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10,000 POUND THRUST LIQUID PROPELLANT ROCKET ENGINE

Status Report; September, 1994

by

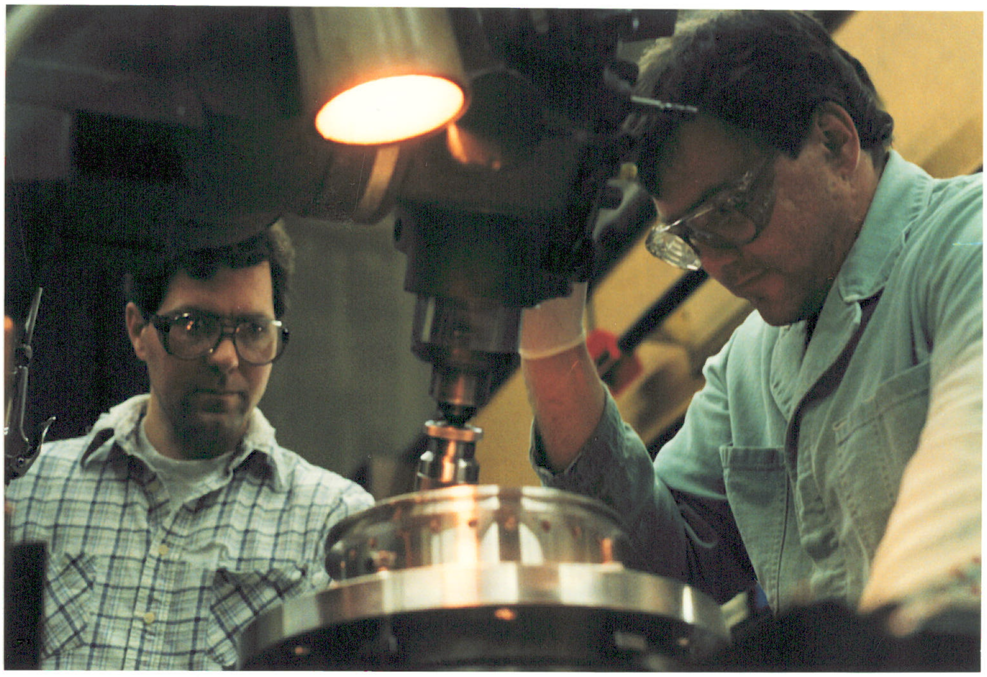
D. Crisalli, B. Wherley, S. Claflin, G. Garboden, T. Mueller, M. Grant

For no apparent reason other than an inherent interest in doing so, a few members of the RRS decided one day that it would be a worthwhile exercise to build and test a ten thousand pound thrust liquid propellant rocket engine. After several initial tongue-in-cheek discussions about how crazy such an undertaking might be, we decided we could not continue to live with ourselves if we did not do it. So, with that popular mandate from amongst our own ranks, we began.

The design concept was fairly simple. The engine would burn liquid oxygen and kerosene. It would be cooled ablatively and propellants would be pressure fed from test stand tanks. It would be a static test engine only and, as a consequence, would not be built to a flight weight configuration. The initial design parameters were quickly established and are listed in table 1.

Hardware fabrication for the thrust chamber components began rather quickly to take advantage of some surplus castable ablative that became available. The design (figure 1) shows that the combustion chamber wall is protected by the cast ablative (Dow Corning 93-104). The throat is machined from solid graphite, and the exit nozzle is an ablative made of laminated hardwood (oak) impregnated with a polymer resin. All of this hardware has been completed and is shown in photo 1.

From the start there was some difference of opinion among the team members about the design of the injector. Some wanted to build a large pintle type while others were more comfortable with the traditional flat faced style. The solution was to build one of each and test both. (Sort of a LOX / kerosene shoot-out at the MTA coral, as Mark Grant called it in a previous status report). Both designs were completed and hardware



Brian Wherley (right) and Scott Claflin drill injection orifices in the flat face 10k injector at George Garboden's shop.

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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. At this location, several hundred rockets, using both solid and liquid propellants, have been static tested and launched. Currently there are over 140 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

TABLE 1
10,000 Pound Thrust Engine Parameters

Thrust (lbs)	= 10,000
Specific Impulse (assumed) (sec)	= 243
Chamber Pressure (psia)	= 350
Thrust Coefficient (assumed)	= 1.400
L* (inches)	= 40.00
Mixture Ratio (O/F)	= 2.20
Chamber Diameter (inches)	= 8.50
Exit Diameter (inches)	= 10.00
Fuel Side Injector Pressure Drop (psi)	= 52.5
LOX Side Injector Pressure Drop (psi)	= 52.5
Throat Diameter (inches)	= 5.097
Expansion Area Ratio	= 3.854
LOX Flow Rate (lb/sec)	= 28.292
Fuel Flow Rate (lb/sec)	= 12.860

fabrication initiated. Figure 2 shows the pintle injector design and Figure 3 shows the flat faced design. Components of the flat face injector are shown in photo 2 and the completed injector appears in photo 3. This injector uses three rows of split triplet injection elements and is configured with face baffles to enhance combustion stability.

The testing of an engine this size is much more complicated than is building the hardware itself. No facilities existed at the beginning of the project that were usable for such a test. The first major undertaking to remedy this situation was to design and construct a test stand capable of supporting engines producing up to approximately 25,000 pounds of thrust. This effort was spearheaded by George Garboden. The pad for this test stand was built at the MTA last year using half a ton of rebar and forty thousand pounds of concrete (Photo 4). An iron support structure was also designed and built to fit the hard attach points on the 10K pad, (as it is now refered to). This structure will eventually support the engine thrust mount, tankage skid, and peripheral support equipment.

The existing blockhouses are not adequate protection for the test crew when firing an engine of this size. A new underground blockhouse has been designed and will be built as part of the general MTA improvement plan. The blockhouse will be located just outside the old compound fence line between the bunkers and the 10K pad. This will move the blockhouse crew further away from the 10K pad than they would be in the older blockhouses, but will not obscure the view from the bunkers. The new blockhouse will also include a data collection room where the new digital data systems now being assembled can be set up in a somewhat cleaner and cooler environment.

The firing of an engine this large requires that large quantities of propellant be supplied to the engine every second, and the tankage needs to have the capacity to run the engine for some respectable period of time. An added complexity for the RRS is that this equipment cannot be left at the MTA without the risk of theft or vandalism. To solve these issues, a portable skid has

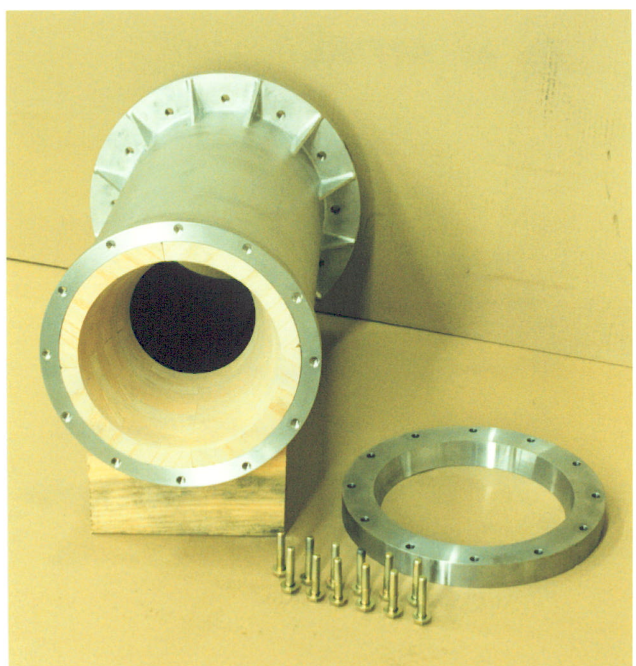

Photo 1 - The completed thrust chamber assembly including the housing, throat, chamber liner, exit nozzle, and nozzle retaining ring.

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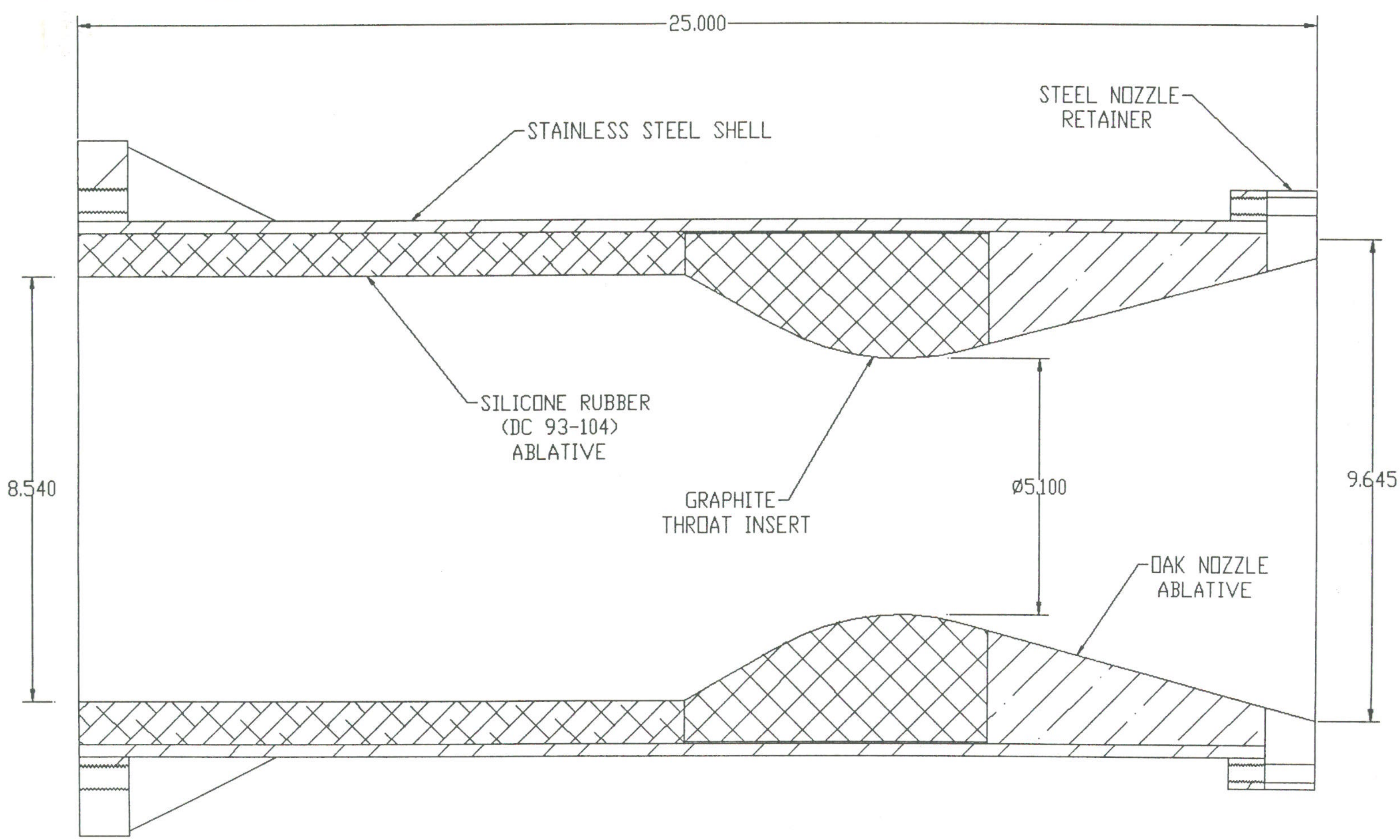


Figure 1 - Cross section of the 10k thrust chamber.

