

Chapter Six

Gerard Elverum

TRW - Lunar Descent Engine



Gerard "Jerry" W. Elverum began working at California Institute of Technology's Jet Propulsion Laboratory (JPL) in 1949, where he spent ten years performing pioneering research and development on propellants and rocket propulsion. In 1959, he joined Space Technology Laboratories (later called TRW). In May 1963, Grumman and NASA selected his patented design concept for a deep-throttling liquid bi-propellant rocket engine for a backup development program for the descent engine of the Lunar Excursion Module (LEM). In December 1964, NASA committed to the Space Technology Laboratories design, and the first flight engine was delivered to Grumman in August 1966. Elverum was program director and chief engineer for the LEM descent engine throughout this time. He joined the American Rocket Society in 1951, received the American Institute of Aeronautics and Astronautics' (AIAA) James H. Wyld Propulsion Award in 1973, and was elected an AIAA Fellow in 1983. He was elected to the National Academy of Engineering in 1987. He retired as vice president and general manager of TRW's Applied Technology Division in 1990.

The rocket engine companies Rocketdyne and Aerojet were the big names back in the late 1950s and early 1960s; they got the Apollo astronauts into orbit around the moon. But, how in the world did it happen that those guys were sitting in the descent module of the Apollo Program on top of an engine that was developed by a few engineers at a space engineering company called Space Technology Laboratories (STL) instead of one of the big rocket companies?

In 1960, Space Technology Laboratories was a system engineering company that was doing engineering and technical direction on the Atlas, Titan, and Minuteman programs. But, at that time, they decided they also had to get involved in space technology because they were beginning to build military spacecraft. One of the things they decided was that in order to be able to make certain maneuvers in space, and to do some of the things they wanted to do with missiles, they were going to have to come with a deep-throttling rocket engine to perform those missions. At the time, I was the head of the Advanced Propulsion Development Department at STL. They came to me and said, "We need a 20:1 throttling engine. It has to be storable. It's going to be pressure-fed. It has to be stable, and we need it in order to move into the next phase." So, I got that job. But, the story of the Lunar Excursion Module descent engine really starts back many years before that at Cal Tech's Jet Propulsion Laboratory (JPL).

At JPL, I was playing around with a pair of concentric capillary tubes. The reason I was doing that is that we had responsibility for developing the Corporal, which was the Army's intermediate, tactical, ballistic missile. At the time, they decided they had to lower the freezing point, so they went to something called stable fuming nitric acid (SFNA) from red fuming nitric acid (RFNA). The SFNA, we put a lot more NO_2 , and then, we put some water in. When we did that, the Corporal engine got very rough. It had all kinds of performance problems. They turned to me and said, "You get into your laboratory and see if you can figure out what's going on." Well, I concluded that the liquid phase reactions of hypergolic propellants were really controlling what was going in the combustion of that engine, and that when we put the water in, we stopped the early reactions that led to the distribution of the propellant from the injector we were using on the Corporal. As a result of that, they no longer were mixing the way they should. They were being separated, and then, they were detonating in certain parts of the engine, and it was causing it to be rough. That meant they had to change the injector considerably and back off on how we were doing it. (See Slide 7, Appendix H)

At JPL, I was working on the liquid phase reactions with concentric tubes, and we were able to demonstrate that these early, fast reactions would occur as fast as we could mix a propellant. (See Slide 8, Appendix H) There was basically zero activation energy that was driving the system. When you have zero activation energy reactions like that, you can separate the propellants, and that was giving us problems on the Corporal. When you put some force mixing into

it, you can get the reactions to go in less than a millisecond. (See Slide 6, Appendix H) But, in building that thing, it seemed like a very good idea that as long as we could get those reactions to go, we'd put that concentric tube together and see if we could make an injector out of it. If we had these early reactions going very rapidly, we didn't need distributed injection in order to be able to run the thing as a rocket engine. When we ran this with dinitrogen tetroxide (nitrogen tetroxide or nitrogen peroxide or N_2O_4) and hydrazine, we were unable to pull the inner tube back away from the face. As soon as we pulled it back in about fifty thousandths of an inch, the whole end of the tube would just detonate. That showed how fast those reactions could be because that was going off even when just a small percentage of the material was mixed. But by pushing that forward a little bit, we were able to run this engine, run it at very high combustion efficiency, and run it stably.

We were running it with N_2H_4 -hydrazine that everybody else was having terrible problems with, because it was very difficult to handle that combination from a stability point of view. When I went down to Space Technology Laboratories and they said, "Let's start a throttling rocket engine program," I determined that a fixed area injector with a separate flow control valve was one way you could throttle an engine. (See Slide 4, Appendix H) You could have a variable area injector - that was another way that you could throttle it - or you could have a variable area injector with a separate flow control valve. If you were going to vary the area of the injector, you had to be able to do that in a reasonable fashion. If you had 1,000 holes, trying to vary the area of those was going to be very difficult. But, there are problems with throttling with a fixed area injector and a separate flow control valve. If you back off on the flow rate and let the pressure drop, you need some minimum delta pressure across your injector at the low end. Then, that drives the delta pressure up so fast that you can't throttle 10:1. You just can't get there. It required tank pressures of thousands of pounds per square inch (psi), or if you start at 300 psi, you would have to have like one psi across the injector at the low end and that isn't going to work either.

The second way that you could talk about throttling an engine is with just a variable area injector. (See Slide 5, Appendix H) But, that has its own set of problems if you look at the tank pressure. Let's say we set it at 450, and we wanted a 300 psi chamber pressure. If you are going to throttle it just 10:1, this would be the chamber pressure, you get an enormous pressure drop across the injector, and the injector has to be very, very tiny. In small thrust engines, which we were working on at the time, 500-pound thrust, the dimensions around that throttling injector would be so small that we had no possibility of building it and still be in uniform flow. That was a major drawback to that type of system. But, if we go to a system that combines both a valve and a throttling injector, then we have an entirely different kind of process going on in an engine.

If you have a 300 psi chamber pressure, you are going to throttle it down to thirty psi at 10 percent thrust. You would like to be able to set the delta pressure across the oxidizer and the fuel in the injector at every part of this throttling diagram to be optimum in order to get high performance and to become stable. So, you'd like to be able to do that independently. One way to do that is to look at the pressure drop across the valve; you can go to cavitating flow. If you drop the delta pressure below the inlet to the valve below a given number, it will go into cavitation and at that point, we can set the delta pressures across the oxidizer and fuel to be any value that we want for purposes of combustion. That will totally isolate the problem of controlling the flow rate from the problems of optimizing how the injector will work. So, the one fundamental principle that came out of this was – separate the functional things whenever you can, so that you are free to optimize the function in this case, controlling the flow rates and the mixture ratio to an exact number from the functions and the processes of optimizing the injector. Then, we had what is called a cavitating venturi valve, and we designed it to be a throttling cavitating venturi. Then, we had it linked 1:1 with a single element injector in the middle of the chamber, which was optimized as far as pressure drop across the oxidizer and fuel at every flow rate to maximize the performance in the chamber. By separating these basic functions, we were able to spend our time optimizing the injector for the chamber and the mixture ratio control and the flow rate. So, over this total complex duty cycle of the Apollo descent, we knew we would have absolute control of the mixture ratio, no matter what profile they decided to run.

In a nutshell, this became the basic concept of the Lunar Excursion Module (LEM) descent engine. The injector has a sliding sleeve that controls both the oxidizer in the middle and the fuel coming down the outside. That was linked 1:1 with two cavitating venturi valves, so that for every single flow rate that was going through those cavitating venturi valves, there was an optimized delta pressure and injector configuration, totally linked together. (See Slide 9, Appendix H)

In 1962, we heard that the Apollo Program was probably going to the descent stage concept, and Rocketdyne had been given the role of building the descent engine because they were the engine guys in the United States at the time. But, we also heard that NASA was getting concerned about the basic throttling engine design and were going to go to a backup program. They wanted somebody to do a backup. Well, I took this diagram to NASA and said, “We want to bid on the Apollo engine.” They said, “Who are you, and why are you in our office?” I came back with some data on this engine. It showed that we were using hydrazine and N_2O_4 , which nobody else had been able to run effectively. We showed them that we had throttled it over not 10:1 but 20:1, that we'd kept a flat performance profile over the whole 20:1 throttling range, and that we never had any indication of instability or any other behavior. So, they gave us a Request for Proposal (RFP) and said, “Okay, we'll let you bid on the backup.”

We had to scale that up by a factor of ten because that turned out to be the biggest thing we could put in our test stand. We only had a small test stand in the middle of Inglewood, California, at that time. (See Slide 13, Appendix H) We scaled it up to 5,000 pounds from the 500-pound engine. We built an ablative line chamber, so that it would have to be uncooled and ran that at 5,000 pounds of thrust. Everybody was saying, "Okay, it's going to go unstable and that will put you out of business." It didn't go unstable. So, that gave NASA and the Grumman Aircraft Engineering Corporation, who was the prime contractor, a problem because now we were still in the middle of competing, and they weren't sure they wanted a little outfit like us competing. But there we were with this engine running and giving good performance and throttling 10:1 in our facility. (See Slide 13, Appendix H)

The competitors for the backup engine were Reaction Motors Inc., a liquid rocket engine company that had done the Vanguard engine, and Aerojet. One of the things they said was: "It's okay to have a 5,000-pound engine, but the LEM is going to have to be 10,000 pounds." That meant it was going to be this diameter, and if it was that diameter, it was going to go unstable and that's it. So, they said, "You've got to demonstrate with a bomb, in the right diameter chamber and show that you are going to be stable." Well, it took us a while to build that big diameter chamber, and it turned out that by the time we got it all set up on the test stand, NASA said, "We're coming out." Grumman said, "We're coming out." It was a Saturday morning, and we had never been able to fire that engine in order to see whether it was going to work. We used steam-blow in the stand in order to be able to run 10:1, and run full on the 2:1 nozzle that we were using.

