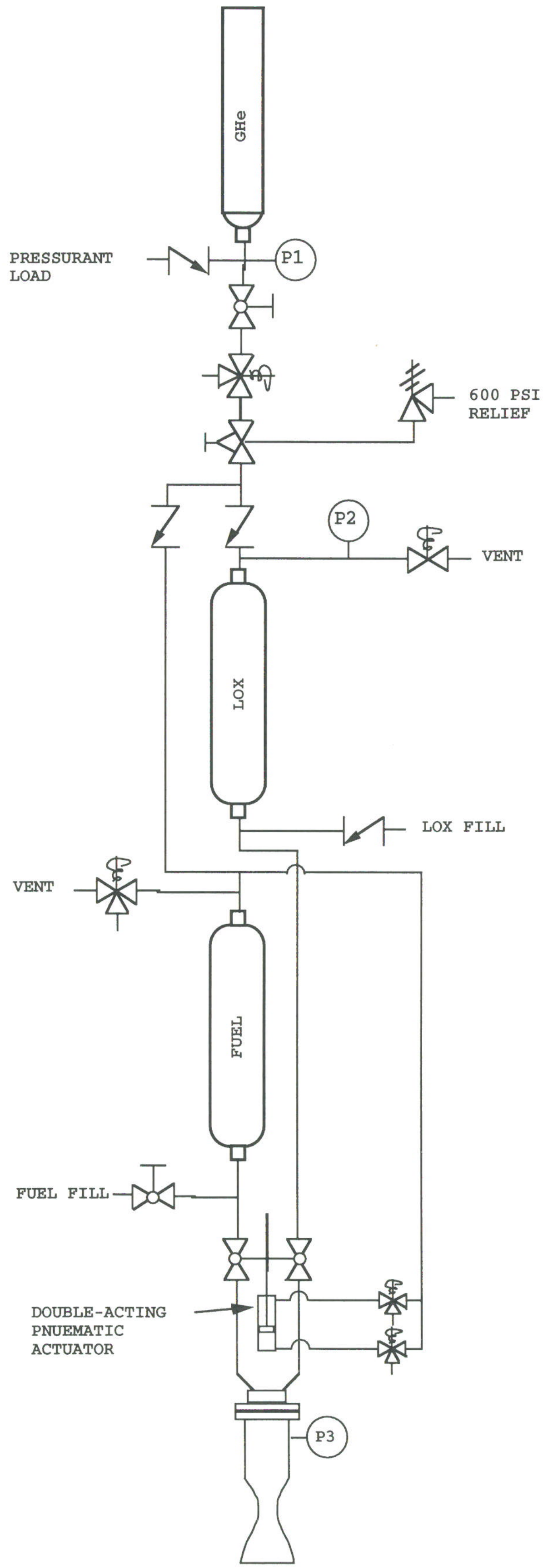


FIGURE 5. MODIFIED (TEST CONFIGURATION)

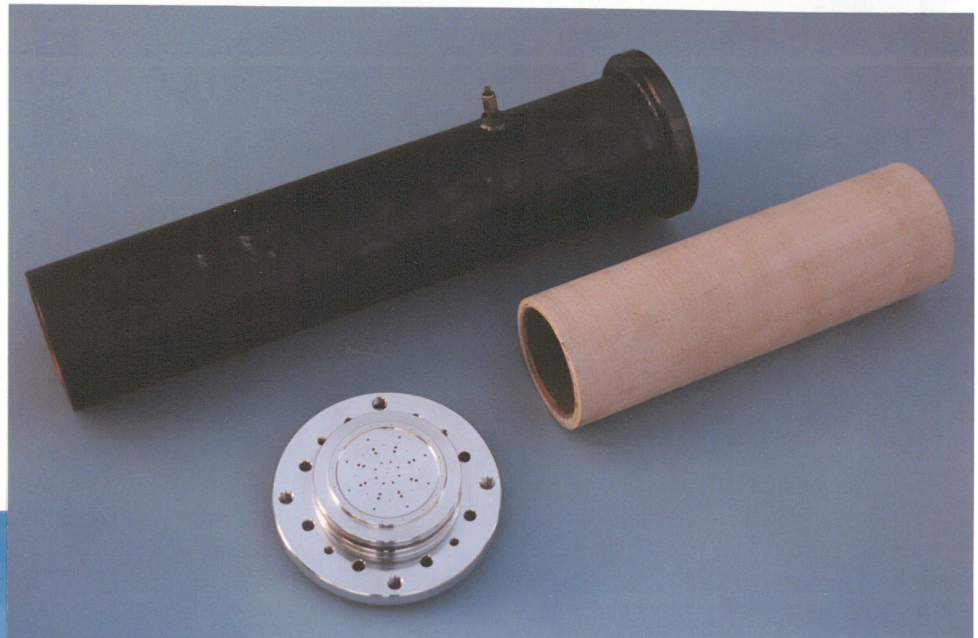


tion line. These valves would not be used in flight, but would allow the main propellant valves to be opened or closed at will during the static test. In the flight configuration, the two solenoid valves would be removed and the pressure line from the top of the fuel tank would go directly to the pneumatic cylinder. It would be plumbed to drive the main valves open as soon as the fuel tank was pressurized. A schematic of the final modified fluid system used in this test is shown in figure 5.

On 20 March 1993, the test was finally run. Just after Scott Claflin's successful 1600 pound thrust LOX / alcohol static firing, the test skid and engine were erected on the stand. LOX, fuel, and pressurant were loaded and the count initiated. The run went flawlessly producing very close to the expected 620 pounds of thrust for over six seconds. The engine delivered a c* (pronounced cee star) performance of nearly 90% and was in perfect post test condition. The ablative liner showed very little erosion and the nozzle was untouched. Data from the test is shown in figures 6 and 7. Photo 2 shows the system installed on the test stand and photo 3 shows the engine firing. In addition to this strip chart data, chamber pressure was also recorded by photographing a large, test stand mounted pressure gage and on a digital data collection system set up by Tom Mueller.