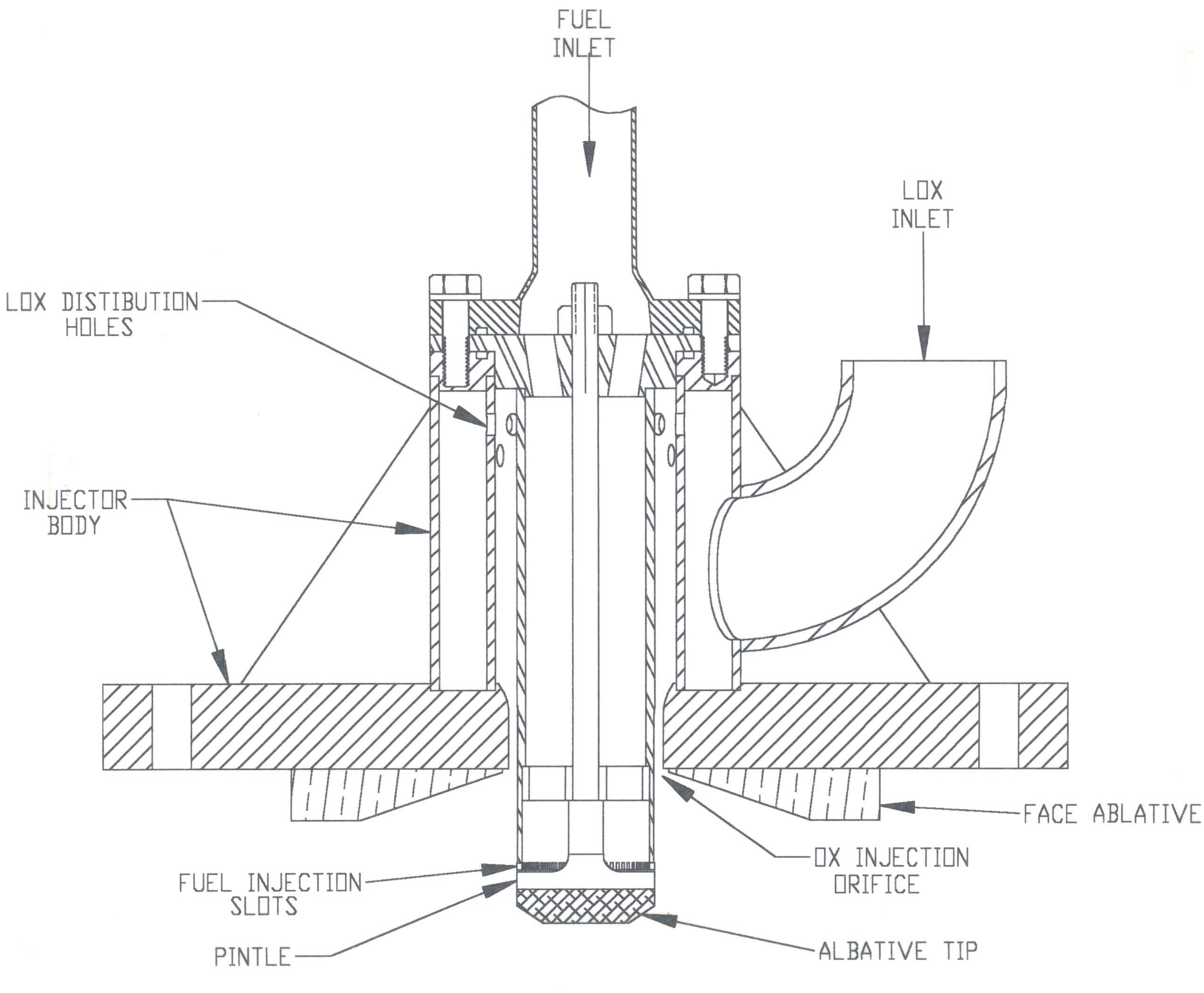


Figure 2 - Cross section of the 10k pintle injector.

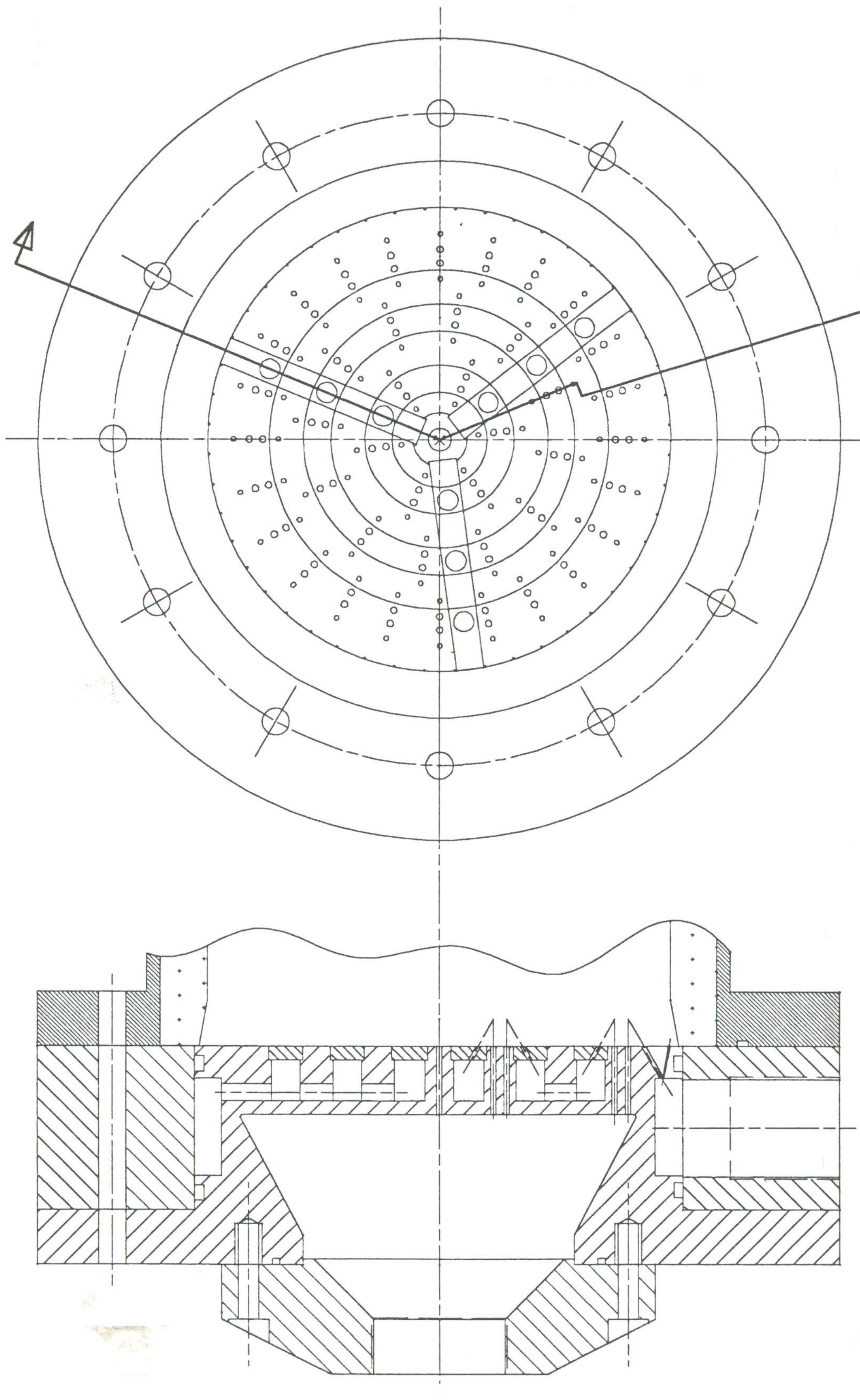


Figure 3 - Diagram of the flat faced 10k injector.

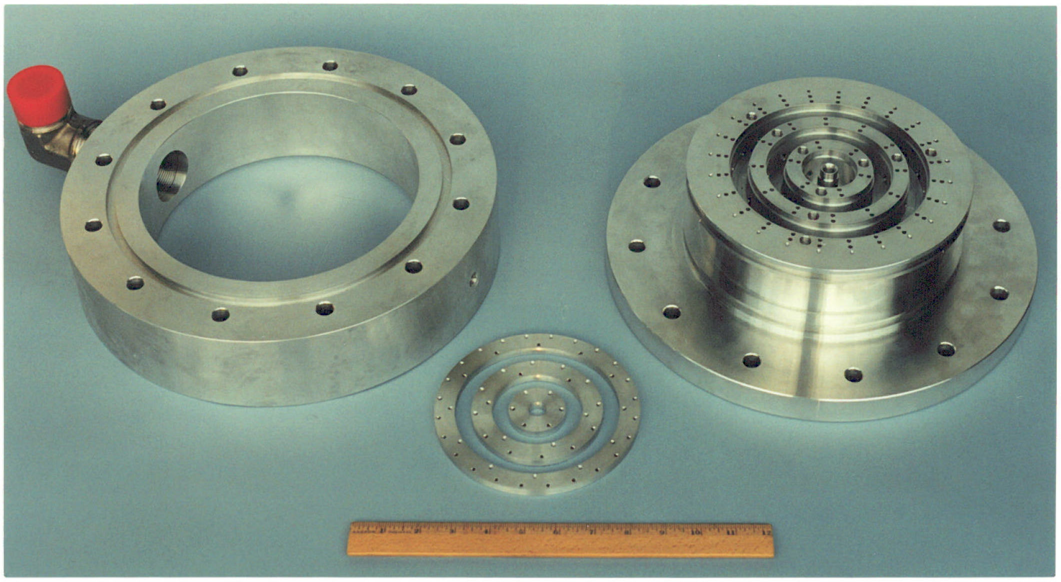


Photo 2 - Completed components of the flat face injector. The fuel manifold ring (left), the face rings, and the injector body (right) are shown here.

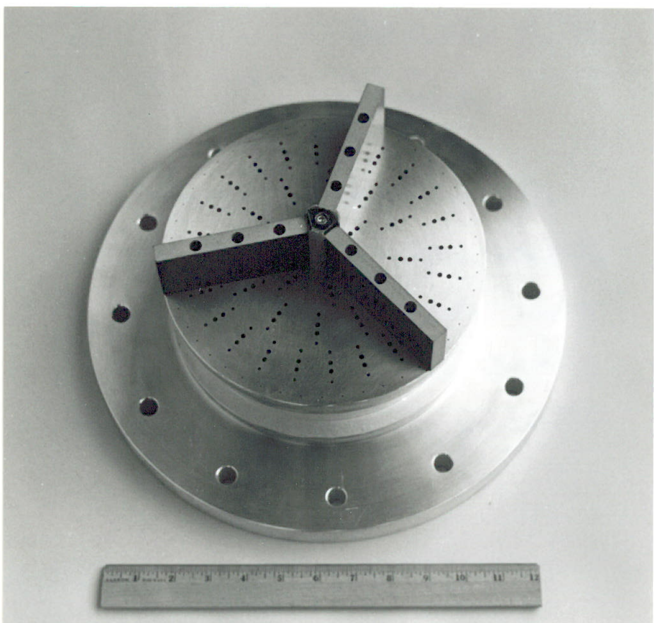


Photo 3 - The completed flat face injector with the fuel rings dip brazed in place and silica/phenolic acoustic baffles installed.

been designed that will hold all the tankage, valves, electrical control, and instrumentation equipment necessary to run the engine. With some financial assistance from PacAstro of Herndon, Virginia in exchange for data and information about the project, material for the 18 inch diameter, 7 foot tall tanks has been procured. In addition propellant flex lines and two main propellant valves have been purchased with the remainder of these funds. The tanks will hold enough propellant to run the 10K engine for 20 seconds and will be capable of being pressurized to over 1200 psig. The tanks, and all the rest of the necessary test support equipment, will be built onto a dedicated trailer. This will allow the entire unit to be assembled and checked out in the shop and then transported out to the MTA in support of hot fire testing as required.

Activity in the design and construction of all the required engine hardware, support equipment, and facility improvements has started. Some areas, such as the thrust chamber and flat faced injector, have already been completed. The new blockhouse will probably take the most time and will be somewhat dependent on outside forces such as transport, monetary resources, and weather. Architectural drawings are now in work for both the blockhouse and new concrete bunkers. When the plans are completed, bids will be solicited from local contractors to see if this work can be done at reasonable cost. If not, plans are being made to carry out the required construction with slave RRS labor. Because of the scope and complexity of this effort, progress is dependent on how fast the equipment and raw materials we need can be acquired. All of those working on this project have contributed large amounts of their own money and even more of their time toward the goal of firing this engine. Although short on resources, the group is long on enthusiasm and ingenuity.