We built what we called the "iron pig." (See Slides 14 and 15, Appendix H) It was seventeen inches in diameter. We ran the 5,000-pound injector into it and filled it up with bombs around in various different locations. I had to fire the first one of those tests in front of NASA and in front of Grumman, because we had just barely got it on the stand Saturday morning. Obviously, my management was not too happy with the idea of firing for the first time in front of the customer, but that was it; you either go or you didn't go. Well, we fired it, and it was completely stable. At 100 psi, we fired a bomb, went up to 210 psi. Within about ten milliseconds, we had immediately stabilized the engine at that size. Well, as a result of this test, and as a result of the fact that we were throttling 10:1 with good performance, they gave us the backup contract. It took us about a year and one-half of very intense competition with Rocketdyne, but at the end of that time, NASA came back to us and said, "You're now on for being the descent engine contractor." Well, what we had to do was promise that we would put a whole facility together at San Juan Capistrano, and we built a whole series of test stands there. (See Slides 16 and 17, Appendix H)

Armstrong finally set it down with less than a fifteen-second margin. If we had gone fifteen seconds more, he would have had to abort

For the duty cycle for the LEM, we started out at 10 percent each time. They wanted to start the duty cycle at 10 percent in order to get the vehicle stabilized. We started at 10 percent for about thirty seconds, and then kicked it up for about six or seven seconds. This was the duty cycle in order to go from the orbit they were in, into the orbit that was required to start the descent down to the surface of the moon. Then, we waited for about one hour and had the restart. We restarted again at 10 percent for about thirty seconds to stabilize all their stuff. Then, we went up to full thrust. This was the braking phase coming along on the descent. Then, we dropped down to what we called the “flare out.” When we dropped from the full thrust position, we went from a calibrated full-thrust engine into the cavitating venturi region of the valves. From that point on, we could do whatever we wanted in this duty cycle. We never knew exactly what they were going to do, particularly once they got into this phase. (See Slide 18, Appendix H)

It turned out that the most critical mission we ever ran was Apollo 11. That was the most difficult landing. It exercised the engine to the absolute maximum in terms of the ablative liners capability, fuel control, and everything else. One of the things we had to do was to be able to set down on the moon and land on top of a rock. It turned out, like everything else, NASA came up with a specification for the rock this engine would have to land on and for the way the nozzle would have to crush on it. When I was sitting at Grumman during that first landing, the first thing that Neil Armstrong did when it came down and he found out they were over a boulder field, was to begin hunting around to figure out where he could find a NASA specification rock on which to land. That’s because we had demonstrated the nozzle would crush okay on a rock of the specified size. At least, that’s the way I felt back at Grumman. So, Armstrong kept looking and looking and looking, and I said, “Find the damn rock and set it down.” We were facing problems with propellant getting ready to run out. Armstrong finally set it down with less than a fifteen-second margin. If we had gone fifteen seconds more, he would have had to abort, but Armstrong was not about to abort that flight, so he went clear to the end. We went almost to the absolute maximum of everything we could do, and I was probably the only guy in the world that knew where all these margins were and how close we were getting to them. They talked about a few guys at Houston getting blue when they finally set down. I told them, “You’ve got one back here at Grumman who was blue a long time ago.”

The overall landing mission duty cycle involved 60 percent and 10 percent hovering modes. Talking about the descent engine, the thrust chamber design had a titanium case lined with ablative material out to a nozzle expansion ratio of 16:1. Then, we had a columbium nozzle extension from there out to 48:1. We used the Wah Chang Company to get the columbium nozzle and get the coating done. Those nozzles were very successful, but they had to be very

thin. That nozzle had to be like seven-thousandths of an inch at the exit cone, and that was out about five feet in diameter. So, we stabilized it with a single flat ring in order to keep its shape. But, the reason it had to be that thin was that, otherwise, it could have kicked the whole descent stage over if it had hit some of these boulders at whatever angle. It was very successful that we were able to make that columbium nozzle work. (See Slide 19, Appendix H)

If you look at a cutaway of the final descent engine injector, you can see how the central element injector, the oxidizer, came down the middle and deflected off of the columbium tip. The fuel came down the outside of the tube, and a single moving part allowed us to change both the area of the oxidizer and the area of the fuel with a single-sleeve, driven and mechanically linked up 1:1 with the two cavitating venturi valves. We used a barrier coolant in the engine because it was ablatively lined. This engine had to run for 1,000 seconds ablatively, which is a long, long time if you are using ablative materials. We barrier cooled this just enough to keep that 1,000-second capability on the engine, and it cost us a little bit of performance. We lost about 1.5 percent of overall performance by doing that, but it allowed us to run the engine for up to 1,000 seconds. (See Slide 21, Appendix H)

When it came to the flow control valve, it was the first time I know of that anybody built a controllable cavitating venturi control valve. The reason I decided to go with a cavitating venturi on the engine was that I had used them at JPL, trying to run fluorine and hydrazine, and chlorine trifluoride and hydrazine. If you know anything about chlorine trifluoride and fluorine and hydrazine, you are never sure what's going to happen in the chamber. We would put fixed, cavitating venturis in the line in order to be sure that we were getting the right propellant flow at the right mixture ratio. We then could study what was really happening in the chamber without having what was happening in the chamber change the mixture ratio and change the flow rates all over the place. When we needed a control valve, we said there's no reason we can't take a cavitating venturi and control the area of it, which would give us whatever throttle profile we needed over the entire throttle range. That was the valve that was put on the descent engine. On the head end, there were cavitating valves on each side. The actuator was an electrical actuator done by Bendix Commercial Vehicle Systems. (See Slide 22, Appendix H)

The upper end of the valves was tied to a cross-arm, which was set on the pivot. The other end of the cross-arm goes to drive the sleeve on the center element of the injector. We went to quad ball valves on the fuel and oxidizer sides in order to assure that we had a positive chance for a shutdown or startup. The injector was spring loaded against the cross-arm, so that if we lost any electric power, the engine automatically would go to full-thrust position. That was

important, because if we lost electric power on that actuator during the descent, the astronauts had to be able to go back up into orbit. The engine had to be able to go back up to full thrust where they could then turn around and go back up and, hopefully, get clear back into orbit, so they could tie up with the command module again. That was a spring-loaded system where the actuator had to drive it down to the low-thrust position. (See Slide 23, Appendix H)

The lightweight chamber liner evolved over a whole series of optimizations. We ended up with oriented silica fabric on the inside. These were oriented about sixty degrees to the flow. Then, we had a lightweight metal felt material as an insulator outside of that in the chamber, and that had a titanium structural shield around it. Then, we had a radiation exit heat shield because of the way this engine was buried down underneath the landing system where it was only the bottom part of the nozzle that really stuck out there. We had to insulate that. (See Slide 24, Appendix H)

This was a major development problem and a major lesson that we learned. When we had this thing almost finished and designed, the people who made the silica fabric in the United States decided they didn't want to have anything more to do with it. Unbeknownst to us, they got their last material from some French company, and when we got that material and ran an ablative liner, everything came to pieces. The silica, even though it was supposed to be the same spec material as we had been developing, did not, in fact, perform the way the American silica fabric worked. It took us a long time to be able to sort that out and to get back to a supplier that would provide a material that was okay. The qualification specifications on something like those composite materials were extremely sensitive, and we had to be able to watch every single detail. A little thing like somebody deciding to get their fibers from someplace else could totally change the characteristic, and that happened to us on this engine development.

The "all-up" engine had a 48:1 exit cone. The injector was in the center, and the cavitating valves sat on the outside of it. We had a square gimbal and had to gimbal the engine about six degrees. We gimballed at the throat, not up at the head end. The gimbal worked very well. We never really had any problem with our gimbal design. (See Slide 25, Appendix H)

We did combustion stability tests; thirty-one bomb tests were conducted with five- to fifty-grain charges. We induced spikes from up to 175 percent of chamber pressure in all those tests. We had to bomb it at 10 percent, 25 percent, 50 percent and 100 percent thrust. The recovery was always less than ten milliseconds. We would give it a spike, and it would damp out immediately. We would recover in every test. At low thrust, it would take longer to recover as the pressure was much lower. But, we never sustained a high-frequency instability. In all of

the engines and all of the tests throughout development and flight, we had never had the case of combustion instability in any of the central pintle injectors. As far as I know, that hasn't been the case for any other engine. In some engines, people put in baffles. They knew that we had to put them in on the Atlas engine when it went unstable on Atlas 29 and Atlas 30 in Florida and destroyed the launch stand. Those instabilities came out of the blue when people thought we had it understood. We went to baffles in the Atlas engine, but that's a damping system; it is not a dynamically stable system. It prevents the instability from growing by damping it out, and if you can damp out the disturbance, then you have to wait until it might get a new disturbance. Then, it would damp that out, but you won't racetrack it around and cut the thrust chamber off the injector. In all of these other types of engines, they either go to baffles or they go to acoustic chambers, which is another method of damping. But in the LEM engine, we have never been able to sustain an instability in that engine. (See Slides 26 and 27, Appendix H)

In my opinion, the reason is that this was what we sold to NASA and Grumman after we did the iron pig test. Why was it stable? Well, it was basically stable to the first tangential mode, because it is like trying to play a violin while your bow is bowing at the nodal point of the string. You can't make music that way. If you put the energy in at the nodal point of the tangential mode, you can't support a dynamic, unstable combustion. So, we had the central injection system operating in a stable mode, and the basic flame front was where most of the energy was being converted. If we bombed that engine, we drove that energy release zone in towards the nodal point and as soon as we drove it into the nodal point, it could not support that frequency anymore. It could not couple and drive the instability. We've done this at every size engine diameter you can imagine, up to three and one-half feet. In fact, there was one at NASA's John C. Stennis Space Center that was a 650,000-pound thrust engine operating with hydrogen and oxygen, which everybody said must go unstable. We did it at Edwards Air Force Base in California with a 250,000-pound thrust engine, where again we set off bombs, and it was stable. In every known case, both for the tangential and the radial, this injector configuration goes into dynamic stability. (See Slide 28, Appendix H)

Through March 1967, we had conducted more than 1,700 injector tests for 70,000 seconds of operation. We had twenty-six head-end assemblies overall with 55,000 seconds of operation. We did high altitude tests – twenty-seven builds, 195 different starts, and 18,000 seconds. We had a lot of experience on this hardware. But, the key to the Apollo Program success, one of the keys that I believe, is it was not a hardware-poor program. It was not, "let's just build one engine, and somehow, we'll solve all the problems as we go." It was a program that put in enough hardware that we could go to the corners of the box and find out where the failure modes were. What did it take? What were the design criteria that would, in fact, provide you

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with margin against the critical failure modes? You had to have enough hardware to demonstrate that was true. I think that was true of the F-1 engine program and on the service module program. We had hardware. We weren't nursing a few little pieces of hardware on some test stand some place and trying to figure out whether not we had really uncovered all of the failure modes and demonstrated the technical design criteria necessary to assure that those modes were adequate. (See Slide 29, Appendix H)

This program had a lot of hardware in it even though it went very, very fast. The program was an extremely fast program. The Rocketdyne engine program started in February 1963. We were told that we could start our backup program in July 1963. We had to bring on a whole Capistrano¹ test facility, build all of those sites. We did our first, full throttling range test down at that site in November 1964. That was after bringing in the steam systems, and the ejectors and everything else. We finished Phase A qualification in 1966, completed Phase B qualification in 1967, and delivered our nine engines to the customer by May 1967. (See Slide 30, Appendix H)

It was a very fast program. We had a lot of hardware in it. But, in fact, they gave us the resources that were necessary in order to carry out a program as we saw we had to do. One reason for that was we had a guy named Joe Shea, who was the NASA chief engineer of the Apollo Program during most of the early critical development. Shea was one of the best system engineers I've ever come across. He would make the decisions that said, "We're not going to stick with some specification that we arbitrarily set up at the beginning of the program. I'm going to balance my specifications and my requirements, and so, if I have a limit on this component, I'm going to find out if I can change that around and take the load over here someplace else." And he did that in real time many, many times, which is one of the reasons I think we were able to get a successful program in a short amount of time, as we did with Apollo.