With the successful completion of this full propulsion system test, we were ready to start the design and build of the flight vehicle. But first we broke out a bottle of our "151 Rum" alternate rocket fuel and toasted our good fortune thus far. We also toasted our luck in saving valuable floor space in the garage machine shop and in having learned to avoid hybrid rocket engines. There was no really "neat" but useless hardware to store this time.

(A complete set of engine plans and fabrication instructions is available from the Society for a nominal charge. These plans can also be scaled up or down to build larger or smaller engines running on LOX / alcohol or other propellants.)



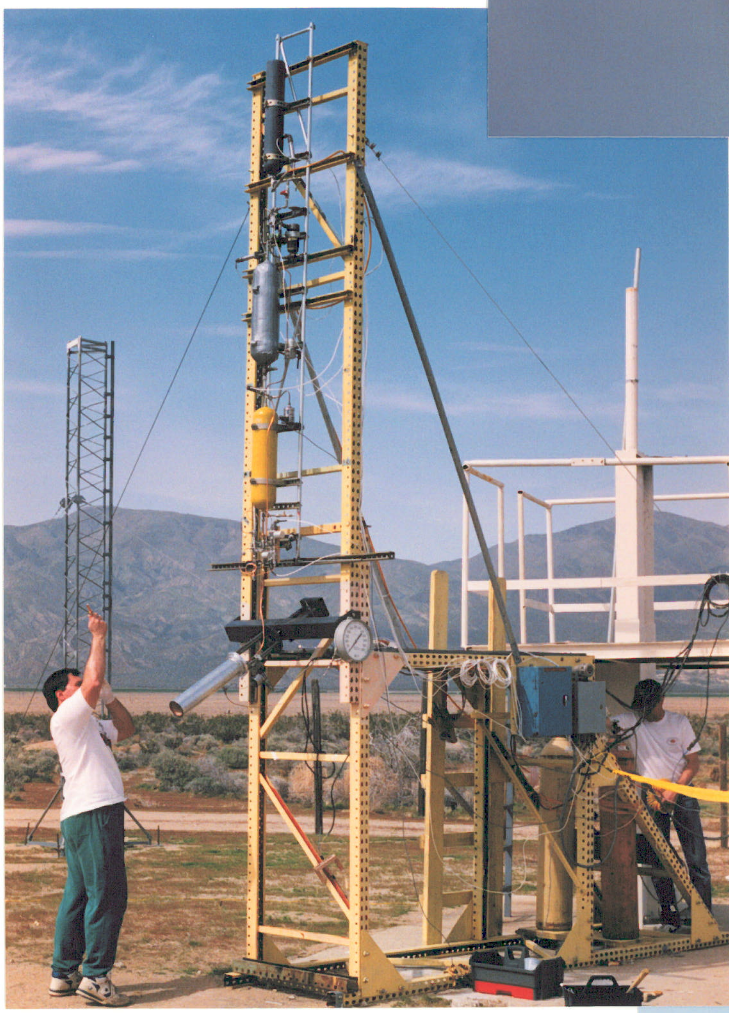


Photo 1. (Above) Thrust chamber housing, injector, and ablative liner ready for final assembly. The graphite nozzle has already been installed in the chamber housing.

Photo 2. (Left) Final preparations for static test are being completed. The flight pressure bottle (top), LOX tank (middle), and alcohol tank (bottom) can be seen installed on the test skid. Brian Wherley (left) is plumbing the liquid oxygen line to the engine.

Photo 3. (Right) The engine undergoing test on 20 March, 1993. The engine produced over 600 pounds of thrust for over 6 seconds and was in excellent post test condition.

Figure 6. Static test strip chart data.