Without a doubt, this is one of the most extensive and energetic group projects undertaken by RRS members in its history. It also promises to be one of the most spectacular when the testing starts. We will keep you posted on progress and the schedule for testing.

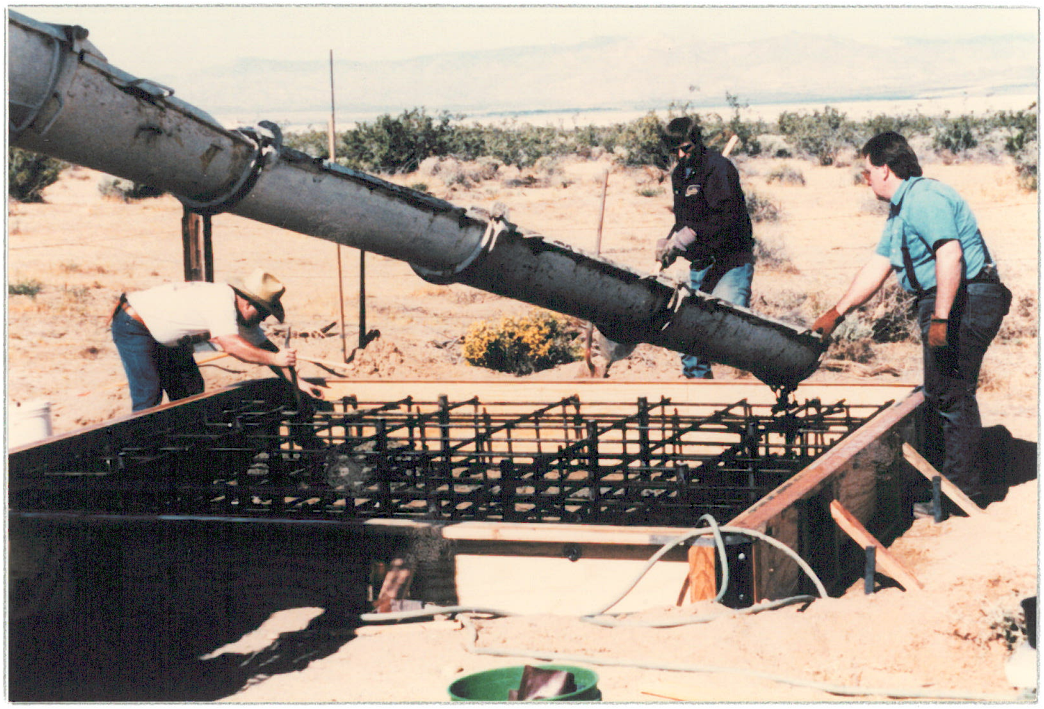


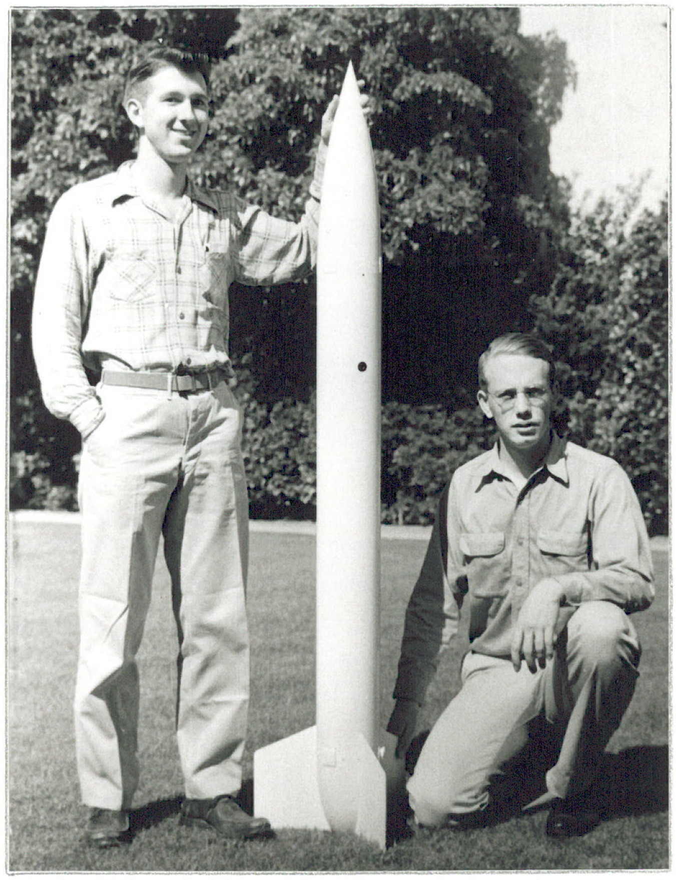
Photo 4 - Chip Basset (left), George Garboden, and Mike Gotlieb (right) start the pour of 10 yards of concrete during the construction of the new 10,000 pound thrust test stand.

HYDROGEN PEROXIDE AND DETERMINATION

by

David E. Crisalli

Over the years, there has been much discussion within the RRS about the classic and award winning Rosenthal and Elliott hydrogen peroxide work. Due to the rapid growth in membership over the past year and a half, there are many newer members who have never seen information about this rocket and the two young students who built it. This article is offered to bring the technical excellence of their work back into view. It is also worthy of note that both Dr. David Elliott and Mr. Walter Lee Rosenthal are now honorary members of the RRS. Mr. Robert DeVoe and Mr. Carroll Evans, also honorary members, helped with the testing of this rocket in 1950 and contributed many of the photographs used in this article.



David Elliott (left) and Lee Rosenthal, designers and builders of the hydrogen peroxide-solid catalyst rocket, pose beside their completed product.

The Second World War had only been over for four years, but the world had been forever changed by many aspects of that great conflict. For the technical community in the United States, the possibility of space exploration had loomed larger than ever before as the first V-2's lifted off from the proving grounds at White Sands, New Mexico. While Dr. Robert Goddard had been looked upon as a crackpot here in the U.S. for most of his career, the Germans had been busy buying copies of everything he patented or published from the U.S. Printing Office.

When the first intact V-2 was captured by the Allies during the War and brought back to the United States for inspection, Dr. Goddard was asked to examine it. Even before the access covers were removed, he would tell the Army people with him exactly what component they would find beneath. They were amazed. How could he know all this about a newly captured, top secret, enemy weapon? He didn't tell them that the V-2 was almost identical to a drawing for a vehicle he had proposed in a paper published in 1919.

Dr. Goddard died in 1945 before the magnitude of his contribution to space exploration was realized. But the seeds had been planted, and by the late 1940's and early 1950's, the world watched as the Soviet Union and the United States rushed headlong into the "Space Age". Besides their utility as weapons, the rockets being constructed fired not only their engines, but the imaginations of millions about the possibility of flight beyond the atmosphere of earth. The dream burned particularly brightly in the minds of many young students for whom the future held all the allure of sailing to the New World. In 1948, two such students and a newly formed research society would try their own hand at "slipping the surly bonds of Earth".

I was eight years old in May of 1961 when, on a small black and white television screen, I watched Alan Sheppard roar into the heavens on a gleaming, graceful machine. I had always been interested in science, and stories of invention had always held my attention. And there on the TV that day was the culmination of all man's scientific knowledge and power of invention. Everything we had learned over the past few millennia had been brought together to allow one of our own kind to escape the bonds of earth briefly and return safely. We had opened the door, if even only the tiniest crack, to a universe beyond our traditional realm.

There was a feeling in those days that, with hard work and dedication, we would certainly travel to the moon and possibly beyond before the end of the century. It was an exciting time as the nation progressed from that first Redstone flight to the mighty Saturn V that would take the first humans to another celestial world before the decade was out. The dream of a Space Age had emerged from the rubble of the Second World War and now it was becoming reality.

In those years I read all I could find about Goddard, and rockets, and space travel. I lived in southern California where many of these great space machines were being built. The excitement of the Space Age was all around. But in 1966 as an eighth grade student, I wanted to do more than read. I wanted to build and fly my own machines to really understand how they worked and to learn more than the books could teach. I first learned of, and then joined, a group of amateur rocket builders called the Reaction Research Society. They had been founded in 1943 and in their ranks I saw students, engineers, tinkerers, and philosophers. Some were engrossed with the thought of traveling to distant nebulae. Others, with a more practical bent, started smaller and only worried about how to get to a rocket up to a 1000 feet in an orderly fashion. I watched many of the RRS members design and build, what were to me then, fantastic rockets that exploded out of their launch racks in the Mojave Desert on towering pillars of fire and smoke. These were not cardboard models with minuscule motors producing ounces of thrust. These were thundering metal machines, many feet long, producing thousands of pounds of thrust, and they flew into the clear desert skies at unbelievable speeds. Their construction required machining parts and welding structures just like the gargantuan vehicles that were taking men into space. Much to the concern of my parents, this was for me.

In 1967, at a Reaction Research Society meeting in a tumble-down little clubhouse in Gardena, California, I saw something that impressed me so strongly that I remember the feeling to this day over 27 years later. It was a silent, 13 minute long 16mm film about the rocket project documented in this article. I sat in utter amazement watching the fabrication of engines, struts, nose cones, and launch towers. Two high school / college students, who, at the time of the project, were not much older than I was then, were designing, building, and successfully flying a liquid fuel rocket. I was dumfounded. I was awestruck. I was inspired.

Over the next 25 years I built many solid and even liquid rockets of my own. But always I would compare my work to Rosenthal and Elliott and would strive to emulate their skill, professionalism, and technical excellence. Even during my many years of serving at sea in the U.S. Navy, I would often tell the engineers and ordnancemen about the rocket built by these two young students. But the report that Walter Lee Rosenthal and David Elliott wrote on their project tells the tale so well, I will unfold it for you in their own words with only limited interference from me. So let me begin by setting the stage.

Once upon a time in a land of unlimited sunshine and promise (i.e. southern California before the collapse of the aerospace industry), two young high school students began to dream of building a machine designed to explore the upper reaches of the atmosphere... and possibly beyond....

In the opening paragraphs of their original report, Rosenthal and Elliott confess candidly why they undertook this particular project. "After the financially profitable Rocket Mail Flight held by the Reaction Research Society at Trona, California in March, 1948, the authors of this report felt that the RRS was in a position to undertake a modest liquid propellant program having as its goal the development of a simple vertical sounding rocket capable of carrying a few pounds of payload. The uses of such a rocket are few, if not entirely nonexistent, but the possibility of using such a rocket for inexpensive upper-atmosphere research furnished us with an excuse for undertaking the project. Actually, we were motivated chiefly by the intrinsic interest of building a liquid propellant rocket." This last sentence provides a good insight into the inquisitive nature of these two young experimenters.

Their report continues..."During the summer of 1948 we carefully considered all of the possible propellant combinations that might be used, seeking, in particular, a propellant combination that would minimize the amount of work that would have to be done in constructing the rocket. We finally chose hydrogen peroxide as the propellant because it would permit us to build the simplest possible liquid propellant rocket. The rocket would use only one liquid and would not need to be cooled. The performance of the rocket would not be high, because hydrogen peroxide when used as a monopropellant gives a specific impulse of only about 120 seconds, but the simplicity of the rocket would outweigh this disadvantage."

The vehicle was designed to carry a two pound smoke generator as a payload to aid tracking. The desire for a rapid take off led to the establishment of a thrust goal of 200 pounds and a maximum gross take off weight of 50 pounds. The rocket was designed around the use of a war surplus, stainless steel, D-2 aircraft breathing oxygen tank as the main propellant tank. Twenty three inches long and just under six inches in diameter, these tanks were compatible with concentrated (90%) hydrogen peroxide, had a rated working pressure of 400 psi, weighed only 4.5 pounds, and could hold 25 pounds of peroxide. The design of the rocket is shown in Figure 1.