For any new program, one of the things you really must have is a guy who is the system engineer and is in charge of the system engineering. You have to build that discipline in such a way that you have a complete system engineering process, and you know how requirement allocation decisions are being made. The same thing is true for the Air Force now. They are relearning that same thing – you must have a system engineering process and a leader with authority. In

¹ San Juan Capistrano, near San Diego, is where most of TRW's propulsion test areas were located. Today, Northrop Grumman Corporation (NGC) has begun dismantling facility assets at Capistrano Test Site (CTS) after taking over the former TRW Corp. in a merger.

the end, that is what controls these kinds of things. Otherwise, you back yourself into a corner with a whole set of arbitrary things, and you don't have the money to get out from that corner. I would argue that one of the first things you must think about is the structure of your system engineering process and the authority of your chief engineer. Everything else falls out from that. In our case, Joe Shea was that guy. He did the job very well. I think he got tagged with a bum rap that was not necessarily his fault at the end of things, but during this critical part of the development, he was the guy that let it go ahead, let it go forward.

We got kind of on the wrong side with some of the guys at Houston. The astronauts didn't mind what we did, but NASA wasn't too happy. We put this billboard up on NASA Road One going into Johnson Space Center, where every time people would go in, they'd see this sign saying, "The last ten miles are on us." That's because the astronauts went into orbit for lunar descent at 50,000 feet or about 10 miles. That's where we started our descent firing. (See Slide 32, Appendix H)

Gene Cernan was the guy assigned in the astronaut corps to follow the descent engine development. He was out at our facility many times. He knew everything about the engine. He was commander of Apollo 17, the final Apollo mission. He was the last man to step off the surface of the moon. When he got back to Houston, I sent him a letter with a little drawing I had done for him. It showed two astronauts hanging onto to a descent engine as it travels to the surface of the moon. One of the astronauts is saying, "When TRW said, 'The last ten miles are on us,' I didn't realize that they were going to ride down on the engine instead of the lander." (See Slide 33, Appendix H)

***Editor's Note:** The following information reflects a question-and-answer session held after Elverum's presentation.*

QUESTION: In looking at the statistics chart showing the number of starts, there was a ratio of about 20 percent sea level tests and 80 percent altitude tests in the descent engine program. Maybe that ratio was comparable for the other engine, the Aerojet engine. But, given that, let's say that thrust is an important parameter that you had to measure, especially with the throttling 10:1. We often say, "Test like you fly, and fly like you test." In terms of this engine, how did you know that the facility thrust measurement and the facility altitude simulation were sufficient to replicate or represent flight conditions?

ELVERUM: As we went through the program, what we determined, and what we all agreed on, was that the thrust coefficient (C_f) of the nozzle, after you get past a certain point, is really an engineering parameter. It's not a fundamental parameter that is going to be highly variable. Once we knew what the contour of the nozzle was, and once we knew what its characteristic was out to 2:1, we could calculate what the 48:1 thrust coefficient was going to be. In every case that we made a test, the calculation was precise. We weren't looking for a problem out at 48:1. Once we crushed the nozzle and said, "Yeah, we can land on the boulder," and once we had the thermal profile of that columbium nozzle, we did not require a lot of effort there. The real characterization was done in throttling over the 10:1 with the injector and controlling the mixture ratio on that – the whole head-end assembly – out to 2:1. I think everybody at NASA and Grumman agreed that flying like you test is great, particularly if you are using an aircraft engine. But, in this case, the thrust coefficient of the nozzle was not an issue.

QUESTION: How many engines did you test in the altitude chamber?

ELVERUM: I can't remember. We had twenty-seven complete 48:1 engine builds that were acceptance tested and tested in various types of throttling programs, which is a lot of hardware. I've been following the SpaceX program, for example, as part of the role that I do for the U.S. Department of Defense. There, they just have a few engines. And, a whole bunch of other commercial rocket companies who want to get in the business talk about, "Well, if I could get a little more money, I would be able to build two engines." They think that's going to demonstrate that we ought to put a 500 million-dollar payload on top of a space vehicle that's had two engines tested and only tested to nominal conditions. They never had a chance to burn out three of the engines and go way off nominal.

It takes a lot of hardware to be able to demonstrate that you have understood the criteria that established the risk of an engine. The risks of an engine means you have to define the failure modes and you have to do some real "honest-to-God" risk assessment. I mean *quantitative* risk assessment, not just qualitative. By quantitative, I mean you assess what the profile of the risk is, then you test that profile and see if you can collapse the uncertainty of that risk. That takes a lot of hardware. It's something that has to be put into a new program. We did all kinds of stuff like that, but it is very expensive, very expensive.

QUESTION: Do you have any comments on how the descent engine was used in Apollo 13?

ELVERUM: In Apollo 13, it was very strange. I've always had a kind of an eerie feeling about the whole program because when I was at JPL as a young engineer working with hypergolic propellants and cavitating venturis, I had no feeling or concept of an Apollo Program. Apollo 5 was the first time that we ran the descent engine in space, and it was hooked up in space after separating from the S-II.

We had the tandem configuration of the service module, the command module, and the LEM sitting out there, and we were to fire the LEM. On Apollo 5, we were firing the LEM to show how it would work. There was a problem. I can't remember where the problem was, but something caused a problem before that engine had finished its burn. It was not in the engine, but there was some other problem, and NASA made a controlled shutdown. Then, they came to us and asked, "Hey, we're up there. We want to finish this test program. Is it okay if we restart that engine again in space with this tandem configuration?" We said, "As long as it has been more than forty minutes since you shut down, our analysis says that you will be okay in terms of the thermal characteristics of the inside of that chamber." They restarted it and pushed that system around in orbit on Apollo 5.

It turned out, that when it came to Apollo 13, we went back into the record, and said, "Hey, we have pushed this system around up there on Apollo 5, and we have also restarted this tandem configuration." The requirements on Apollo 13 were to put it back into play. The spacecraft was out of free return to the earth at the time of the accident. It would not have come back. NASA said, "Okay, we'll use the descent engine to put the spacecraft in a free trajectory; it will go around the moon and be on free trajectory back to Earth." Then, as it came around the far side of the moon, the guys found out that they had an oxygen problem. As you remember, things were getting pretty bad in there. They said, "We've got to get it back as fast as we can. Is it okay if we re-fire the engine? Now, we're in a free trajectory, so we want to put as much delta-v (or change in velocity) in as we can. Can we re-fire right now?" We said, "Yes, the data says it has been this period of time." We could re-fire the engine, run the rest of the duty cycle up as far as we needed while preserving enough fluids to make the final correction as the spacecraft got near Earth, and restart the engine. It was pretty fortuitous that we could give them those answers.