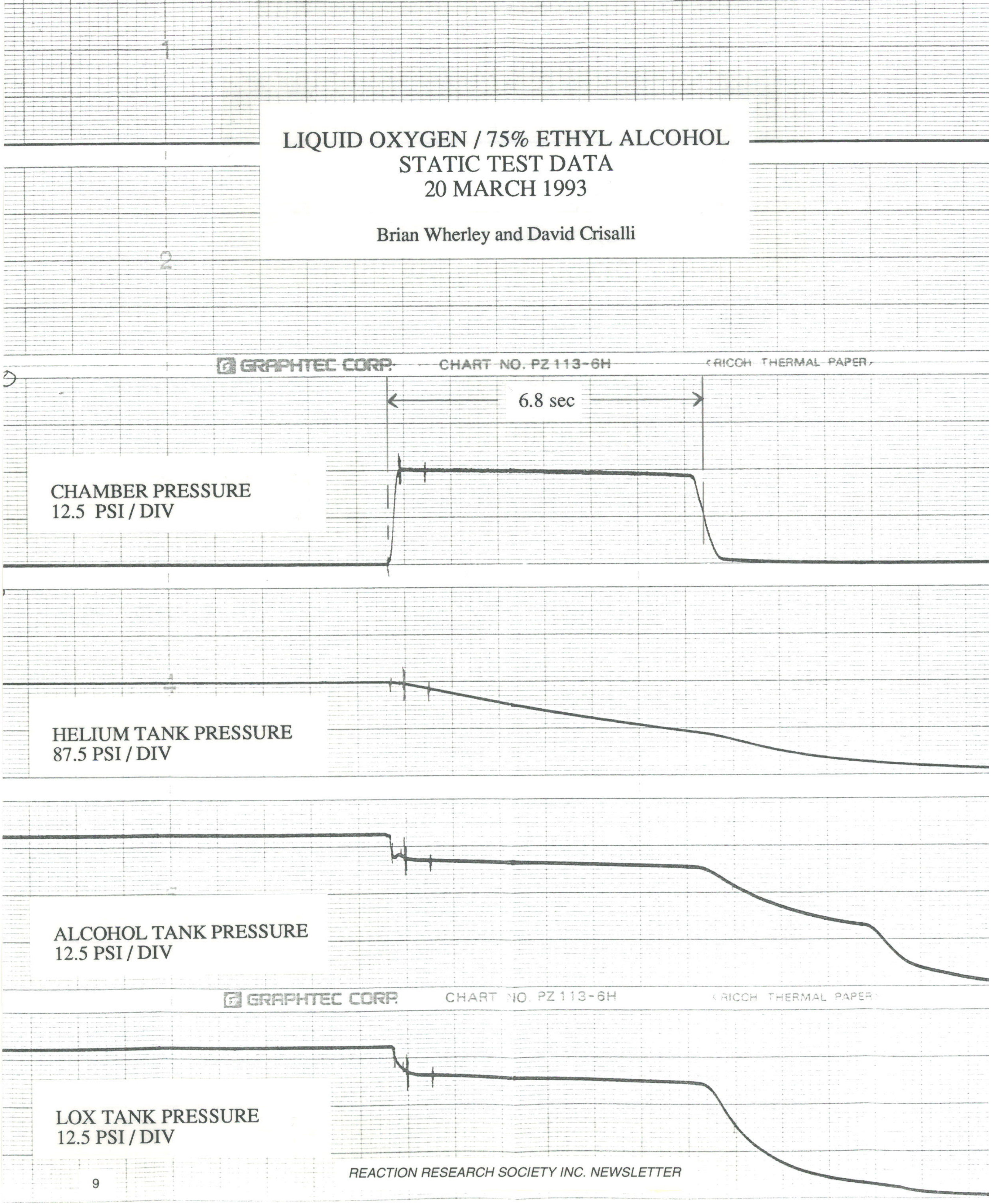
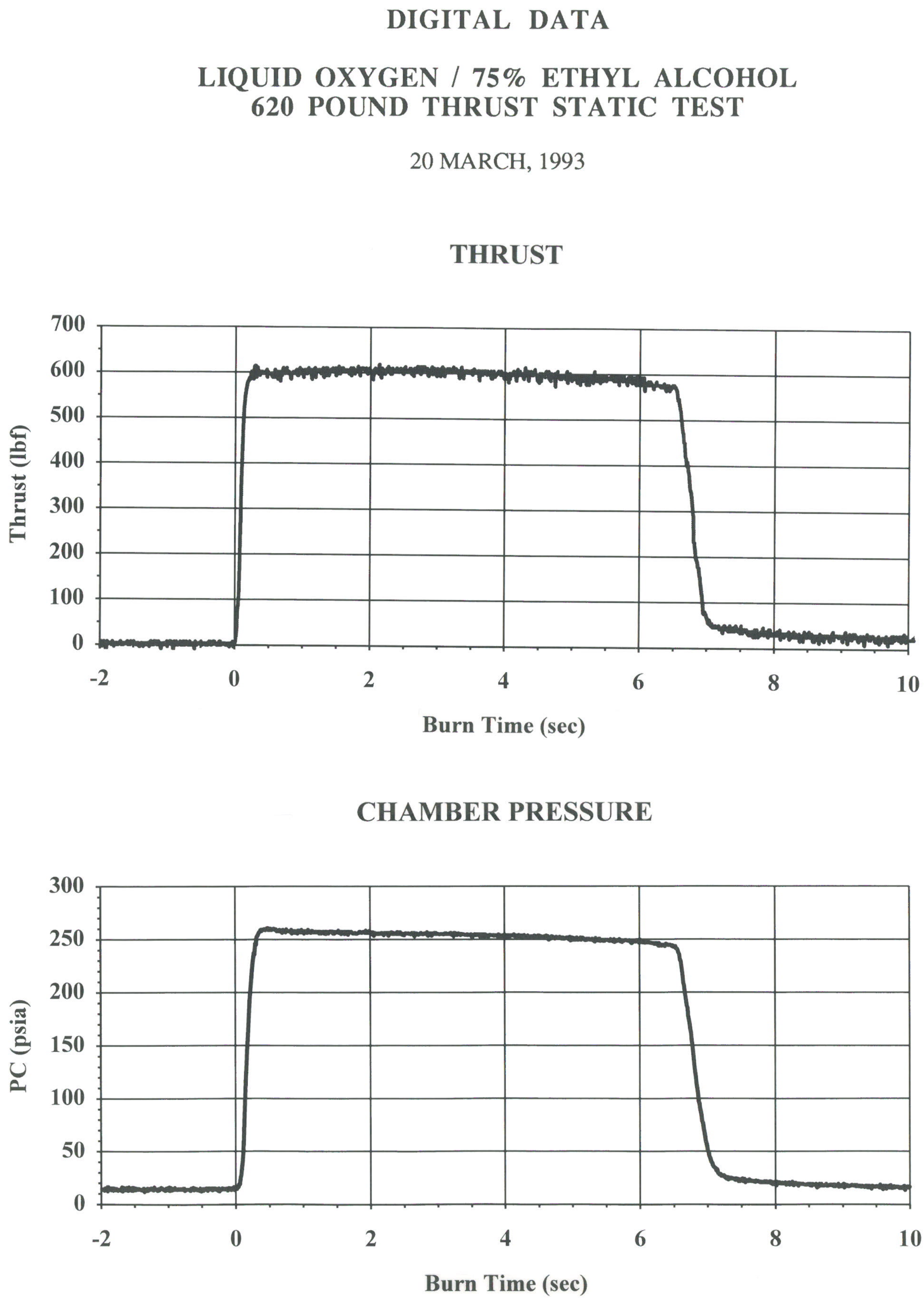


Figure 7. Static test digital thrust data.