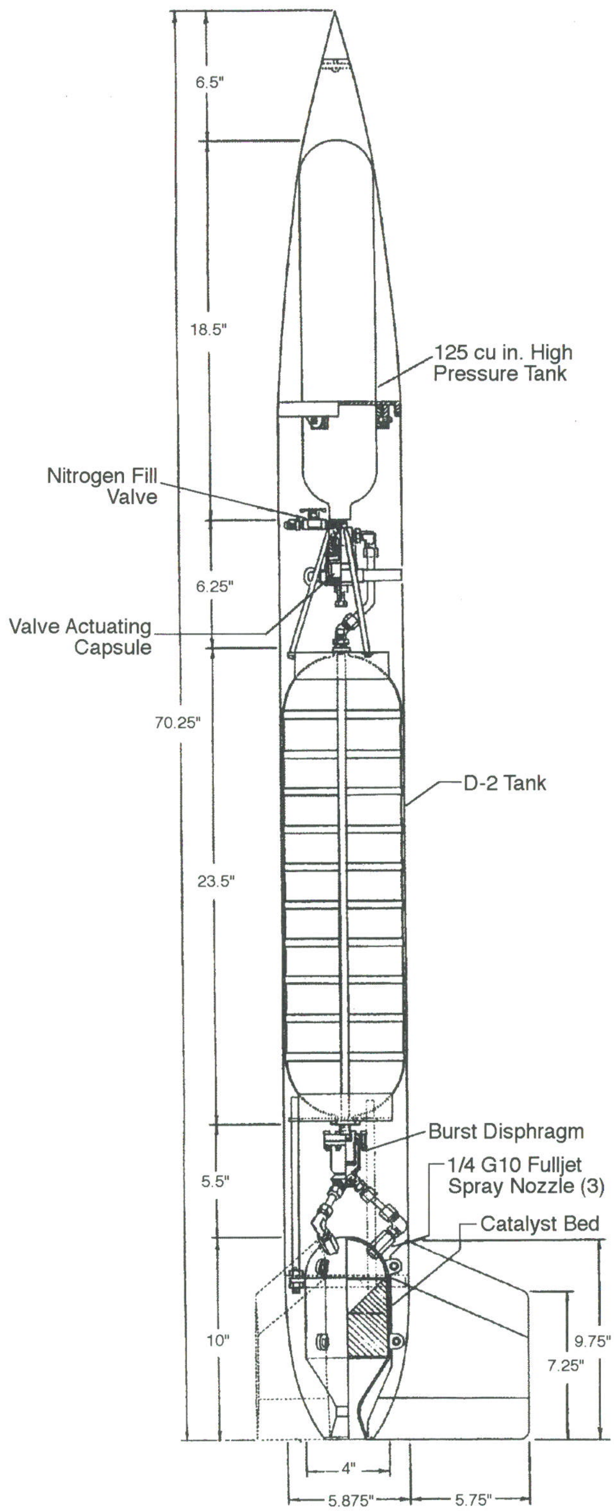


Figure 1 - Cross section of the hydrogen peroxide monopropellant rocket.

The engine design began with a 4 inch diameter chamber. This allowed adequate room in the planned 6 inch diameter boat tail for mounting provisions. The catalyst bed was designed to decompose the required 1.7 pounds of peroxide per second and would have a pressure drop across it of 100 psi. With the tank pressure established at 400 psi, a 100 psi pressure drop across the injection spray nozzles, and a 100 psi drop across the catalyst bed, the engine was designed to produce 200 pounds of thrust at a chamber pressure of 200 psi. The nozzle was correctly expanded for 5,000 feet above sea level.

The engine nozzle and head end closure were machined from 1020 steel. The cylindrical thrust chamber was made from a section of 4 inch stainless steel tubing. The motor nozzle and chamber were originally flanged to allow removal and replacement of the catalyst bed if required. These flanges would later be machined away and the parts welded together to lighten the engine for flight. The catalyst bed, prepared by producing a coating of manganese dioxide on alundum pellets, was held in place by stainless steel screens. The lower screen was strengthened by backing it up with an eighth inch thick plate perforated with 68 quarter inch diameter holes. Mounted to the engine with an arrangement of struts, the peroxide tank was also fitted with a burst diaphragm valve to control peroxide flow. It was designed to release the peroxide when the tank pressure reached 120 psi. With this much of the propulsion hardware designed and built, Rosenthal and Elliott built a static test stand and data collection equipment in preparation for a series of several static tests.

"During January of 1949, we constructed the static test facility for the rocket in Mint Canyon north of Glendale, California. The test stand consisted of a rigid mount, anchored in concrete, which would hold the rocket in a vertical position (photo 1). The nitrogen feed control valves and instruments would be located 30 feet away behind an earth embankment. The data we wished to obtain were nitrogen tank pressure, peroxide tank pressure, chamber pressure downstream of the catalyst bed, and time. These would be obtained by photographing three pressure gages and a sweep second timer with a 16mm movie camera."

The first tests were run on 26 February, 1949 on the flight peroxide tank and engine, but using a facility

Photo 1 - Lee Rosenthal (left) and David Elliott set up the propulsion unit in the test stand for one of the first static tests. The facility nitrogen pressurization is being attached. The "I" beam in the background provided protection for the test crew.

gaseous nitrogen supply. The peroxide tank pressure was initially controlled with a manually operated needle valve to establish the correct nitrogen flow rate. The first run was made with only six pounds of peroxide in the tank, but the tank pressure only reached 200 psi before the peroxide ran out. The second test used eighteen pounds of propellant. Again started with the needle valve, tank pressure built to 120 psi where the burst disk ruptured and started propellant flow. The needle valve was opened further until the tank pressure rose to the required 400 psi. The engine ran smoothly until peroxide was again expended. The third run used a full twenty five pounds of peroxide. The needle valve had been left in its final position after test 2 and, this time, the engine had been started with an in line quick acting (ball) valve. The chamber pressure rose quickly to 190 psig and the motor ran for 15 seconds (photo 2).

The data from this test was used to calculate the engine performance. Again from the original report, "The average chamber pressure was 175 psig, the weight of peroxide used was 24.3 pounds, and the throat area of the nozzle was 0.720 square inches. Thus the overall c* was 2700 feet per second. We felt this was sufficiently close to the theoretical c* of 2950 to indicate that the motor was functioning properly. In addition, our data and calculated exhaust velocity included both starting and stopping. Using 1.33 as the thrust coefficient, we calculated that the thrust obtained was 197 pounds."

Originally, Elliott and Rosenthal planned to use a regulator between the flight pressure bottle and the peroxide tank to control the flow of nitrogen. However, the tests with the needle valve had been so successful that they decided to use a simple orifice to control the peroxide tank pressure. Although the chamber pressure and thrust would decay slightly over the operating time of the engine, this was a much simpler, lighter, and less expensive option. (This same method was again used by RRS member Mark Grant in his successful bipropellant liquid rocket launched over forty years later.)

To determine the orifice size required, two more static tests were run in early March of 1949. Since they wanted the thrust a little higher than that produced during the first test runs, they opened the needle valve enough on the next test to bring the peroxide tank pressure up to 500 psi. The stem of the needle valve was then soldered in position and another test run to verify the setting was still correct. The valve was removed from the apparatus and flow tested in the shop to determine an equivalent orifice size.

Photo 2 - Static test Number 1 run on 26 February, 1949.

The flight nitrogen tank, which we next added to the rocket, was a war surplus 125 cubic inch oxygen tank weighing 4 1/2 pounds. We hydrostatically tested the tank to 3000 psig. This tank was attached to the peroxide tank with three struts in the same manner as the motor. The outlet of this tank was connected to the inlet of the peroxide tank through a valve which was to be actuated by current from a dry cell. The stem of this valve was held in the closed position by a piece of solid propellant 7/8 inches in diameter and 5/8 inches thick, cast from a mixture of 75% potassium perchlorate and 25% Baker casting resin. This capsule was held firmly against the stem by a screw to seal off the compressed nitrogen, and, at the same time, keep the peroxide tank vented to atmosphere. Closing the fire switch ignited this capsule (which burned in less than a second)

Photo 3 - Close up of the pyrotechnically actuated high pressure nitrogen valve installed between the peroxide tank and the flight nitrogen bottle. The struts held the pressure bottle in place. The bolt head at the lower end of the pyro valve was used to push the solid propellant capsule against the valve stem closing off the high pressure nitrogen and keeping the peroxide tank vent open. The valve opened when the capsule was burned.

permitting the nitrogen to force the stem into the open position. This sealed off the system from the atmosphere and allowed nitrogen to flow through the metering orifice into the peroxide tank." This part of the rocket is shown in photo 3.

By late March 1949 the flight pressure tank and valve had been added and were ready for test. In the next static run, the peroxide tank pressure only reached 400 psi instead of the 500 intended. The metering orifice was enlarged slightly and another test was run. A plot of the data from this test is shown in figure 2. Before this last run, the engine had been lightened by removing the flanges and welding the nozzle to the chamber. Although no degradation of the catalyst bed had been observed during the many static tests, it was replaced with a fresh batch before the engine was welded shut. Photo 4 shows the propulsion system undergoing on of the final static tests.

"Now all that remained was to add fins and shell. The fin area required for a fin-stabilized rocket increases with the Mach number at which the rocket is to operate. The highest Mach number the peroxide rocket could possibly attain would be about 1.8, and we chose the fin

Photo 4 - Static test Number 6 run on 27 March, 1949. This test included the flight pressure bottle and valve assembly.

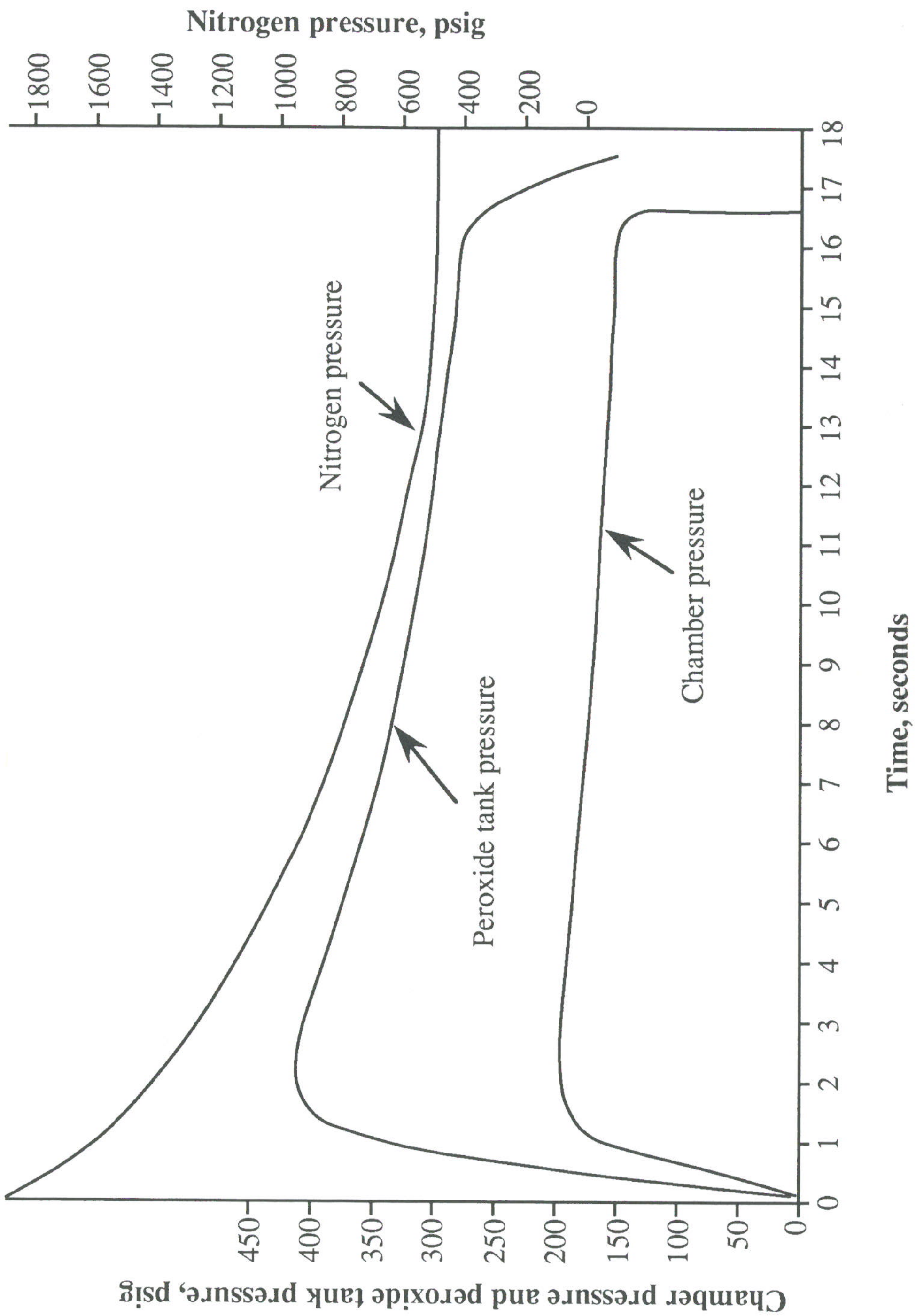


Figure 2 - Test run #7 data plot (April 14, 1949)