Lunar Module Ascent Engine

Appendix H

Jerry Elverum's Presentation Viewgraphs

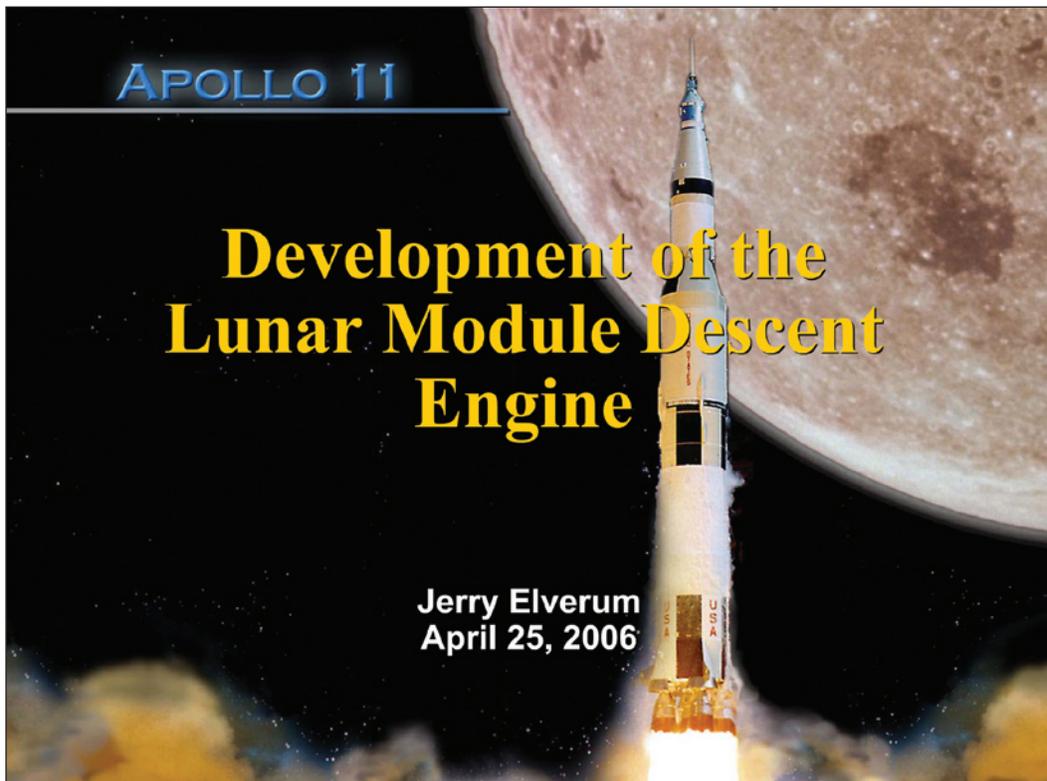


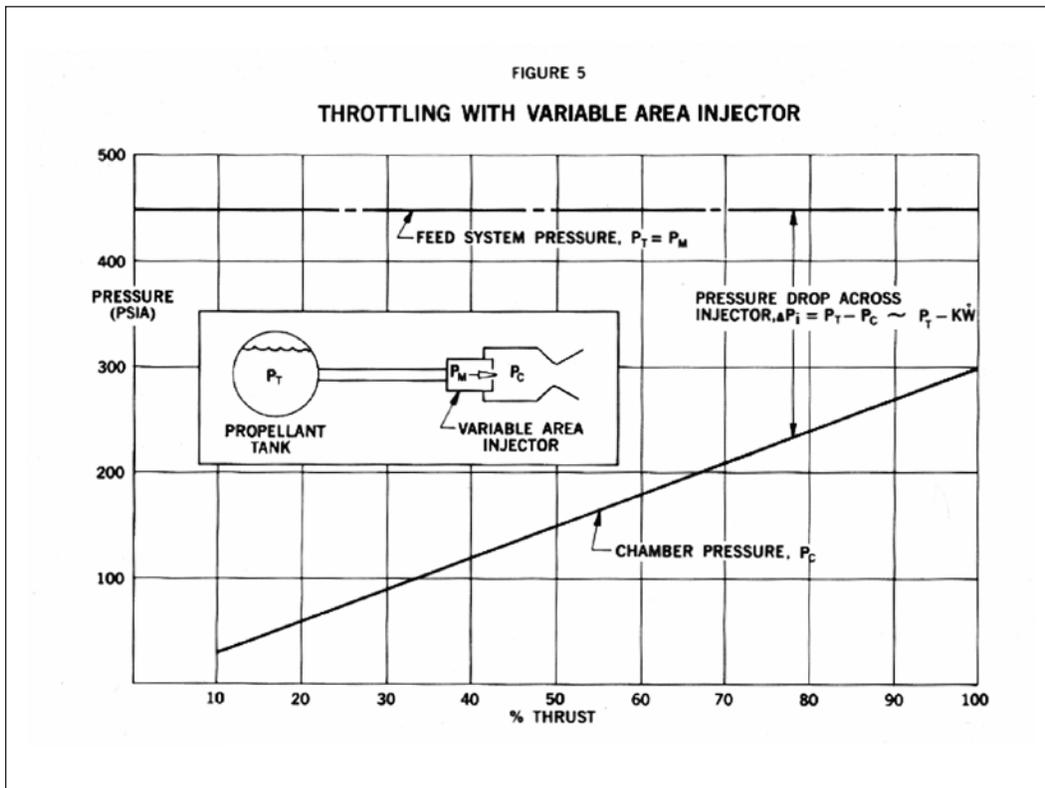
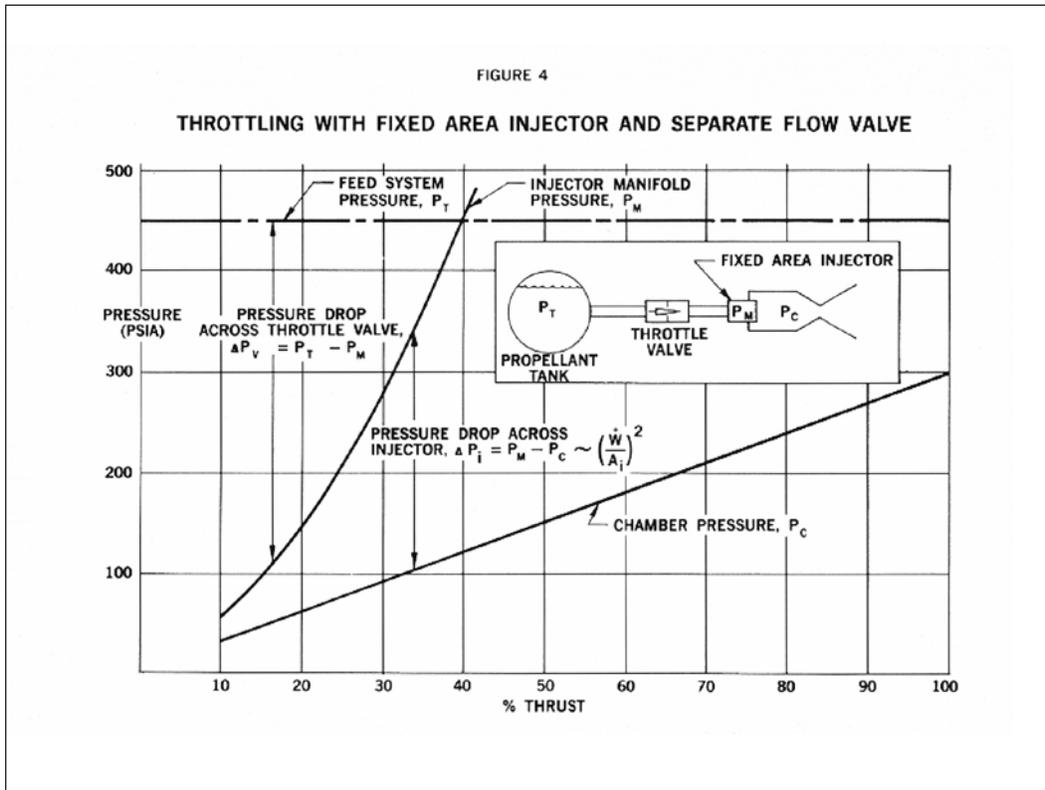
FIGURE 2
**VARIABLE THRUST ROCKET ENGINE
 APPLICATION AREAS**

SYSTEM	MANEUVER	PURPOSE	VEHICLE TYPE
SPACE VEHICLE REACTION CONTROL	EXTRATERRESTRIAL SOFT LANDING	ATTITUDE & VELOCITY CONTROL DURING DESCENT AND HOVER PHASE	SURVEYOR PROSPECTOR
	RENDEZVOUS AND DOCKING	AXIAL & LATERAL VELOCITY CORRECTIONS FOR PRECISE ORBIT TRANSFER	SAINT SATURN STAGE
	REENTRY	ATTITUDE CONTROL REACTION FORCES FOR ORIENTATION & ATTITUDE DAMPING	APOLLO
WEAPON SYSTEM PROPULSION	PROPORTIONAL NAVIGATION	WEAPON ACCELERATION PROPORTIONAL TO TARGET-INTERCEPTOR SPECIAL RELATIONSHIP	BAMBI
	VACUUM TRAJECTORY	THRUST EQUAL TO DRAG AT VARIOUS ALTITUDES	MISSILE B

- ## METHODS OF ACHIEVING THRUST LEVEL MODULATION
1. *FIXED AREA INJECTOR WITH
 SEPARATE FLOW CONTROL VALVE*

 2. *VARIABLE AREA INJECTOR*

 3. *VARIABLE AREA INJECTOR WITH
 SEPARATE FLOW CONTROL VALVE*



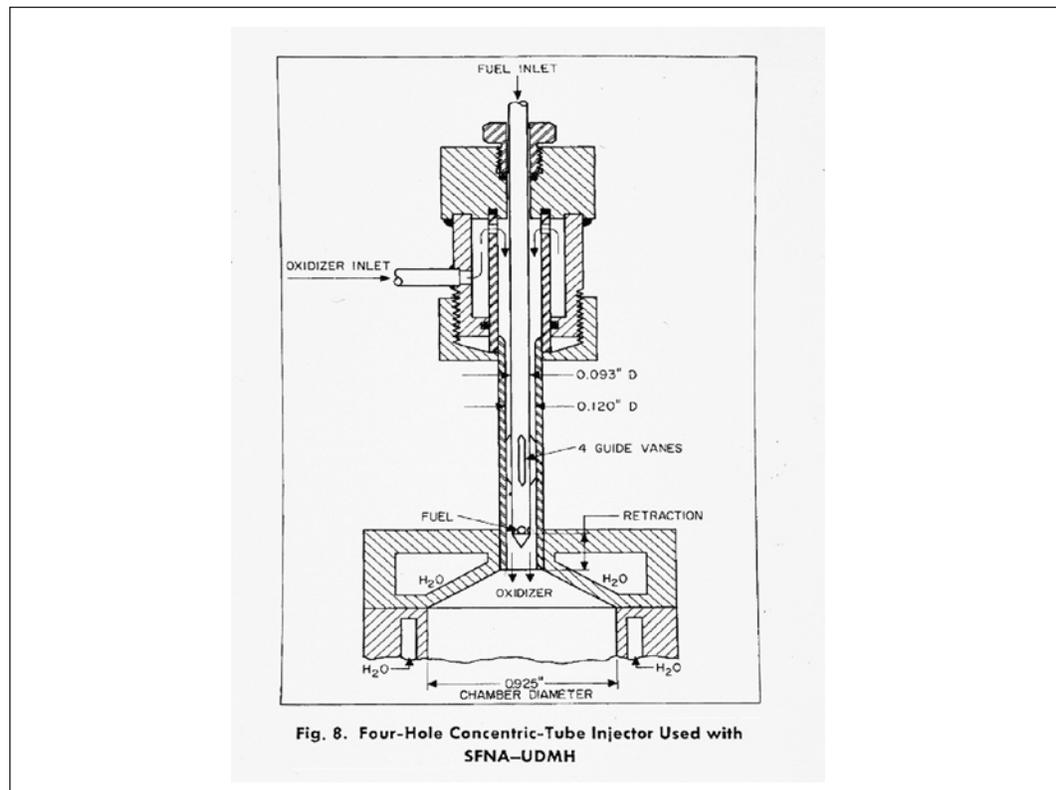
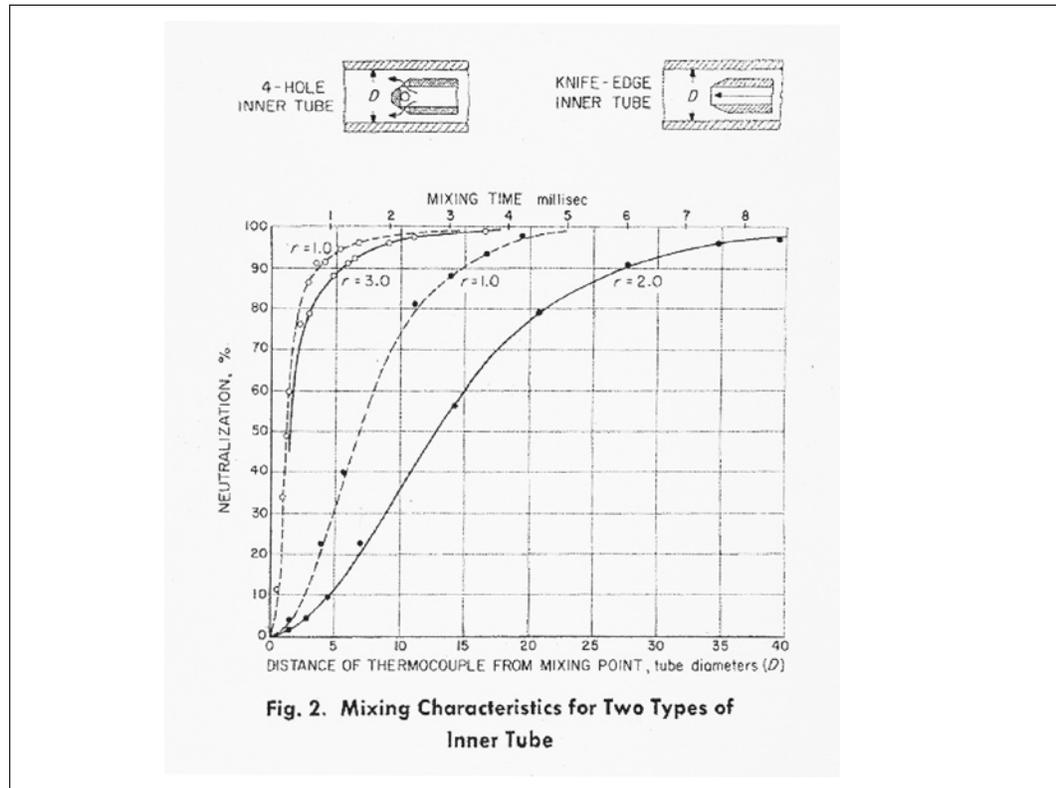