Public Relations Note

by David Crisalli, June 1994

In my new capacity as the public relations coordinator for the RRS, I would like to pass on a little information about what is going on and to ask the membership for help in a few areas. First, I would like to let everyone know that I am trying to get articles about past and present RRS activities published in magazines such as High Power Rocketry, Final Frontier, Smithsonian Air and Space, etc.. The intent here is to gain some notoriety for the Society and to show off the type and variety of the work undertaken in the field of amateur experimental rocketry. The first two of these articles have already been published in the February and June issues of High Power Rocketry. The first covers the history of the recent liquid propellant rocket work conducted by RRS members, and the second is about large zinc/sulfur rockets. The fourteen page article on liquid rockets came out very well and the many photographs included of activity at the MTA were printed in full color. If anyone wants a copy of either of these two issues and cannot find one, they can be ordered directly from the publisher at the following address for $5 each plus $2 shipping ($0.50 shipping for each additional copy);

Mr. Bruce Kelly
High Power Rocketry
P.O. Box 96
Orem, UT 84059-0096
(801) 225-3250

A greatly expanded version of the liquid rocket information is also in work to be published as a consolidated report and sent out to all RRS members. This document will be somewhere between 80 and 100 pages with dozens of photographs and drawings. It will contain detailed information on several liquid projects in 16 appendices.

To do this report well (including some color xerox and the reproduction of all the other black and white photographs), it will run from $10 to $15 per copy. With production and mailing of over 140 copies, this will add up to a fairly healthy amount. Many of the contributors to this document are donating $50 each to defray production costs. Due to the costs involved and the limited resources, I would ask the membership for help in two areas. First, if any member is NOT interested in receiving a copy, please contact me by mail (David Crisalli, 3439 Hamlin Ave., Simi Valley, CA 93063). At $15 a piece, we do not want to produce or mail copies that are of no interest to individual members. Second, for any of those magnanimous and philanthropic members out there, any monetary contributions to the publication fund for this report would be greatly appreciated. Ten dollars would cover the cost of your copy and any additional may allow us to include more color photographs, etc.. Please contact me if you are interested in giving your money away. In this same vein, if any of the membership has any connections to anyone capable of doing color xerox (or any sort of printing) who might be persuaded to help us put out a better product, I would very much appreciate your help. This report on the liquid propellant activity will hopefully be followed by a similar one on the advanced composite propellant work now going on.

Members of the RRS have been doing impressive and sophisticated work for many years. They have, however, been woefully inadequate in documenting this work for other members. This effort to produce a combined report will capture a tremendous amount of information for current and future members, and add to the prestige of the Society as a whole.

As part of this publication effort, I would also like to ask for help in the area of photographic documentation. Thousands of pictures are taken at the events at the MTA, but everyone takes their photos home and puts them in a drawer. If everyone attending a firing would have double prints made and send a set to me, I will keep a file. If a particular file photo is needed for some publication or report, I will know who to contact about getting more prints. Of course, this system will only work if you keep your rocket negatives all together and can find them if a print is needed. If you don't want your negatives, just send them along and I will keep them in the file as well.

Along the lines of my request for anyone who has information on printers who might be willing to donate their services (or at least give us a break on things), I would also like to hear about any of the following type of people you may know;

- Building supply vendors or manufacturers (block, lumber, steel, concrete, rebar, rail road ties, etc.)
- Fencing supply vendors or manufacturers (chain link fencing supplies)
- Contractors (who might be willing to supply construction support on the odd weekend or two)
- Equipment rental types (who might be willing to give us free use or reduced rates on cement mixers, welders, airless paint sprayers, etc.)

If I can compile a list of these people with the help of the membership, I will write them a letter explaining who we are, what we do, and the type of help we would like to ask them for. As an example, the retail price for cement block is 80 cents each. Retailers buy them for less than 40 cents each, and manufacturers make them for about 15 cents each. If we can get a manufacturer to donate 1000 block it would only cost him $150. Even if we can arrange to buy them from him at cost, we can get a lot farther buying 1000 block for $150 instead of the $800 it would run at retail. Thanks in advance for any help you can provide.

LIQUID ROCKET FLOW CONTROL WITH THE CAVITATING VENTURI

By Tom Mueller

The cavitating venturi is a very useful device for controlling and measuring the flow of propellants in a liquid rocket engine. They can also be useful for inhibiting feed system oscillations by decoupling the engine from the feed system upstream of the venturi. This article, presented in two parts, will help describe the design and application of a cavitating venturi for rocket use. This installment will describe the operation of a cavitating venturi and how to design one for desired rocket operating conditions. In the next newsletter, I will describe how to make and calibrate a cavitating venturi.

In order to determine the specific impulse of a liquid rocket, two pieces of data are needed. First is the thrust of the engine, and second is the total propellant flow rate to the engine. The I_{SP} is simply the thrust divided by the total flow rate:

$$I_{SP} = \frac{F}{\dot{W}}$$

Where:

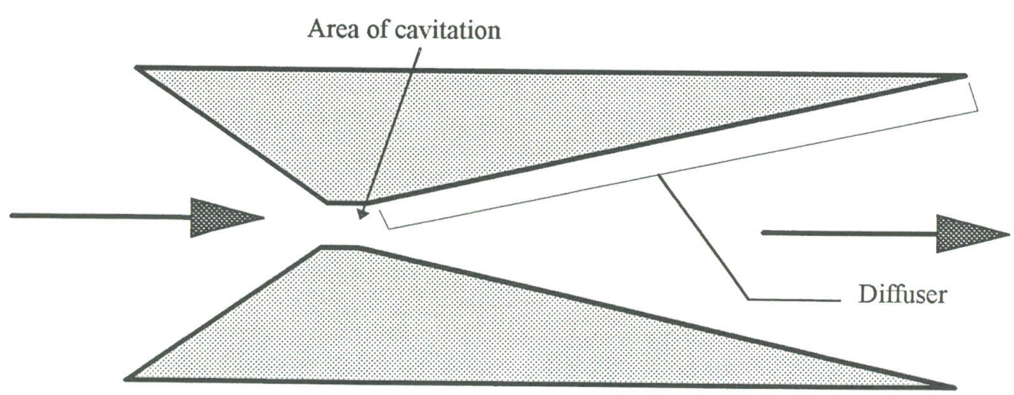
F = Engine thrust (lbf)

$\dot{W}$ = *Total Flow (lbm/sec)*

The thrust is usually fairly straight forward to determine by using a load cell. The flow rate is more difficult to measure directly, however, because it typically requires the use of turbine type flowmeters and specialized instrumentation to convert the flowmeter signal to volumetric flowrate. If the propellant is a cryogenic fluid like liquid oxygen, the equipment becomes even more exotic.