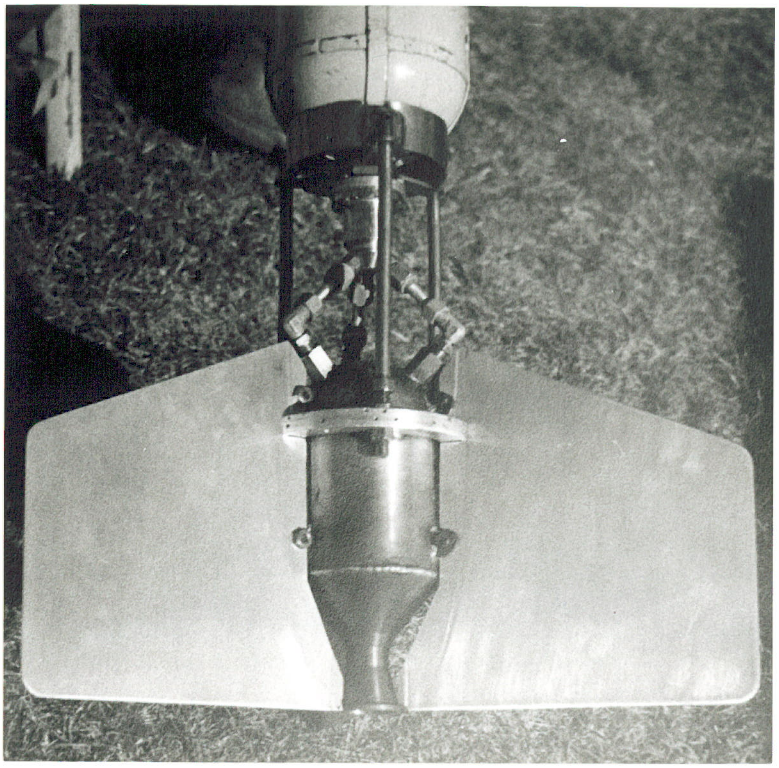


Photo 5 - **The decomposition chamber/nozzle and lower portion of the peroxide tank are shown here after the addition of the three sheet magnesium fins. The burst diaphragm valve can be seen just below the tank.**

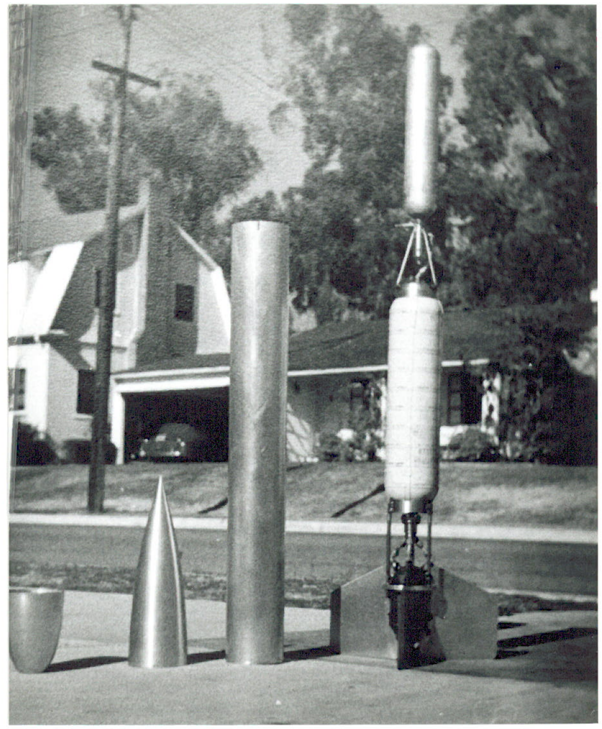


Photo 6 - **All the major subassemblies of the hydrogen peroxide monopropellant rocket.**

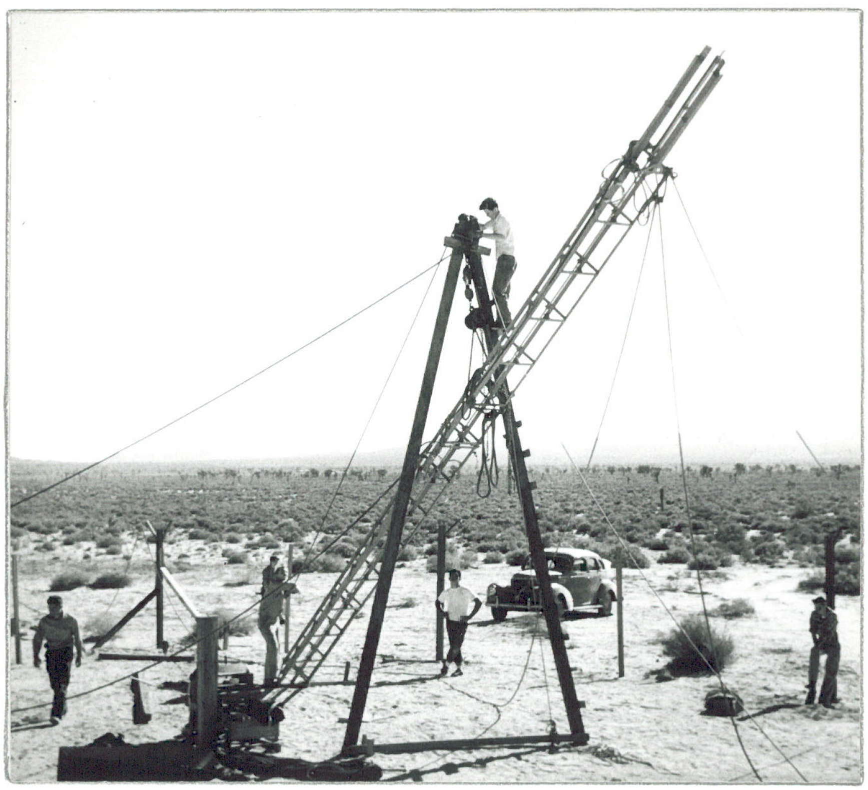


Photo 7 - **Members of the RRS erect the 40 foot launch tower at the test site near Mojave, California.**

area so that the center of pressure of the rocket would be six inches aft of the center of gravity at that Mach number. The three fins were cut from 1/8 inch sheet magnesium and tapered to a knife edge at the leading and trailing edges. They were bolted to lugs on the motor, the lugs having been carefully machined parallel to the axis of the nozzle. (Photo 5). The shell which covered the rocket consisted of spun aluminum nose and tail sections, and, between them, an aluminum tube rolled from 0.020 inch sheet. ... The completed rocket weighed 24.5 pounds empty, and cost about $100 to build." (Dollars apparently went a heck of a lot farther in 1950!) Photo 6 shows all the major components of the rocket.

Having completed the flight vehicle, Rosenthal and Elliott began work on what became the hardest part of the project - a forty foot tall, heavy steel launch tower. They designed it to be built in two sections out of welded steel channel.

Triangular in cross section, it had adjustable steel guide rails and could accommodate rockets of various diameters. It was built by September of 1949 and transported to the Mojave test site in October. But after all their effort, Murphy's Laws were about to take effect. "The raising operation proved to be of considerable magnitude, and it was only with the aid of much manpower and a large supply of ropes, hoists, and auxiliary wooden framework that we succeeded in raising the tower to a vertical position. (Photo 7). Unfortunately, there was insufficient manpower and rope to control the tower in the high wind which arose during this time, and the tower was blown down. The mishap demolished 10 feet of the tower and twisted the rest of it, so we were forced to bring the tower back to the shop for rebuilding. On November 26, we returned to the test area and succeeded in raising and anchoring the tower without mishap. Finally, with the aid of a transit, the tower was aligned to a vertical position by adjusting turnbuckles of the six guy cables." (Photo 8)

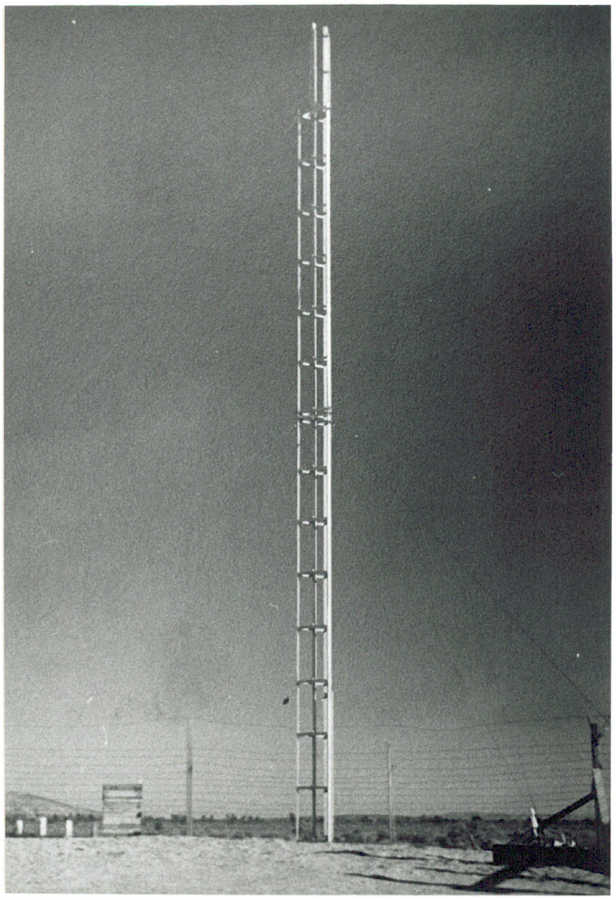


Photo 8 - After one catastrophe, the tower is repaired, raised, and anchored in place

On 12 February, 1950 the rocket was static tested one final time. It was fully assembled and anchored inside the base of the launch tower. This test was to be a complete rehearsal for launch with the exception of letting the vehicle fly. It proved to be most fortunate that this test was conducted because it uncovered a simple flaw that would have proven catastrophic in flight. Ignition of the solid propellant capsule that opened the nitrogen pressurization valve generated so much hot gas inside the skin of the rocket that the skin seam burst. This was a surprise since the three 5/8 inch diameter holes provided in the skin to vent these gasses were thought to be more than sufficient. The design was modified, new exhaust tubes fitted, and a new skin section made. The final touch was a one pound smoke flare in the nose and a white paint job. (Photo 9)

Photo 9 - Lee Rosenthal poses with the rocket.

Elliott and Rosenthal had planned to track the rocket optically and, so, built a phototheodolite from a 16mm movie camera and a Buff theodolite. (I can hardly pronounce it let alone build it!). "The movie camera had a 13 inch focal length main lens and a 35mm auxiliary lens which, with the aid of a small mirror, placed an image of the edges of an angle-of-elevation scale and a sweep second timer on part of the 16mm frame. The operator would watch the rocket through the eyepiece of the theodolite while moving the instrument with a handle-bar."

This is just another example of how these two senior high school / college freshmen thought of, designed, and built everything they needed to complete this project even, as in the case of the launch tower destruction, in the face of great disappointment and set backs. The phototheodolite was not the limit of their tracking and data collection efforts either. "In addition, we hoped to measure the velocity of the rocket at burnout with a 4 x 5 Speed Graphic camera located with the phototheodolite three miles from the launch site. The camera would be aimed at the area of the sky in which the rocket should be at burn out, and a record of time between the two exposures would be obtained from a flashlight bulb mounted beside the timer in the phototheodolite and connected to the Speed Graphic flash synchronizer. A 16mm movie camera mounted 600 feet from the launch tower would complete our effort at tracking the rocket."