Plate 26. Concentric-Tube Injector of Plate 25 with 0.15-in.-long Chamber

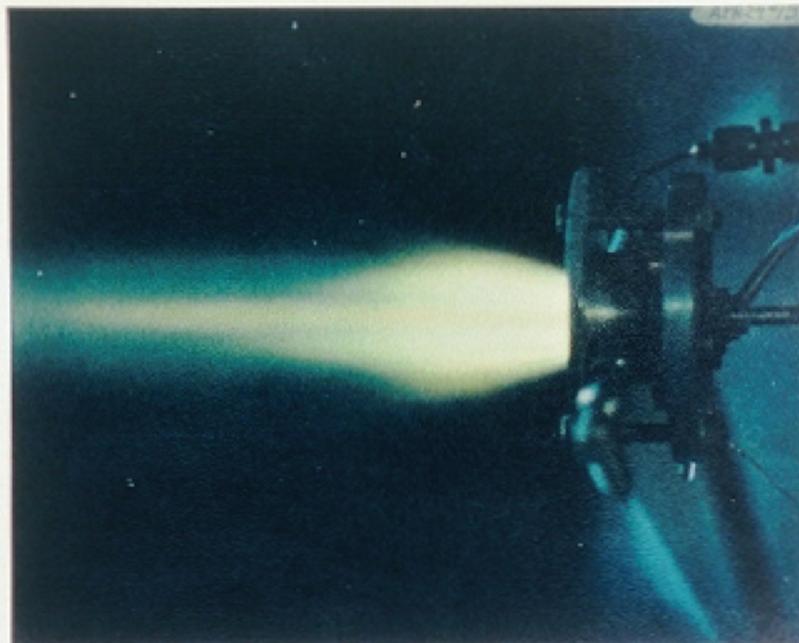
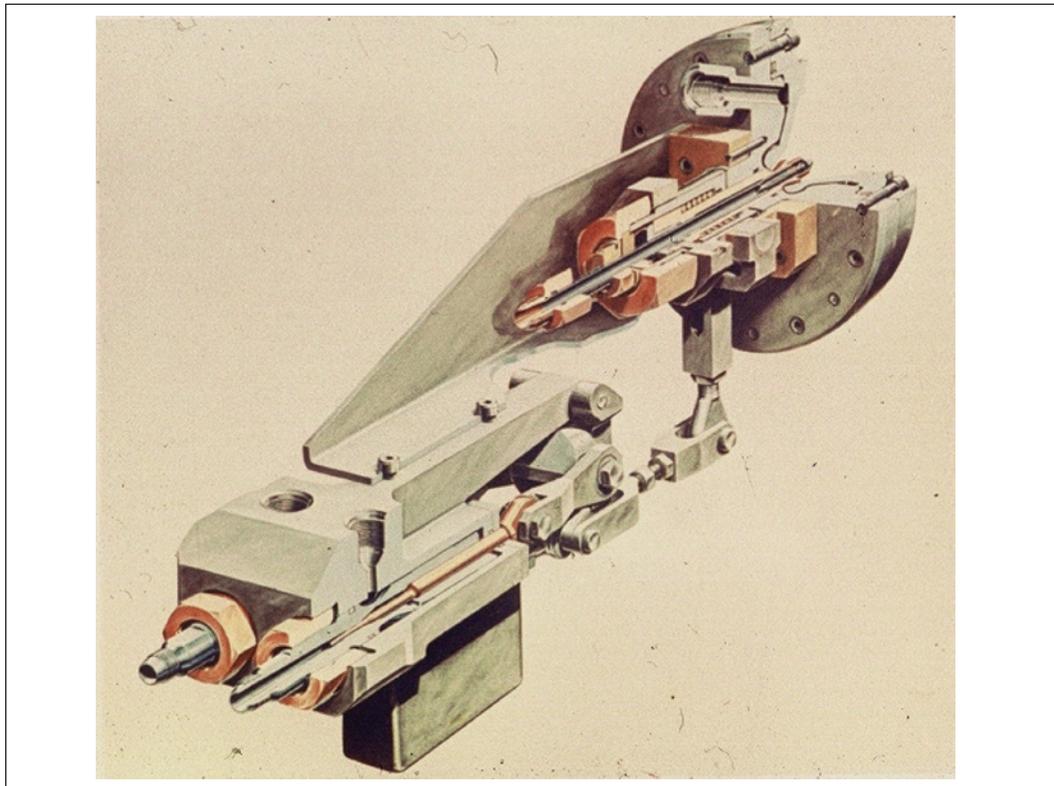
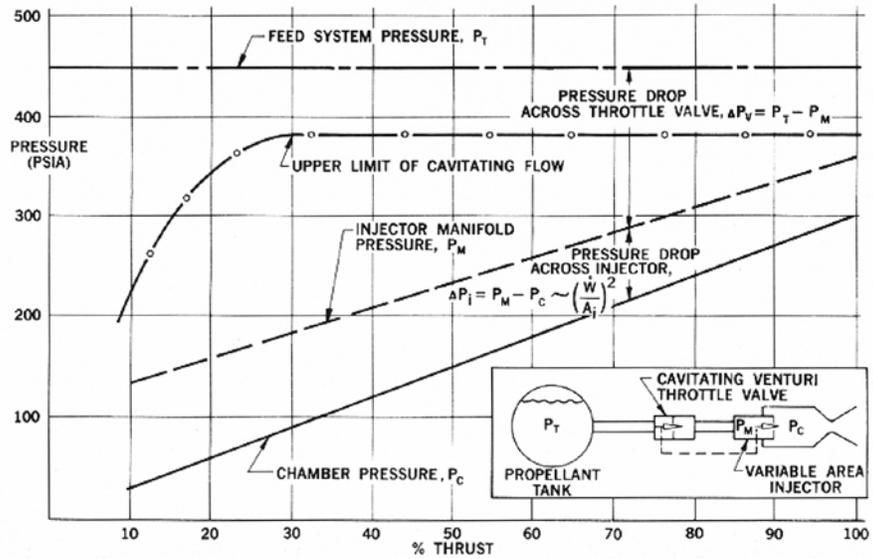
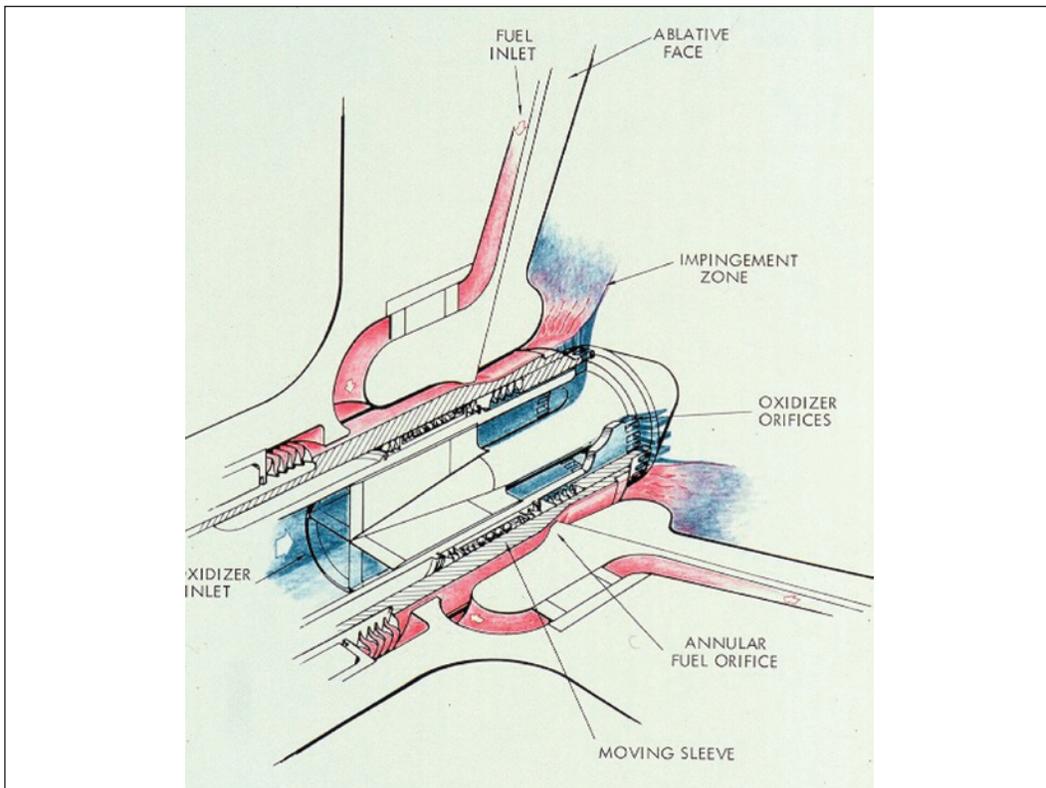
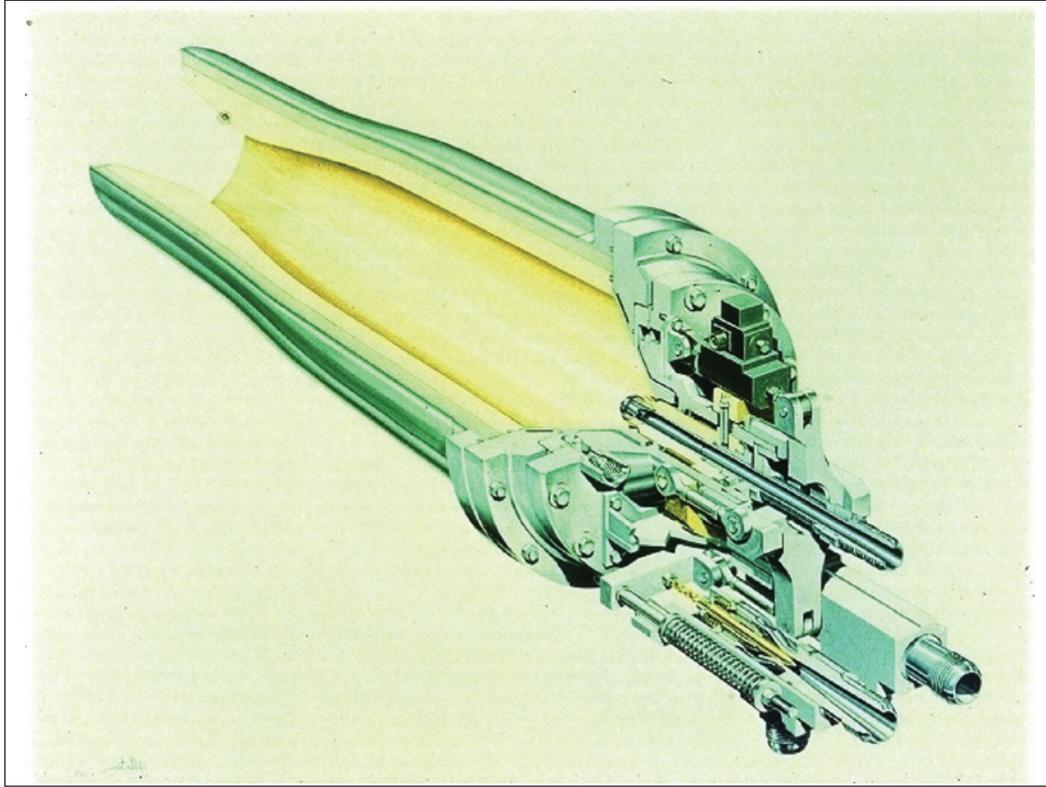