A simple way around this is by use of a cavitating venturi. Figure 1 shows a diagram of a cavitating venturi. When installed in the propellant line between the tank and the engine, the cavitating venturi will give an accurate, steady propellant flow rate based on the tank pressure and the propellant temperature. As its name implies, a cavitating venturi is a venturi shaped flow passages that operates at flow rates high enough to ensure that the propellant is cavitating at the throat, which is the point of smallest flow area. The throat is where the fluid velocity is highest, and therefore, by Bernoulli's principal, the pressure is the lowest. If the velocity is high enough, the pressure at the throat will be below the vapor pressure of the fluid, causing vapor

Figure 1. Cavitating Venturi Schematic



bubbles to form, a phenomenon also known as cavitation.

The reason cavitation is desired is because the vapor pressure of the fluid now becomes the downstream pressure that the cavitating venturi "sees". When a venturi is not cavitating, it behaves like any orifice, and the flow rate is proportional to the square root of the pressure drop across the venturi, as shown:

$$\dot{W} = CdA\sqrt{2\rho g\Delta p}$$

where:

$\dot{W}$ = flow rate

Cd = venturi discharge coefficient

A = venturi throat area

ρ = density of propellant

g = gravitational constant

ΔP = pressure drop

The pressure drop when the venturi is not cavitating is simply the inlet pressure minus the outlet pressure. In this case the outlet pressure is just the pressure downstream of the venturi:

$$\Delta P = P_{in} - P_{out}$$

When the venturi is cavitating, the outlet pressure is the vapor pressure of the liquid, or $P_{v.}$ The flow rate through the venturi is now defined as :

$$\dot{W} = CdA\sqrt{2\rho g\left(P_{in} - P_{v}\right)} \tag{1}$$

If the temperature and pressure of the propellant is known at the inlet to the venturi (tank conditions), then the density and vapor pressure of the fluid can also be determined. Because all of these parameters are constant during the operation of the engine, the flow rate to the engine is fixed. The CdA of the venturi can be measured by water flow calibrations, so all of the information that is needed to calculate the propellant flow rate is known.

So how do you ensure that the venturi remains "in cavitation"? Simple, if the outlet pressure is below the recovery pressure of the venturi, then the venturi is cavitating. The recovery of a venturi is the amount of pressure recovered by converting the liquid velocity to pressure in the shallow angle diffuser just downstream of the throat. A typical recovery factor for a well made venturi is 80 to 85%. This means that the outlet pressure must be <u>less</u> than 80 to 85% of the inlet pressure for the venturi to be in cavitation. For example, if the inlet pressure (tank pressure) for a venturi is 1000 psia and the venturi has a recovery of 80%, then the inlet pressure to the engine must be less than 800 psia to ensure that the venturi is in cavitation. If the inlet pressure to the engine is only 400 psia, that's fine, the venturi will still provide the same flow rate as it would if the inlet pressure was 800 psia. That's the beauty of a cavitating venturi, the flow rate is constant no matter how the downstream pressure varies, as long as it does not rise above the recovery pressure of the cavitating venturi. If the engine is running rough such that the inlet pressure fluctuates between 400 psia and 500 psia, the flow to the engine will remain constant since the venturi is always in cavitation. If a cavitating venturi is not used, the flow rate may couple with the inlet oscillations, causing a reinforcement of the roughness. This is a flow condition known as chugging, which a cavitating venturi can help prevent.

EXAMPLE: A liquid rocket engine operating on LOX and kerosene has the following design parameters:

$$\dot{W}_{LOX} = 2.0 \text{ lbm / sec}, \qquad \dot{W}_{fuel} = 1.0 \text{ lbm / sec}$$

$$\text{PC} = 300 \text{ psia}, \qquad \Delta P_{injector} = 100 \text{ psi}$$

Design cavitating venturis to provide the required flow rates to the engine.

SOLUTION: The inlet pressure to the engine is PC plus the injector ΔP, or 400 psia. Assuming that a recovery of 75% can be achieved, the minimum tank pressure is :

$$P_{in} = \frac{400}{0.75} = 533 \text{ psia}$$

In order to provide some margin on cavitation, a tank pressure (venturi inlet pressure) of 600 psia will be used.

LOX Venturi: Assume the LOX tank will be vented just prior to the engine firing so the propellant is at its normal boiling point. At this condition the vapor pressure is equal to the atmospheric pressure of about 14 psia (for the MTA). The density of LOX at the normal boiling point is 71 lbm/ft^3 or 0.0412 lbm/in^3. The gravitational constant g = 32.174 ft/sec^2 or 386.1 inches/sec^2. Rearranging equation (1) to solve for the flow area:

$$CdA = \frac{\dot{W}_{LOX}}{\sqrt{2\rho g(P_{in} - P_v)}} = \frac{2.0}{\sqrt{2(0.0412)(386.1)(600-14)}} = 0.01465 \text{ in}^2$$

The discharge coefficient, Cd, for a typical venturi is about 0.90 (more on this in the next newsletter). Therefore the throat area of the venturi is:

$$A = \frac{CdA}{Cd} = \frac{0.01465}{0.90} = 0.01628 \text{ in}^2$$

Solving for the throat diameter:

$$d = \sqrt{\frac{4A}{\pi}} = \sqrt{\frac{4(0.01628)}{\pi}} = 0.144 \text{ inch}$$

i.e., a 0.144 throat diameter would be required for the LOX venturi.