Photo 10 - Glen Maxon conducting a fit check of the rocket in the launch tower during final rail adjustments.

To test out the tower and tracking equipment before committing to a launch of the peroxide rocket itself, a zinc / sulfur rocket was built and fired on 9 April, 1950. Although the theodolite operator could not track this rocket due to its very high acceleration, all seemed to be ready for the liquid rocket flight. The date was set for 14 May, 1950.

"On 12 May, a small group arrived at the test area to begin preparations. We first spent several hours working on the launch tower, adjusting the rails for 1/16 inch clearance from the blocks on the rocket. (Photo 10). The rocket was pulled up and down the tower frequently as a check. The ignition cable was then laid from the tower to the control box 250 feet away, and a radio transmitter was set up to communicate with the tracking station on a hill top three miles to the southeast. The evening before the flight, we placed the rocket in the tower, bolted the door shut, and left everything in readiness so that the rocket could be fired as soon as possible after sunrise. Early morning in the Mojave Desert promised the least wind and best tracking conditions."

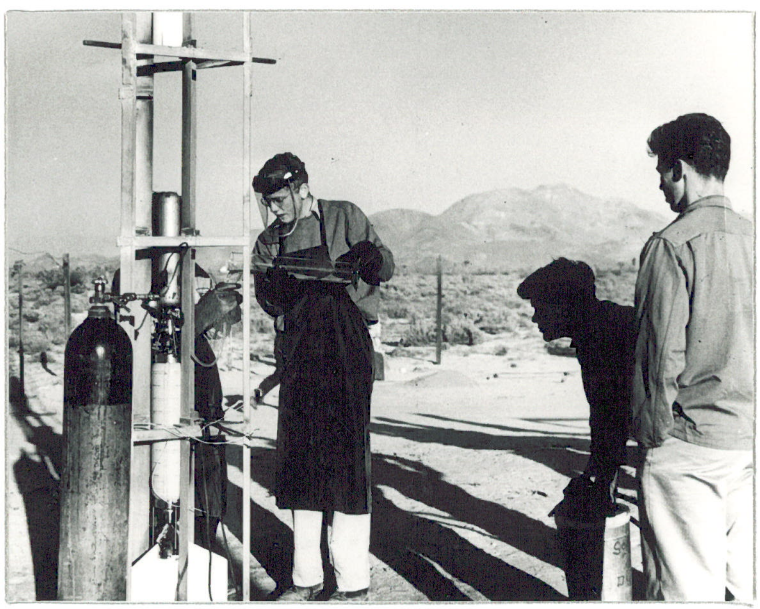


Photo 11 - Lee Rosenthal adds 90% hydrogen peroxide to the propellant tank just prior to the flight. The auxiliary nitrogen tank has been connected to pressurize the flight nitrogen tank. Part of the aluminum shell can be seen resting on a crosspiece in the tower just above the rocket.

"As the sun rose at 6:30 the morning of the 14th, the sky was clear, but there was already a brisk wind blowing. A group left for the tracking station with the instruments and radio transceiver. At the launch site we lifted the shell from the rocket and rested it on a crosspiece in the tower above the rocket. We connected a cylinder of nitrogen to the (flight) nitrogen tank and pressurized the tank to 2000 psig. We then filled the peroxide tank (photo 11), slid the shell back onto the rocket, and fastened it with the set screws. Finally we connected the ignition wires to the squibs in the smoke flare and the nitrogen valve. At 7:45 the rocket was ready for flight, and we retired to the control station 250 feet away."

"We counted down and fired the smoke flare, but the flare failed to ignite properly and gave only a thin stream of smoke. After ten seconds it was burning no better, so we made the final count-down and fired the rocket. The motor roared to life, and the rocket lifted quickly out of the tower, clearing it at 80 feet per second as recorded by the movie camera. As the rocket left the tower, the wind caught the fins, and the rocket rotated 15 degrees from the vertical. It continued to climb at this angle with rapidly increasing velocity. In a few seconds the rocket was only a distant white speck which quickly became too small to see. Twenty seconds after takeoff a thin, distant vapor trail appeared, streaking across the sky to the northwest, and after another twenty seconds this also became too faint to see." Photos 12 and 13 show the rocket leaving the launch tower.

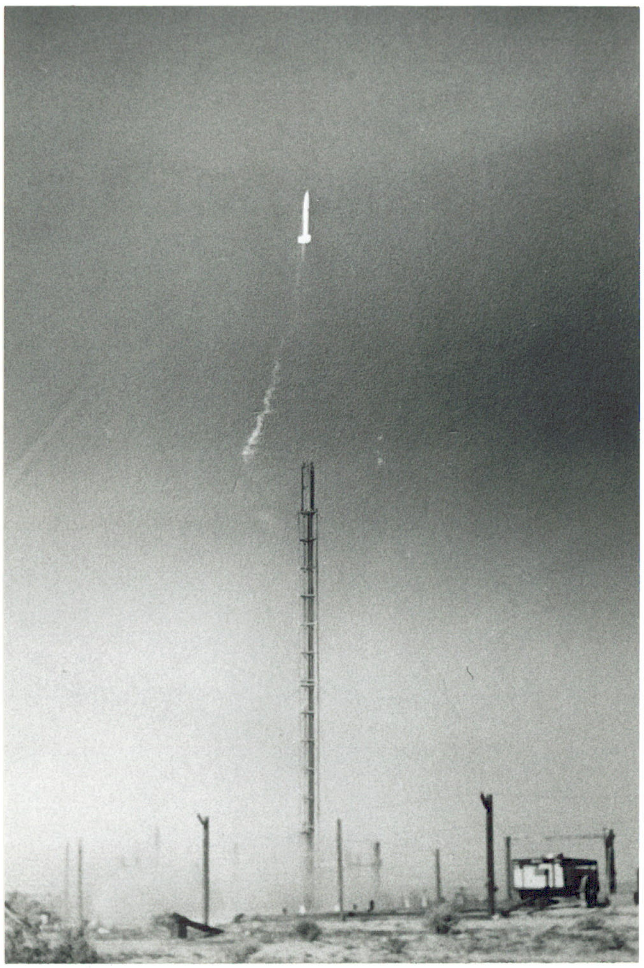


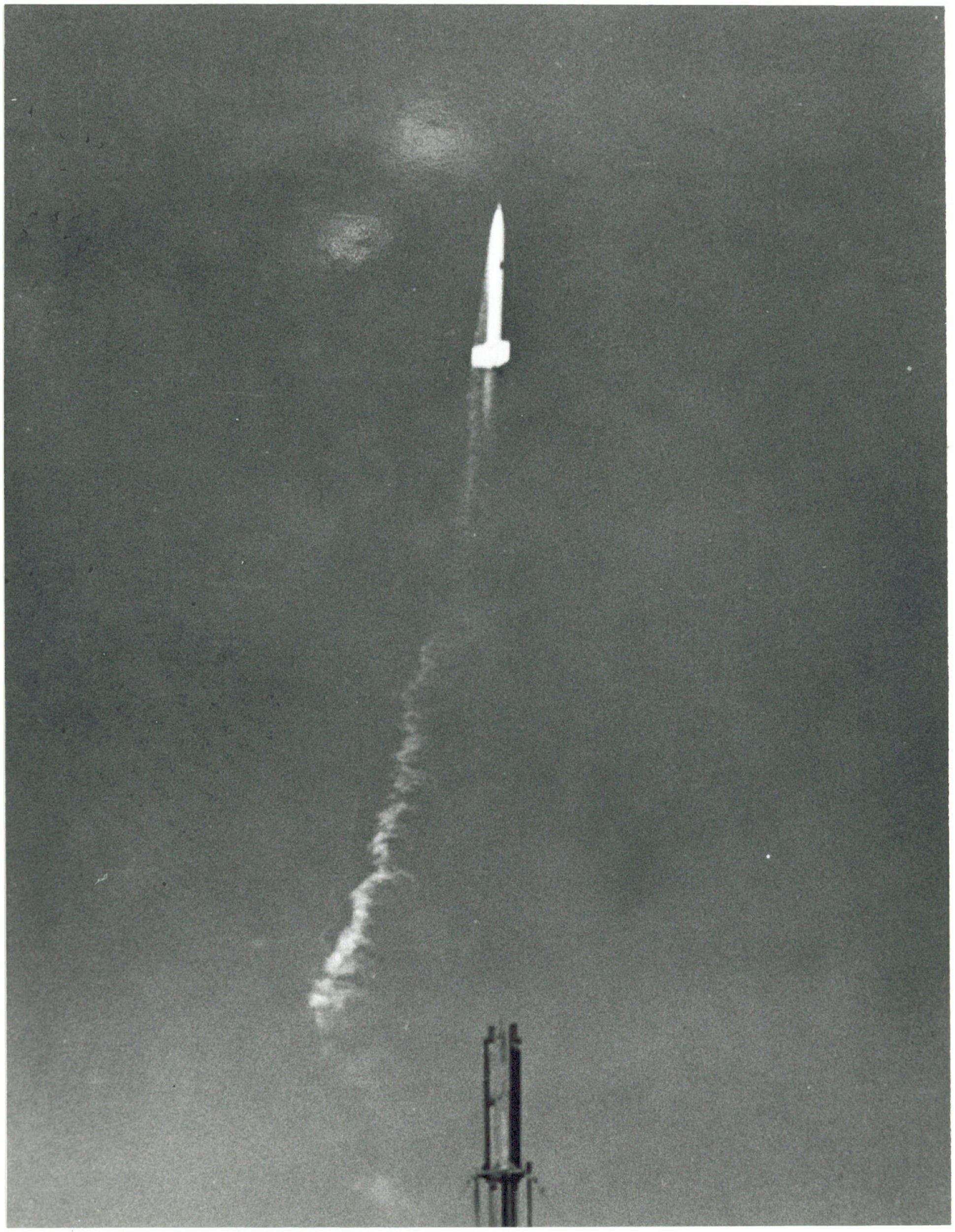
Photo 12 & 13(next page) - The rocket is seen leaving the launch tower on the morning of 14 May, 1950. Tracking was made difficult by the improperly functioning smoke flare which is seen here giving off only a thin trail of vapor.

Although the phototheodolite operator lost the rocket because the smoke flare had failed to function properly, the movie camera data for the first several seconds of the flight allowed the calculation of velocity, position, and direction of motion. This information and the static test data gave Rosenthal and Elliott the capability to determine, step by step, the following;

Velocity at burnout	1,460 feet/sec
Altitude at burnout	9,800 feet
Maximum altitude	23,500 feet
Range	41,000 feet
Total flight time	84 seconds

Elliott and Rosenthal looked for the rocket that day and on a few subsequent occasions. It was never found and probably lies buried in the desert somewhere north of Soledad Mountain just south of the town of Mojave, California. But finding the rocket, or how fast it went, or what altitude it attained was really not very important. It was rather the journey and not the destination that was of the most value.

The significance of the project was not in its technical achievement. No new and revolutionary principles were discovered. Rather it is the story of two incredibly bright young minds that epitomized the hopefulness of their times. Before the onslaught of liability lawyers, before the environmental doomsayers (who daily invent new scenarios for the impending catastrophes that they are convinced will lead to the obliteration of life on this planet), before we became a nation afraid of all the wonders our technology had wrought, there were these bright, hopeful minds. In our more modern and enlightened world of the 1990's, our children are trained to sit mindlessly in front of a television while being entertained by mutant turtles. Reading is slowly becoming a lost art. The technical advances of the 1950's and 1960's are no longer an inspiration, but are now equated with the destruction of the "ecosystem". But there are still bright minds and dreamers full of enthusiasm to learn and build and strive. They invent and improve and advance our knowledge each day because they do not understand that it cannot be done. General Abramson, who was the head of the Strategic Defense Office in the 1980's



was asked once if it upset him when the news media reported that most of what his organization was working on was "impossible". He laughed, said no, and then explained. "The quickest way for us to make progress is for the news media to tell a group of American engineers that what they are doing is not possible." I, for one, am unwilling to accept the premise that the best and brightest days of the United States are behind her. And her greatest hope is in the spirit of those with inspired vision. The story of this project inspired me in my youth and it continues to do so even to this day. It is my sincere hope that it will inspire others as well.