Plate 27. Concentric-Tube Injector of Plate 25 with 1.0-in.-long Chamber

FIGURE 7
STL THROTTLE CONCEPT







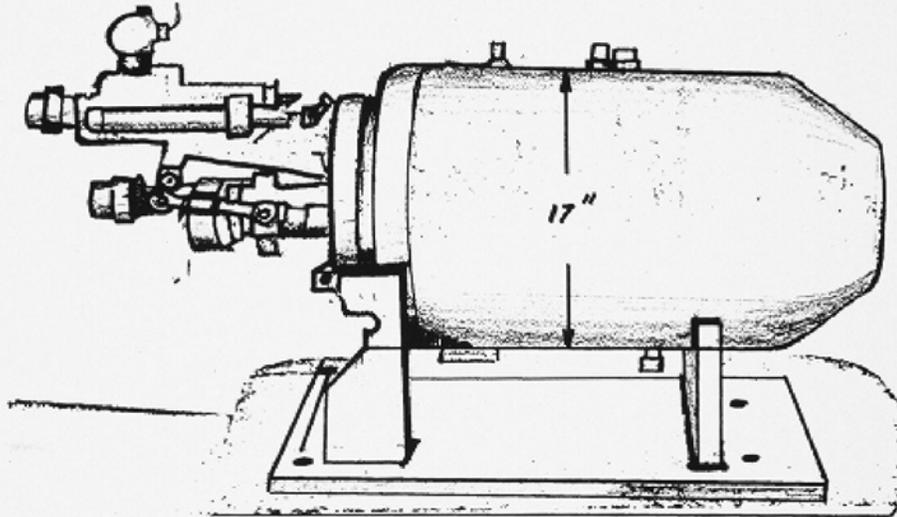
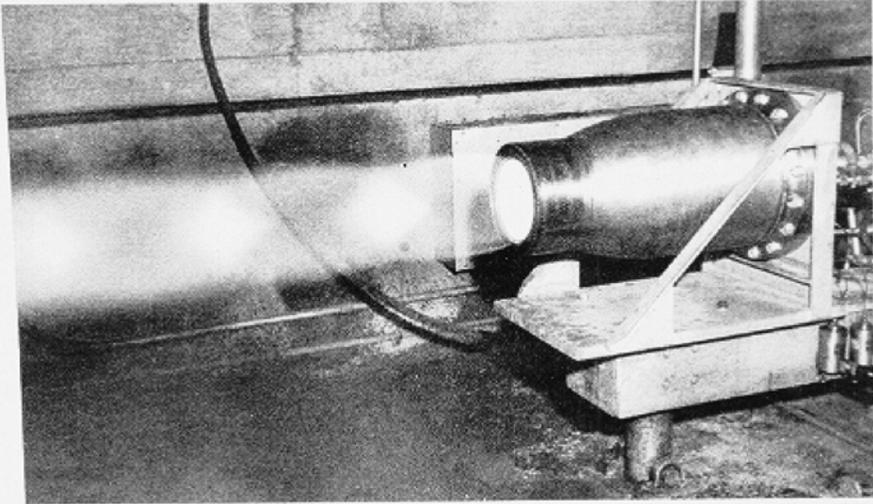
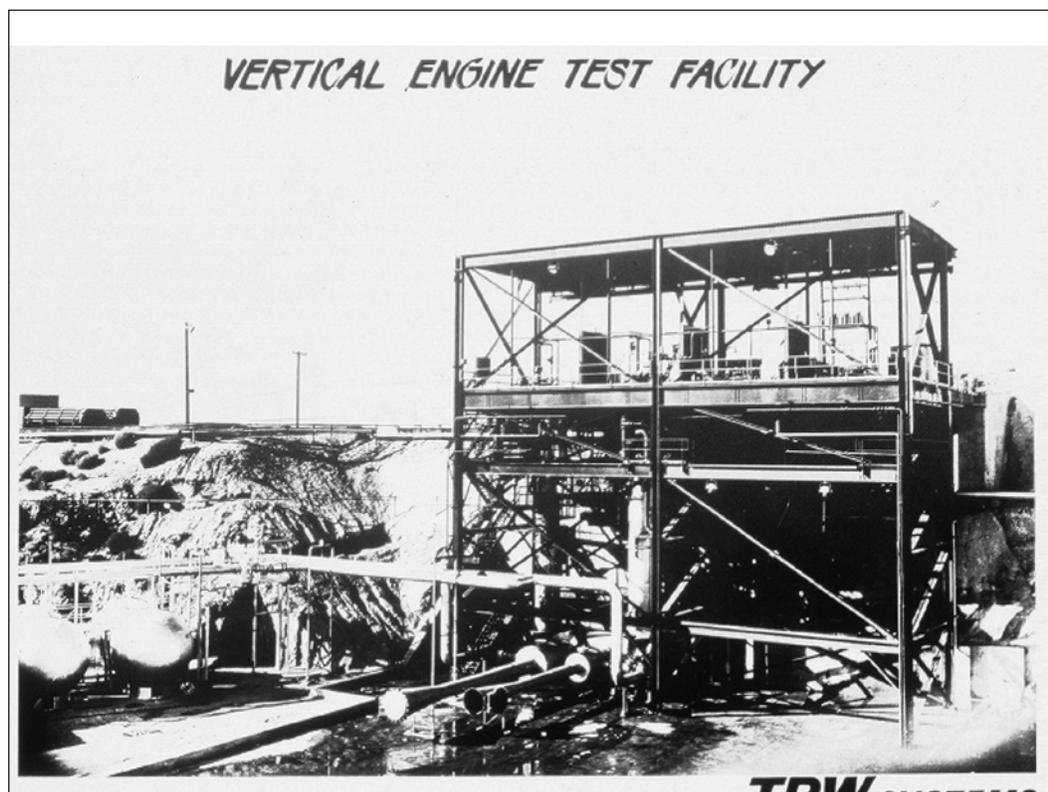
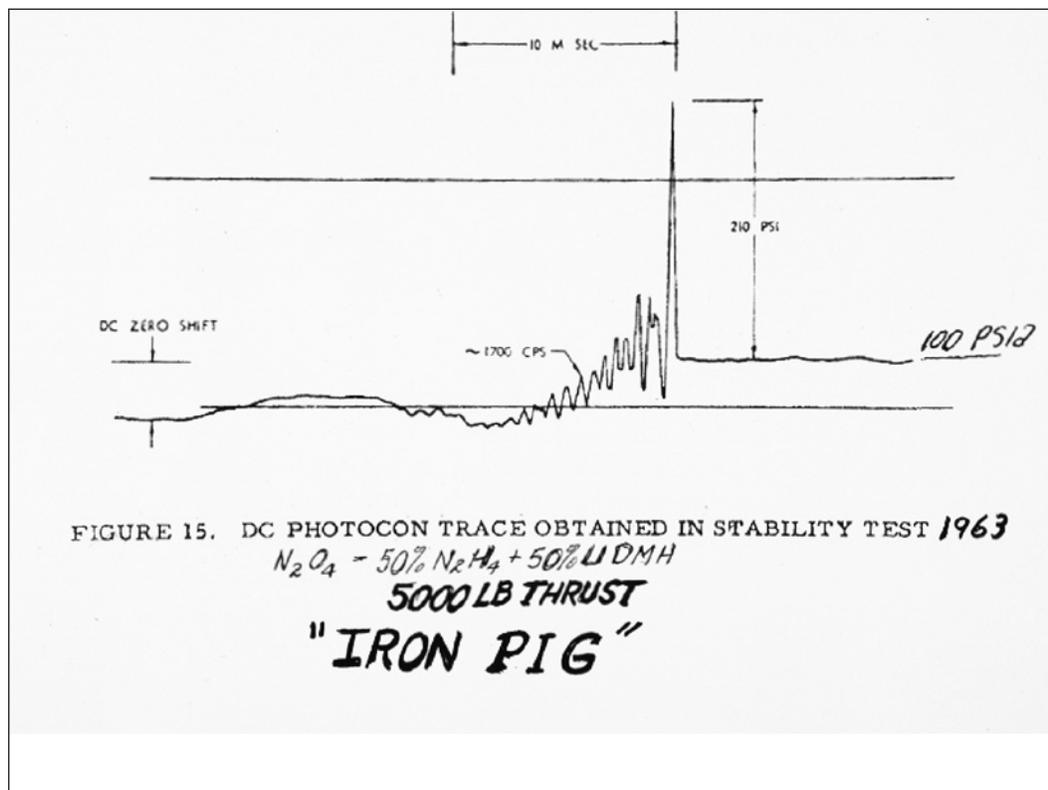
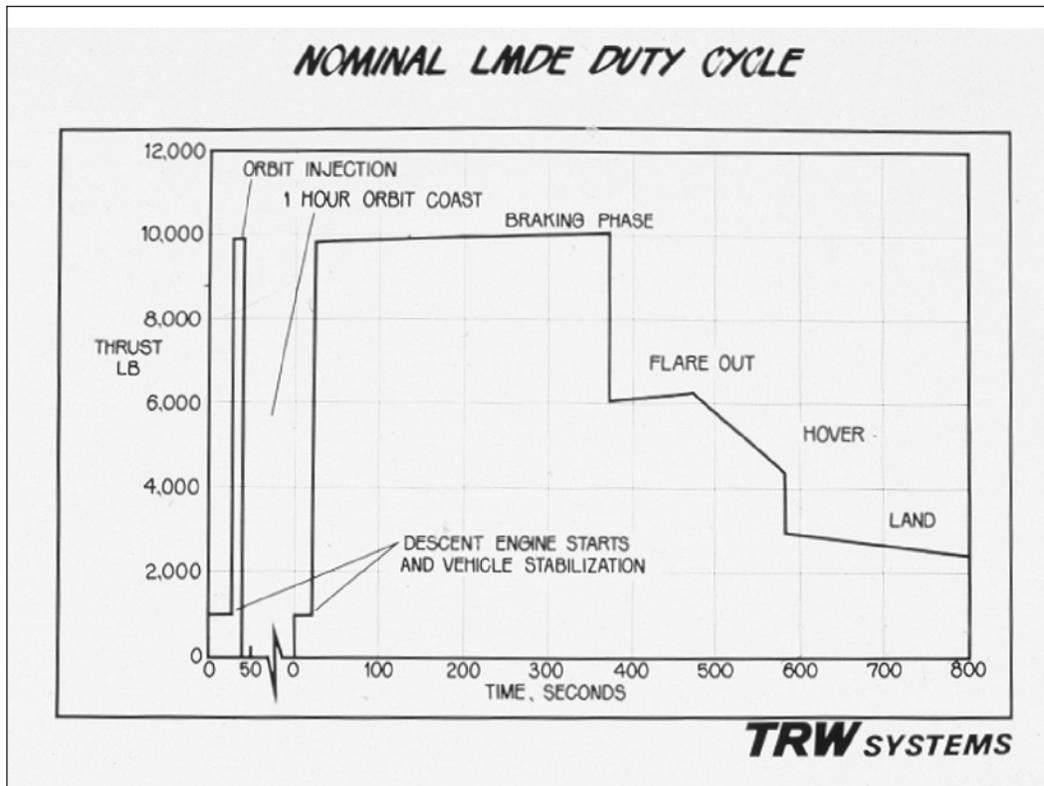