Fuel Venturi: Assume the kerosene is a 70°F prior to the engine firing for which the density is 51 lbm/ft^3 or 0.0294 lbm/in^3. The vapor pressure of kerosene is negligible, so use $P_v = 0$. Rearranging equation (1) to solve for the flow area:

$$CdA = \frac{\dot{W}_{fuel}}{\sqrt{2\rho g(P_{in} - P_v)}} = \frac{1.0}{\sqrt{2(0.0294)(386.1)(600-0)}} = 0.0857 \text{in}^2$$

Using a Cd of 0.90 the throat area of the venturi is:

$$A = \frac{CdA}{Cd} = \frac{0.00857}{0.90} = 0.00952 \text{ in}^2$$

Solving for the throat diameter:

$$d = \sqrt{\frac{4A}{\pi}} = \sqrt{\frac{4(0.00952)}{\pi}} = 0.110 \text{ inch}$$

i.e., a 0.110 throat diameter would be required for the fuel venturi.

PYRO OP

POST FIRING TEST REPORT

LIQUID PROPELLANT STATIC TEST

30 April 1994

by David Crisalli

On 29 April 1994 another entourage of rocket scientists and junk haulers extrordinare arrived at the MTA late in the afternoon. With them they had brought the usual ten tons of equipment required to test liquid rockets. They were greeted, cordially, by Tom Mueller (who had foolishly believed them when they had told him they would be there by 10:00 AM and who had been sitting for hours all alone in the desert contemplating God knows what). But now, with all that behind us, we all hit the decks running to get everything set up before the sun went down and we had to do all this again by braille like we always do. There was a lot to do in preparation for the planned testing of five bipropellant and hybrid rocket systems on Saturday.

Jim McKinnon had built a new injector for his 1000 pound thrust LOX / Jet A engine to replace the one that had been destroyed by a hard start the year before. His would be the first test on Saturday, so his test apparatus was immediately erected on the test stand. Scott Claflin was ready for the third (I can't believe we did this again) test of his 1600 pound thrust LOX / alcohol system. This time the engine had a new split triplet injector replacing the previous unlike doublet. Scott swore that this was the last static test. The next time we see this one work it will be in the air.

Mark Ventura was on his way to the MTA to test an almost all plastic 85% hydrogen peroxide / polyethylene hybrid rocket producing 50 pounds of thrust for 5 seconds. Both the combustion chamber and peroxide tank were built of PVC pipe. The only metal on the test vehicle was a pyro valve built for Mark by Tom Mueller and a few of the fittings used to load propellant, pressurize the tank, and conduct the peroxide into the motor.

Korey Kline was bringing out two nitrous oxide / HTPB hybrid engines, a test stand, and several tanks of N2O. The engines were designed to produced 250 and 750 pounds of thrust for 60 and 30 seconds respectively.

Through some phenomenal luck, and a modicum of skill developed as a result of doing this so many times recently, we had all the equipment unloaded, set up, checked out, and secured for the evening by 10:30 PM. (OK, so we had to do some of it by braille). We even had the fuel loaded in Jim's rig before we retired for the night. By 7:00 AM we were all back on deck and ready for final set up activities, like coffee and doughnuts.

Photo 1. Jim McKinnon's engine ignites and runs normally for several seconds. A hot gas leak developes at the chamber pressure tap off line, and internally, the chamber liner burns through in some places increasing the fuel flow.

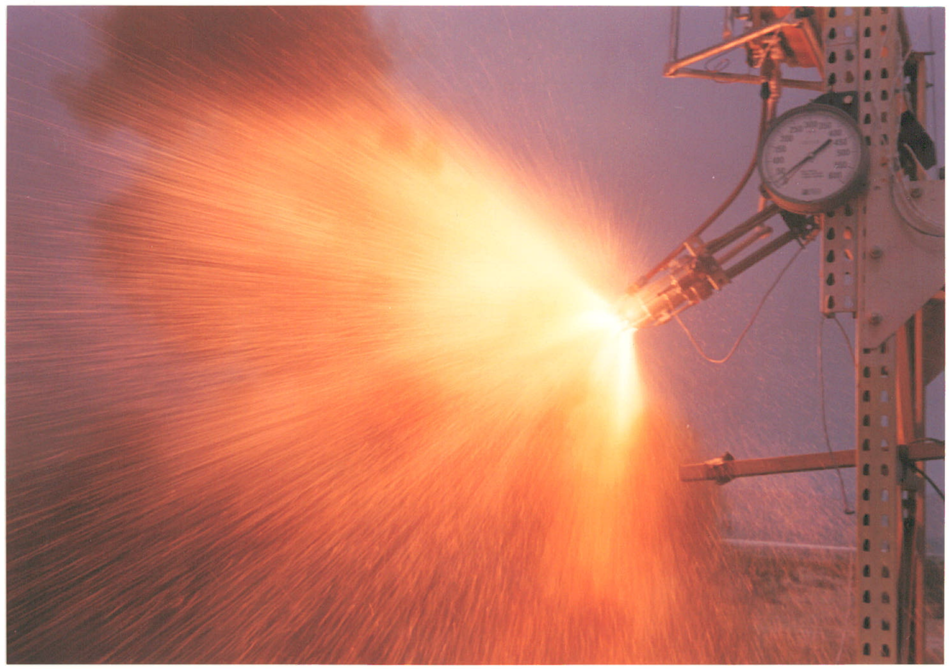


Photo 2. As the fuel tank runs dry, oxidizer impingement with the hot chamber wall produces this dramatic melt down.