Inconsistency between Static Test Data and the Known Flight Performance of Zn/S Rockets

by
Bill Claybaugh

Recent static test results on the Beta rocket design, when incorporated in trajectory prediction programs, produce results which are inconsistent with measured flight performance.

The test data reconfirm the effective burning rate of this propellant to be in the range of 90-92 in/sec., and provide a measured ground test specific impulse of 40 seconds. The test results also confirm visual observations that the Beta nozzle is underexpanded: measured thrust was significantly in excess of the value expected if the nozzle was expanding to sea level pressure.

Assuming actual atmospheric pressure of 14.0 psia at the test location, it is possible, using an iterative procedure, to determine the pressure ratio required to produce the measured peak thrust, given the known throat area and area ratio and an estimate of the specific heat ratio (1.25) and thus determine the peak chamber pressure. This procedure indicates that peak chamber pressure in the full length Beta reached about 1730 psia, with a pressure ratio of about 60, and an exit pressure near 29 psia. Sensitivity analysis shows that these values are within 2% of the above result for actual test atmospheric pressures in the range of 14.2 to 13.8 psia—consistent with the altitude of the test location—and for values of the specific heat ratio between 1.20 and 1.30.

Modeling of the performance data measured during the static test in two different sounding rocket trajectory programs (ALT and Ascent) fails to duplicate the known flight performance of the Beta rocket. Both programs predict, on the basis of the measured thrust curve, a burnout velocity of about 850 ft./sec. and a burnout altitude of about 200 feet. Maximum altitude is estimated at 10,000 feet, with total flight times around 50 seconds.
Known burnout altitude is in the range of 150-200 feet, measured maximum altitudes are in the range of 5000-6000 feet, and actual flight times are typically around 40 seconds. Since both programs are consistent and one (ALT) has been validated against model, high power, and professional rockets, and since variation of other parameters within the trajectory models fails to bring performance into line with experience, the discrepancy may be due to the flight performance of the propellant being significantly lower than the measured static test performance. Static testing of a Beta motor in a nozzle down configuration may show lower performance due to entrainment and loss of unburned propellant through the nozzle. Further performance loss may occur in flight due to loss of unburned propellant through the nozzle from the effects of high acceleration.

The planned flight of the Gamma II, two stage, 3 inch outside diameter S/Zn rocket carrying an altitude recorder may provide an opportunity to test the above thesis. The Gamma rocket has about 2.36 times the burning surface of a Beta, and a larger throat area (.785 sq. in. versus .636 sq. in.). Using the estimate of 1750 psia for the Beta peak pressure and adjusting for the larger burning surface and larger throat yields an estimated peak pressure for the Gamma rocket of 3350 psia. This pressure is about one-third of the theoretical yield pressure for a 4130 steel tube of the above dimension and a .065" wall. Adjusting for weld factors, it is reasonable to suggest that this rocket is operating near the maximum practical pressure. This suggestion is supported by experience showing that the single stage Gamma rocket flown with a Beta dimensioned nozzle bursts at about 30 feet altitude. For the Gamma nozzle, this peak chamber pressure implies a peak exit pressure of about 55 psia.

Based on the above adjustment of the static test results, it is possible to estimate the performance of the Gamma rocket if S/Zn propellant performs in flight as measured on the ground. Burning Time should be about .6 seconds, Average Thrust about 2000 pounds, and Specific Impulse about 40 seconds. Using these values, the ALT and Ascent programs predict single stage burnout altitude of about 350 feet at about 1200 ft/sec. and a maximum altitude of 21,000 feet. As with the Beta, these values are inconsistent with experience. Measured burnout altitudes are in the range of 200-225 feet, estimated maximum altitudes are about 13,000 feet.

The programs calculate that the two stage Gamma II will burnout at about 700 feet at around 1550 ft./sec. and reach a maximum altitude of around 30,000 feet if the static test results reflect flight performance.

In order to bring the calculated single stage Beta and Gamma rockets' flight performance into line with experience it is necessary to assume that actual flight specific impulse is in the range of 28-29 seconds.

Making this adjustment for the Gamma II vehicle produces an estimated burnout altitude of 550 feet at about 1200 ft./sec. and an estimated maximum altitude of about 20,000 feet. Flight instrumentation for this vehicle consists of a digital altimeter with recovery initiation system (Adept Rocketry part number OBC2B) and an independent digital recovery system initiator (Adept part number ALTS2A), both modified for high acceleration.

The digital altimeter will record altitude (i.e., atmospheric pressure) every 1/10th second for the first forty seconds of the flight, which will include peak altitude even if the vehicle reaches the 30,000 foot level indicated by the static test results. Because the recovery initiator also records peak altitude (only) it may provide backup confirmation for that value. Since the key dimensions of the two stages of the Gamma II vehicle are nearly identical (The first stage combustion chamber is 3" shorter than the upper stage to allow the upper stage nozzle to friction fit the inside diameter of the first stage motor, otherwise all dimensions effecting rocket performance are the same.) it may be possible to gather sufficient acceleration data during the burn (Estimated total burning time is less than 1.2 seconds, implying a maximum of 12 data points, six for each stage.) to estimate the in flight specific impulse directly. In any case, a successful flight should provide absolute altitude data sufficient to model the drag characteristics of the vehicle post burnout and thus to estimate the flight average specific impulse by trajectory modeling.

An additional potential test of this issue might be to fly two identical vehicles, launched at the same time, but fueled by the conventional (poured and packed) method and some other technique designed to prevent loss of propellant during acceleration (for example, by packing the S/Zn propellant in a small number (4-8) of tissue faced heavy paper tubes, or by casting the propellant in such tubes with a high vapor pressure Sulfur solvent [e.g.; acetone] to provide a series of monolithic blocks, or possibly by mechanical compression of the propellant in a fixture to produce monolithic blocks). Relatively large (Beta class or better) vehicles might be required in order to see any difference in performance based on ground observation. Alternatively, each vehicle might carry a digital altimeter and recovery system.

CAVITATING VENTURIS PART 2: FABRICATION AND CALIBRATION

By Tom Mueller

In the first installment of this article I described how a cavitating venturi works and how it can be used to measure and control the propellant flow rate to a liquid rocket engine. In this article I will describe how to machine and flow calibrate a venturi to meet a particular need.

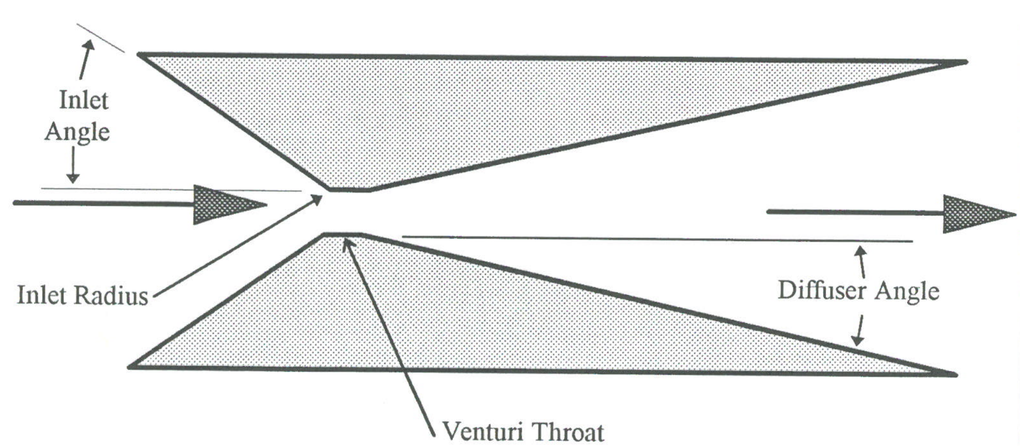
Figure 1 represents a schematic of a venturi. Some important considerations when making a venturi are the throat diameter and length, inlet geometry, diffuser length and angle, and surface finish. Another important consideration is the material from which the cavitating venturi is made. I usually use 6061 aluminum simply because it is easy to machine and finish and is compatible with most propellants. Another good material to use is brass. Stainless steel is ideal from a durability and propellant compatibility standpoint, but is difficult to machine.

Figure 2 shows some dimensions for a typical cavitating venturi. This venturi is intended to slip into a flared tube and seat against an AN fitting. This is a very convenient way of installing a venturi into a propellant run line. The inlet angle is 37°, which is the flare angle of a AN fitting The outside diameters of the venturi are sized to allow it to fit into the tube and match up to the male flare fitting.

As Described in the previous article, the effective throat area is what controls the flow rate of propellant to the engine. Here, "effective" means the geometric area of the throat times the flow coefficient or Cd of the throat. This effective area, the CdA, is what is measured during water flow calibrations of the venturi. The Cd of the venturi varies from about 0.8 to 0.95, depending on several factors. The inlet angle to the venturi and the radius at the inlet to throat are two of the biggest factors. The length of the throat can also have an effect on the Cd. I have found that the Cd of a venturi with zero inlet radius and a 30 to 37° inlet angle is about 0.85. If the inlet to the throat is generously radiused, the Cd can be as high as about 0.95. Increasing the inlet radius can be used to calibrate a venturi to give a desired CdA.

The most difficult part of machining a cavitating venturi is making the low angle diffuser. For large venturis, I have used a 7° half angle end mill as a form tool. These end mills are designed to cut the draft angle on molds, and are available from tool suppliers such as Rutland Tool. The tip diameter of the particular end mill I have is 0.070 inch. If I need to make a venturi with a smaller throat than that, I have used high speed steel burrs that have a 7° half angle and have a tip diameter down to 0.020 inch. A form tool can also be made from steel by turning it in the shape of a cone, milling it into a triangular blade, heat treating it for hardness, and then grinding

Figure 1. Cavitating Venturi Schematic



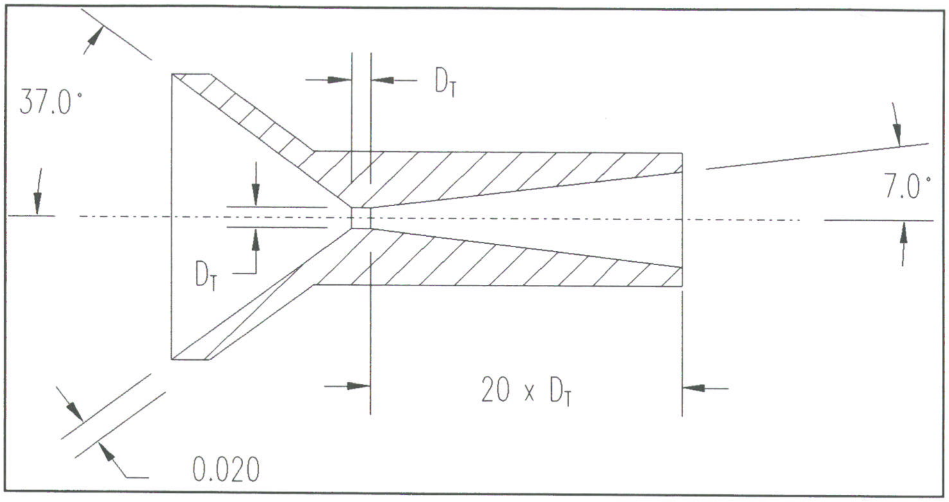


Figure 2. Typical Cavitating venturi Geometry

the cutting edges. In any case, use various drill diameters to get the diffuser roughed out to size in steps, being careful that the drill step does not exceed the finish dimensions of the diffuser cone. I use a CAD program to determine the depth to drill each drill size, leaving a minimum of at least 0.010 inch of material on the radius from the finish contour. Drill the rough contour on a lathe, using the tailstock vernier to determine the drill depth. After the diffuser is roughed out, use the tapered end mill or burr to cut the final contour, again using the tailstock vernier to control the total depth of feed. Use low speeds to prevent chatter during this operation, and retract the tool often to remove the build up of chips, especially if a high speed burr is used. Use a light oil (I use WD-40) as a lubricant.