FIGURE 14. COMBUSTION STABILITY TEST CHAMBER
"IRON PIG"

5000 LB THRUST THROTTLING ENGINE





LEM DESCENT ENGINE DESIGN AND METHOD OF OPERATION TO MEET LANDING MISSION

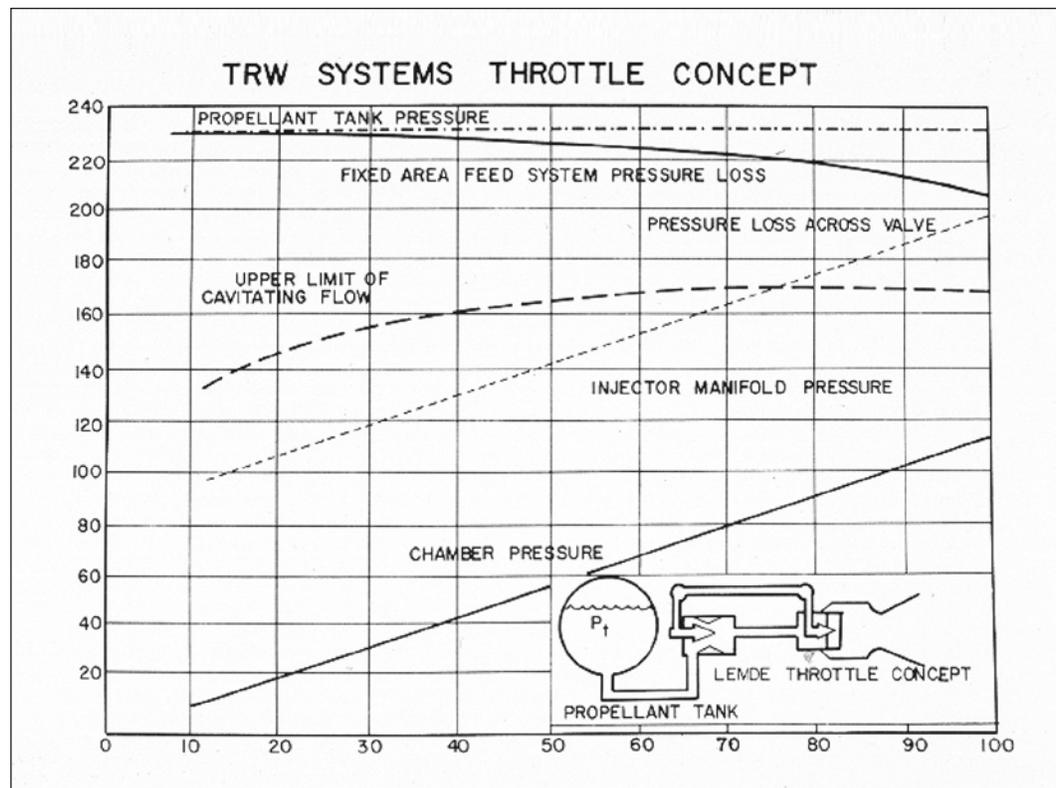
CO-AXIAL CENTRAL INJECTOR ELEMENT WITH SINGLE MOVING SLEEVE TO VARY OXIDIZER AND FUEL ORIFICES.

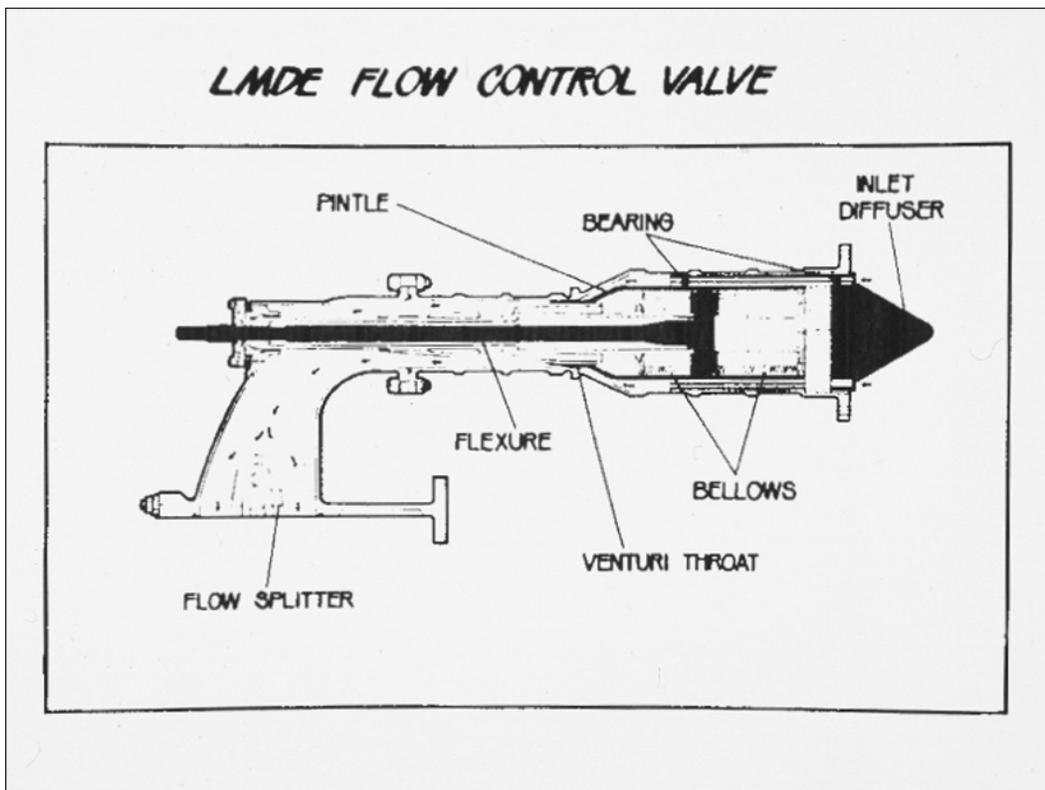
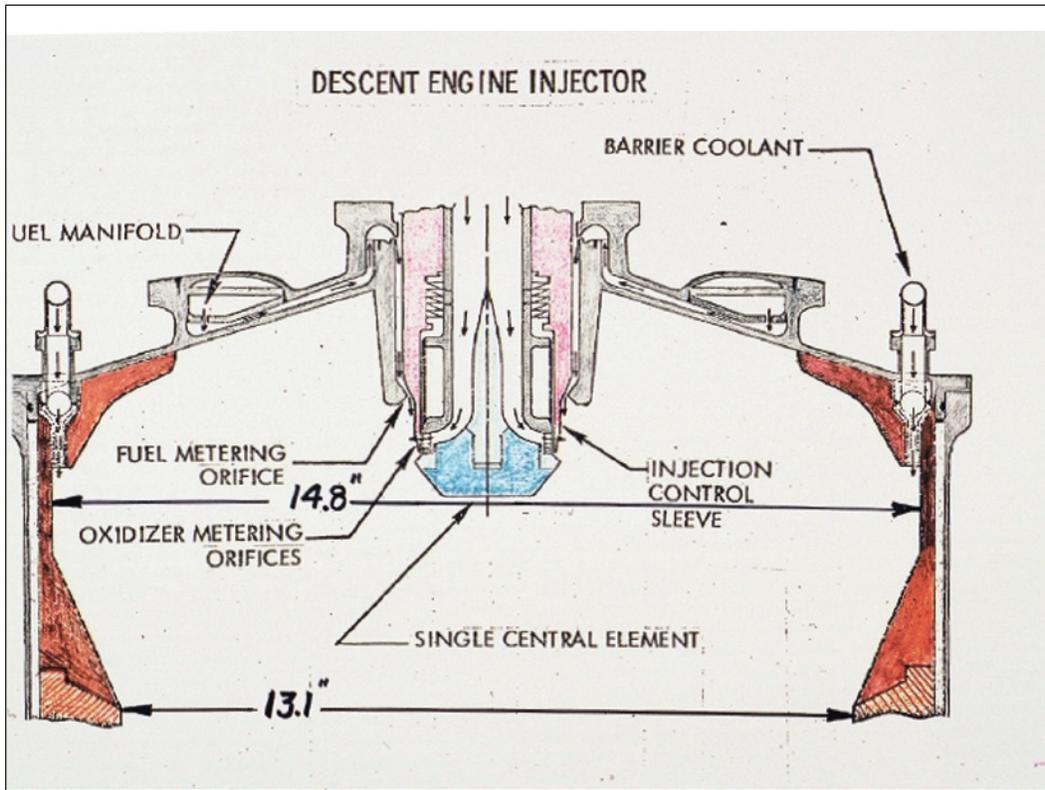
SEPERATE FLOW CONTROL VALVE, NON-CAVITATING AT FULL THROTTLE POSITION, CAVITATING FROM 60 TO 10% THROTTLE POSITIONS.