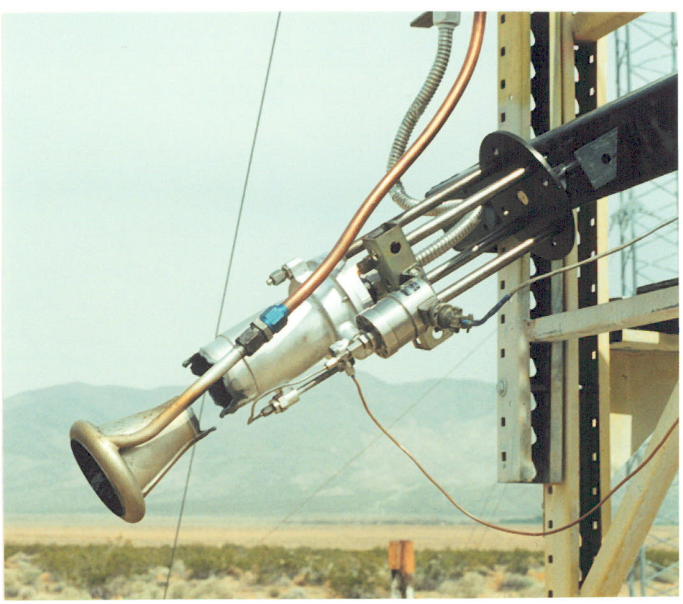


Photo 3. The chamber exit nozzle is held in place by only the fuel inlet line. The chamber is destroyed, but the remainder of the hardware is undamaged.

Photo 4. (Left) Scott Claflin's 1600 pound thrust lox/alcohol engine running for 7 seconds. The black square burning in the lower left is the particle board flame deflector used to reduce the dust kicked up. It did not work well.

Photo 5. (Below) Korey Kline's 700 pound thrust nitrous oxide/HTPB hybrid ran for 21 seconds on a portable vertical test stand.

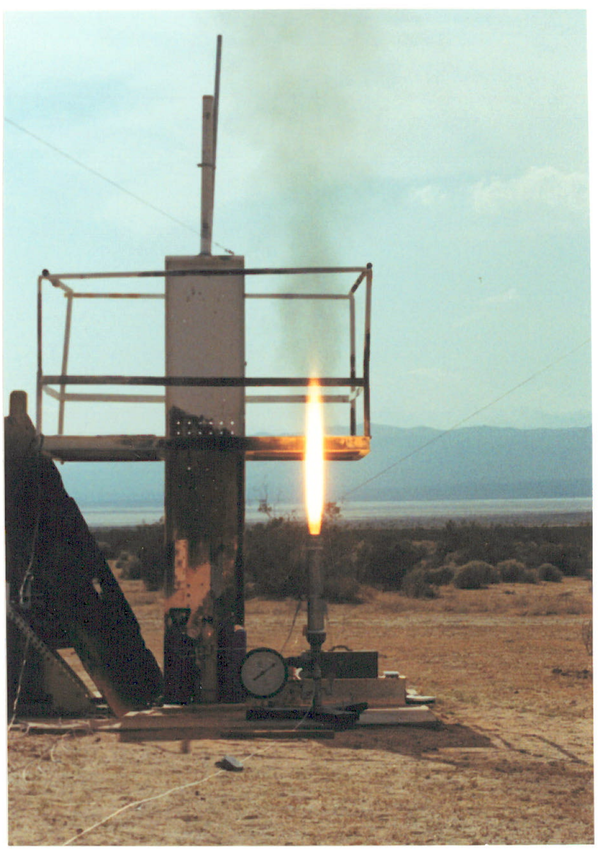


Photo 6. (Above) Mark Ventura's 50 pound thrust, 85% hydrogen peroxide/polyethylene hybrid undergoing test. All the major components were fabricated from PVC pipe.

After the usual briefings, sign ups, and correcting some minor technical problems, we loaded LOX by mid morning and ran Jim's test. The start was magnificent and the engine appeared to be running well for 12 seconds. However, just as the tanks ran out of propellant, there was a huge deluge of sparks spewing from the regeneratively cooled thrust chamber before the flame went out. Photos 1,2, and 3 show the engine at various points during the test. As can be seen in photo 3, the entire mid section of the thrust chamber had burned out. Review of the photos and data from this test indicate that the chamber wall may have burned through in several areas during the test due to oxidizer impingement with the wall. The increased fuel flow caused the fuel tank to run empty first. The resulting oxidizer impingement with the hot, now uncooled (since there was no fuel flow in the jacket) chamber wall led to the dramatic melt down. While the chamber is a complete loss, none of the other hardware was damaged and can be used again. Jim is planning to build a new ablative chamber, make some minor modifications to the injector, and test again.

After the usual rat race to lower one test article and install another, Scott's system was set up in almost record time (about 2 1/2 hours). Propellant loading went quickly and the engine was fired at about 2:00 P.M.. The test was excellent (as usual) with the engine producing 1626 pounds of thrust for 7 seconds. Photo 4 shows the engine firing.

Mark Ventura was the next up to bat, but three consecutive failures of the pyro valve to function properly and release the peroxide into the engine resulted in moving his test to the end of the line. Korey Kline stepped in to set up for the first of his tests while Tom and Mark were off refurbishing the pyro valve in the quonset hut. Mounted on a vertical test stand that would record the maximum thrust attained, the first of Korey's engines produced a peak thrust of 700 pounds and burned for 21 seconds. Shortly after this first successful test, the second engine was on the stand and fired. It produced a peak thrust of 200 pounds and burned for 49 seconds. Again the test was successful except for some asymetrical nozzle erosion near the end of the burn that vectored the thrust about 10 degrees off the engine centerline. Nonetheless, Korey was two for two with these tests, and that is a lot better than anyone else has done with hybrids lately. I guess you just have to know what you're doing. Photo 5 shows Korey's 700 lb thrust engine firing. His next engine, now under construction, is designed to produce 1200 pounds of thrust for 20 seconds. After static testing, this engine will be reconfigured for flight test.

After Korey's last firing, Mark Ventura was ready to try again. This time the pyro valve fired and the little engine roared (and I do mean roared) to life. It was amazing to see all that fire come out of a piece of PVC pipe in an orderly fashion. The PVC peroxide tank was equally impressive having been pressurized to 700 psig to feed oxidizer into the engine. Photo 6 shows Mark's engine firing and producing about 50 pounds of thrust for 5 seconds.