The surface finish of the diffuser will not be very good after the cutting operation, particularly after using a burr as a form tool. To get good recovery with a cavitating venturi, it is important that the diffuser be as smooth as possible. I use metal finishing sandpaper along with WD-40 as a wet sanding agent to smooth up the contour on the lathe. Use a toothpick and a small strip of the sand paper to smooth the inside contour of the diffuser, starting with 400 grit, then 600 and 1000. I obtain a final polish using a paper towel wetted with polishing compound, formed into the shape of the diffuser by twisting it.

The inlet angle of the venturi can be rough machined with a 60° countersink (30° half angle), then cut to the final angle using a small boring bar. On small venturis, it is difficult to cut the region nearest to the throat using a boring bar, so I sometimes use a 60° countersink to finish the inlet angle, blending it into the 37° angle that mates with the AN fitting. The throat length must be no longer than the throat diameter for good recovery, so be careful with tolerances to ensure the proper length of the throat. Keep the throat one or two drill sizes smaller than the final size until the inlet and outlet angles are finished to prevent the throat from getting sanded oversize.

If the venturi CdA needs to be "matched" to a certain value, I leave the inlet radius to the throat sharp, so it can be lightly rounded during water flows to adjust the CdA. This needs to be done, for example, if both a fuel and oxidizer venturi are being made and must be matched to give the proper flows at a given tank pressure. If a single venturi is being used (i.e., a hybrid rocket engine), then the inlet radius can be rounded generously by light sanding on the lathe to give a high Cd. The tank pressure of the rocket is then adjusted to give the correct flow at whatever the measured CdA is.

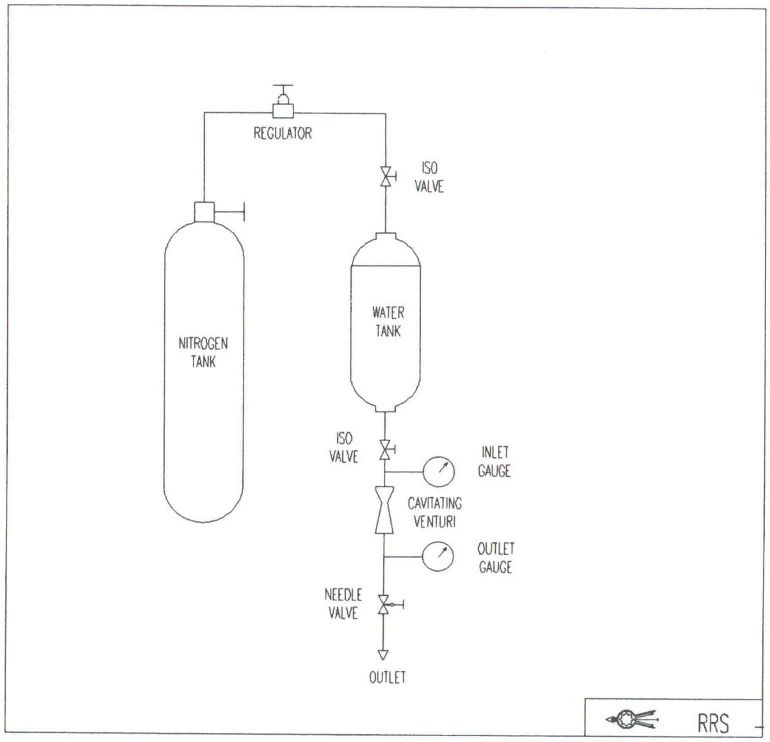


Figure 3. Water flow test setup

Figure 3 shows a typical setup to flow calibrate a venturi. It may be most convenient to use the actual propellant tank as the water flow supply tank. The water flows should be performed over a range of inlet pressures in order to determine the dependence of the venturi CdA on pressure. The water flow rate is measured by the "catch and weigh" technique in which the water is flowed into a container for a time measured by a stopwatch, and then the contents of the container are weighed on an accurate scale. To determine the mass flow rate, simply divide the mass flowed by the time. The CdA of the venturi is determined by :

$$CdA = \frac{\dot{W}_{H2O}}{\sqrt{2\rho g(P_{in} - P_v)}}$$

Where: $\dot{W}_{H2O} =$ Mass flow rate of Water, $\frac{lbm}{\sec}$

$\rho = 0.0361\frac{lbm}{in^3}$, density of water @ 60°F

$P_{in} =$ inlet Pressure to venturi , psia

$P_V = 0 psia$, vapor pressure of water

$g = 386.1\frac{lbm - in}{lbf - \sec^2}$, gravitational constant

In order to increase the CdA (the amount of flow for a given inlet pressure), round the inlet radius by lightly lapping it with a small wood dowel that has been sharpened in a pencil sharpener and soaked in polishing compound. The soft wood will conform to the inlet, and the polishing compound will remove material. After lapping the inlet in this manner for a few minutes, reflow the venturi to determine the new CdA. Be careful to not round the inlet too much, or you will "overshoot" your desired CdA. If this happens, you can reduce the CdA by machining 5 or ten mils off the inlet angle to get a sharp inlet radius. If you find that even with generous lapping you are not getting a high enough CdA, then drill the throat to the next larger drill size and repeat the water flow and lapping procedure.

The recovery of a venturi can be determined by slowly closing the needle valve downstream of the venturi to raise the back pressure until the venturi "drops out" of cavitation. This is usually done by using a flowmeter to measure the water flow rate. As the backpressure is raised, the flow rate will remain constant until the venturi loses cavitation, at which point the flow rate will start to drop. The recovery pressure is the pressure where the venturi drops out of cavitation, and the venturi recovery is the recovery pressure divided by the inlet pressure, expressed as a percent. Remember that the pressures used to determine the CdA of the venturi and the recovery are absolute pressures, so 15 psi must be added to the indicated gauge pressures. Even with out a flowmeter the recovery pressure of a venturi can be determined by watching the needle of the outlet gauge. Usually the needle will start to bounce, sometimes violently, as the recovery pressure is encountered. Bring the backpressure up until the needle begins to oscillate, and record this as the recovery pressure. A good cavitating venturi will have a recovery of 75% to 85%.

The venturi is installed in the run line between the tank and the injector, with a pressure measurement made upstream of the venturi. If the propellant is cryogenic, such as LOX, special precautions must be taken. In this case, the temperature of the propellant at the venturi must be controlled, because the vapor pressure and density of the LOX changes dramatically with temperature. During startup, propellant warming through the lines can cause a "vapor lock" at the venturi, resulting in a low LOX flow rate and poor engine operation, possibly even no ignition. The best way to avoid this problem is to locate the venturi as close to the tank as possible, so the venturi is cooled by the LOX as the tank is filled and during the time prior to engine firing. The LOX tank should be vented until just prior to the engine firing so that the LOX is maintained at the normal boiling point and the vapor pressure is known to be about 15 psia. Under this scenario, the venturi is designed for LOX properties at the normal boiling point and no temperature measurement at the venturi is required. In all cases, whether the propellant is cryogenic or not, the pressure at the inlet to the venturi should be measured in order to determine the propellant flow rate through the venturi. The CdA of the venturi will be constant within the ability to measure regardless of the propellant used. If the propellant is cryogenic, then a small reduction in the CdA will be caused by thermal shrinkage of the throat. For example, an aluminum venturi used with LOX (chilled from 70°F to -300°F) will have a throat area decrease of about 1%.

Well, that about covers my limited knowledge of cavitating venturis. In the next newsletter I will describe a design for a pyrotechnic squib valve that I have used on several occasions.

TRACKING IMPROVEMENTS FOR THE MTA

For those of you who have not been out to the MTA lately, or even for those of us who are usually so busy in the compound that we never make it out to tracking, there have been some recent improvements. A new tracking stand has been built and permanently installed 1000 feet side range from the launch towers. Photo 1 shows the stand in use during one of the latest firings. The addition of guard rails, an access ladder, and a paint job will complete the effort.

Photo 1 - The new tracking stand in use at the MTA.

Photo 2 shows a sophisticated piece of tracking equipment built by Bill Colburn of the Northern RRS chapter. It is basically a phototheodolite built around a video camera and is comprised of six units; the video camera (with superimposed tilting and stop watch reading in minutes, seconds, and tenths of seconds), the Angular Elevation Unit (based on a Smart Level Series, Pro Digital Readout Level Module. This unit is accurate to one tenth of a degree and was coupled with a 1" x 2" mirror and a 2.5" diameter f2 lens for optical insertion of elevation readout on the video frame). The unit is mounted on a large tripod and uses the standard slip joints for azimuth and elevation movement. The video recorder, 12 volt power supply (car battery), and a shade umbrella complete the unit. Azimuth information is recorded on the audio track by having a second operator read the values aloud as the primary operator tracks the rocket.

This equipment was brought out to the MTA for the 23 April, 1994 firing. It was used to track four rockets and several recomendations for improvements were made by the operators after the firing. We hope to see Bill out there again at the next firing with the new and improved version of this sophisticated tracking and data collection aid.

Photo 2 - The video phototheodolite built by Bill Colburn and used at the 23 April, 1994 firing.

BITS & PIECES

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 last news letter, 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 copies have been included in this issue. These forms are for solid propellant as well as liquid / hybrid systems and can be used for either static or flight testing.

Anyone intending to test any type of system 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 any type of system needs to keep a copy of these forms 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.

STANDARD RECORD FORM
LIQUIDS & HYBRIDS

☐ Static
☐ Flight

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

7/94

Owner's Name: ______________________ Date: ______________

Address: ______________________ Affiliation: ______________

City: ____________ State: ________ Zip: ______________

Phone: ____________ FAX: ________ Date Ready to Test: ________

PHYSICAL DATA:

Est. Altitude ______________

Fuel ____________ Oxidizer ________ Chamber Press. ____________

Fuel Vol. ____________ Oxidizer Vol. ________ Regulated Press. ____________

Designed Thrust ____________ Burn Time. ________ Inj. Type ________

Purpose of Test ____

____

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 ______________________

Recorded Thrust ____________ Burn Time. ________ Inj. Type ________

Wt. Fuel ______________________ Total Wt. Flying ______________________

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 ______________________

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 ______________

DECALS: Included with this issue is a complimentary copy of a new RRS logo decal recently produced. These can be applied to rockets, note books, or even foreheads for those really gung ho types. Additional decals can be ordered from the Society for 25 cents each.

NORTHERN CHAPTER NEWS: Congratulations to our Northern Chapter on their excellent news letter. Volume 1, Number 1 came out in September and covered several areas of interest. The timing was not good this time around, but in the future we will reprint selected articles for the benefit of the entire RRS membership. Articles for the Northern Chapter news letter may be submitted to Walt Rosenberg, 3090 Balmoral Drive, San Jose, CA 95132.

BACK ISSUES OF HIGH POWER ROCKETRY: Articles about the RRS, experimental rocketry, and what goes on at the MTA have recently been published in *High Power Rocketry* magazine. The editor, Mr. Bruce Kelly, has taken inputs submitted to him and has done an outstanding job of producing full color articles as part of the experimental rocketry section of this pubublication. Back issues circled here contain articles about RRS activities and can be ordered per the instructions on the enclosed ad.

Although geared specifically toward the high power model builders, the magazine contains much information of interest and utility to RRS members. "How to" instructions for composite material construction, flight instrumentation, electronic components and equipment, recovery system designs, and active guidance information are just a few of the areas covered. You do not have to be a TRIPOLI member to subscribe to the magazine and I would enthusiastically recommend a subscription to anyone interested in any type of rocketry. The current subscription rate is $25 per year. There are two more issues (other than those shown in this ad) that include articles about RRS activities and there will be more to follow.

MEMBERSHIP ROSTER: Included with this news letter is a copy of the latest membership roster. If your address or telephone number information is incorrect, please send corrections to Mr. George Dosa, 18011 South Curt Pl., Gardena, CA 90248. This will insure everyone gets their news letters and other Society mailings. Correct telephone numbers are also a great help when other members want to contact you. Thanks.