After this last test, we all began the even more laborious task of repacking all the gear to go home. It was all down and packed by 6:00 P.M. and we were all having pizza in California City by 7:00. It had been a tiring, but dramatic and successful Saturday.

STANDARD RECORD FORMS

For many years, members of the RRS have been building and testing a myriad of propulsion devices from the extremely complicated to the very simple. Many of these projects are remarkable in what has been successfully accomplished with very basic tools and designs. Equally as important as undertaking these projects is documenting them for the benefit of other members of the society. As I mentioned in the public relations note, also included in this issue, the impressive and sophisticated work being done should receive more notoriety both within and external to the RRS. As an aid to those undertaking projects and to standardize the information collected, the Standard Record Form that has been in use for many years has been updated and a copy has been included in this issue. This form is specifically for solid propellant rockets and can be used for either static or flight testing. A similar form for liquid rocket work is being generated now and will be included in the next issue.

Anyone intending to fire a solid rocket in the future will need to fill out and submit a copy of this form to get your vehicle on the docket. A copy filled out with all required pre firing data should be submitted to the pyro op scheduled to conduct the event and a second copy should be sent to the director of research. On the day of the firing, the pyro op's copy can be completed and then filed with the director of research. The submission of this information can be very helpful in several ways. First, it helps to document RRS activities. Secondly, it is a great aid to the pyro op in charge to know what type of device is to be tested and how the testing should be scheduled to maximize efficiency at the MTA. It will also be a great source of the information that the pyro op needs to submit to the local Fire Marshal as part of the permit process and as a courtesy to the Kern County and Randsberg fire authorities. Thirdly, flight testing at the MTA requires a 45 day in advance notification to the FAA. This information will help the permit process here as well.

There is one other major use planned for the information gathered in this form. At the conclusion of a test weekend, a copy of the completed forms will be sent to the editor of the news letter for publication in the next RRS News issue. This will help the editor by providing meaningful and interesting information about current projects. It should also be helpful to the membership in keeping up with recent events. At the end of each calendar year, these forms and any amplifying information (i.e. photos, graphs, data plots, etc.) provided by the owner of each project will be published as a consolidated report. This will be a great boon to documenting and disseminating information gathered during the year, and will give some well deserved notoriety to the people conducting projects.

Everyone who is planning to fire a solid rocket needs to keep a copy of this form handy. It would also be appreciated , if your firing a standard BETA, if you could prepay the propellant costs at the time the form is submitted to the director of research and the pyro op. Instead of waiting to measure the exact propellant load and charging by the pound, a standard propellant/squib/burst diaphragm cost of $30.00 has been established. For much larger or smaller vehicles a bill will be prepared and sent to the builder. However, for the standard BETA launches, prepayment of the $30 propellant cost will help alleviate the administrative burden of sending out billing long after a firing. This will also be a good indicator of who really plans to show up with a rocket so that the Society does not mix and waste propellant for a rocket that does not make it out to the MTA. Thanks in advance for your cooperation with the standard record forms and propellant payments.

Reaction Research Society
P.O. Box 90306
Los Angles, CA 90009

STANDARD RECORD FORM
SOLIDS

☐ Static
☐ Flight

To schedule your rocket for testing please fill out grey area. 6/94

Owner's Name: ____________ Date: ____________
Address: ____________ Affiliation: ____________
City: ____________ State: ____________ Zip: ____________
Phone: ____________ FAX: ____________ Date Ready to Test: ____________

PHYSICAL DATA:

Est. Altitude ____________
Total Lgth. ____________ Motor O.D. ____________ Motor I.D. ____________
Fuel Lgth. ____________ Fuel Vol. ____________ CG_f ____________ CP ____________
Total Rocket Wt. Empty ____________ Throat Dia. ____________ Exit Dia. ____________
Motor Matl. ____________ Nozzle Matl. ____________ Diaphragm ____________
Fuel ____________ Mix Ratio ____________
Equip. & Instruments Aboard: ____________
Identification (color, shape, etc.) ____________
Purpose of Test ____________
Rocket Inspected by ____________ Date ____________

Date of Firing ____________ Place of Firing ____________ Firing Order ____________
Chief Pyro-op ____________ Asst. Pryo-ops ____________
Total Rocket Wt. Fueled ____________ Wt. Fuel ____________ Density ____________
Total Wt. Flying ____________ Wt. of Total Rocket Recovered ____________

ENVIRONMENTAL DATA:

Time ________ Temp ________ °F Wind From ________ Vel ________ Humidity ________ Baro. Press. ________

LAUNCH DATA:

Launch Angle ____________ Range In Use ________ Launch Rack No. ____________

PHOTO DATA:

Camera ________ mm, f ________, Fr/Sec ________ Location ________ Tripod ☐ Operator ________
Camera ________ mm, f ________, Fr/Sec ________ Location ________ Tripod ☐ Operator ________

TRACKING DATA:

Time Up ________ by ____________ Time Impact ________ by ____________ Sight ☐ Sound ☐
Time Up ________ by ____________ Time Impact ________ by ____________ Sight ☐ Sound ☐
Time Up ________ by ____________ Time Impact ________ by ____________ Sight ☐ Sound ☐
Peak Azimuth ∠ ________ Azimuth ∠ Crossing Range ____________ (ft) Station ________
Peak Azimuth ∠ ________ Azimuth ∠ Crossing Range ____________ (ft) Station ________
Peak Elevation ∠ ________ Deviation From Range ____________ Station ________
Peak Elevation ∠ ________ Deviation From Range ____________ Station ________
X Dist Down Range ________ Y Dist L/R of Range ________ / ________ True Range ________
Time to B.O. ________ B.O. Ht. ________ B.O. Vel. ____________ Acc. Max. ________
Est. Ht. ________ Time Ht. ________ Trig Ht. ________ Computed Ht. ________
Comments ____________

____

____________ Director of Rearch ____________