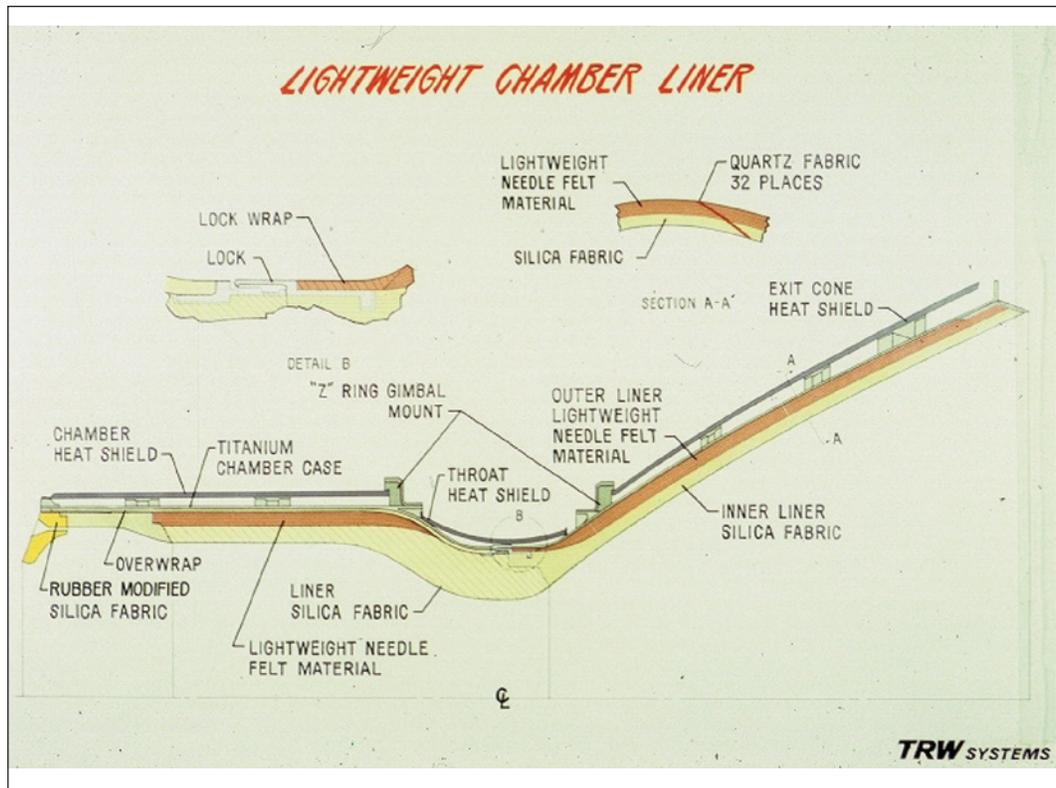
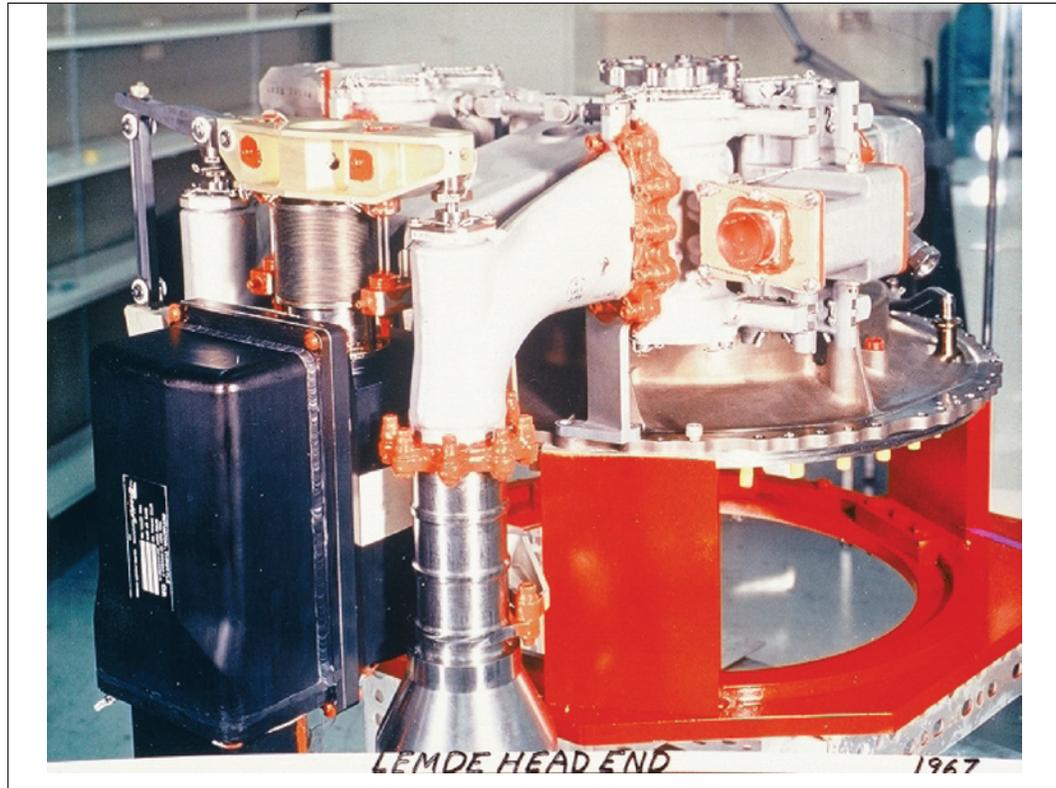
ENGINE DESIGNED TO PERFORM HIGH THRUST PORTION OF THE DUTY CYCLE AS A CALIBRATED ENGINE AT FULL THROTTLE POSITION.

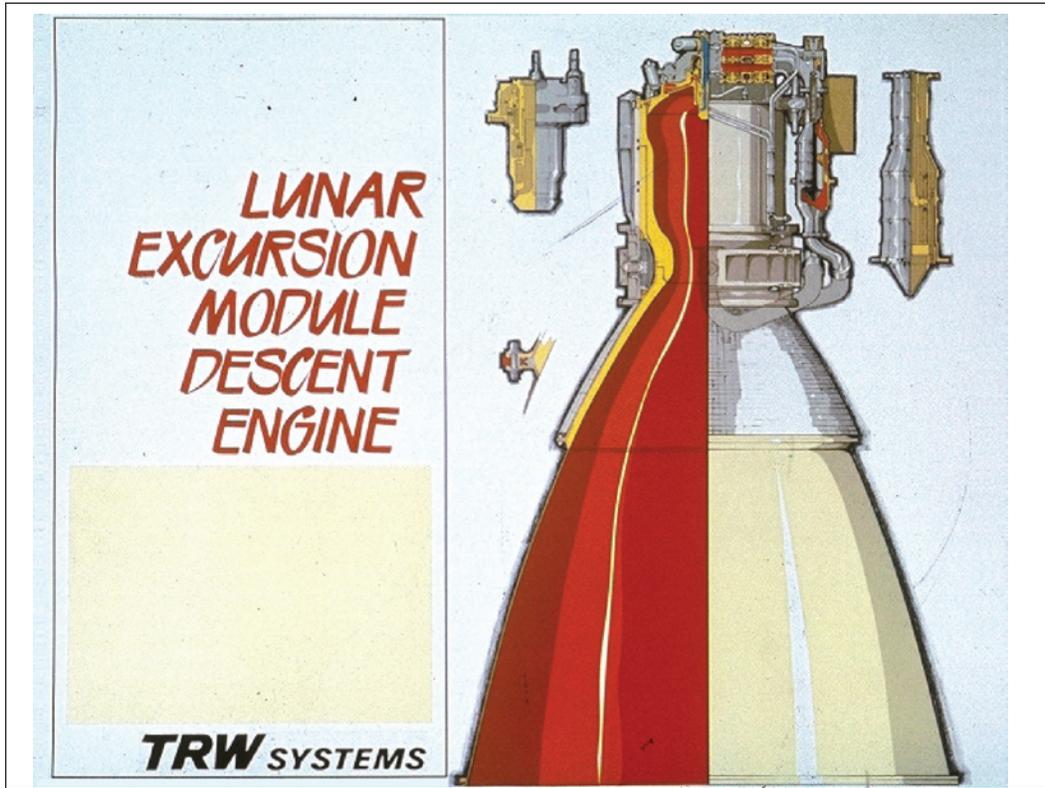
DURING THROTTLE PORTION OF DUTY CYCLE AT 60 TO 10% THRUST, PROPELLANT FLOW RATES FULLY CONTROLLED BY CAVITATING VALVES.

THRUST CHAMBER DESIGN: TITANIUM CASE LINED WITH SILICA-PHENOLIC ABLATIVE MATERIAL OUT OF NOZZLE EXPANSION RATIO OF 16:1; COLUMBIUM TO 48:1.



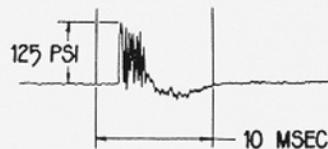






COMBUSTION STABILITY

- 31 "BOMB" TESTS CONDUCTED WITH 5-50 GRAIN CHARGES.
- INDUCED SPIKE $>175\% P_c$, ALL TESTS.
- TESTS AT 10%, 25%, 50%, AND 100% THRUST.
- TYPICAL RECOVERY:



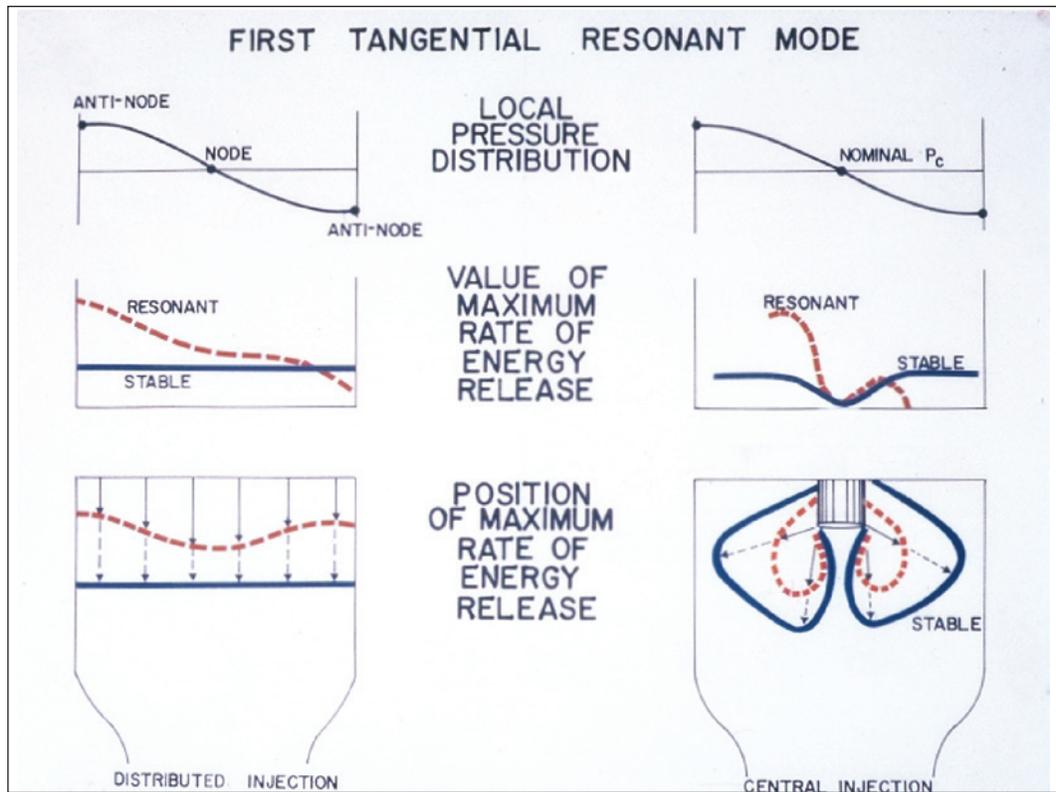
RECOVERY TIME <40 MSEC. ALL TESTS

- HIGH FREQUENCY INSTABILITY HAS NEVER BEEN SUSTAINED

TRW SYSTEMS

Major Engine Development History

Engine	Thrust Level	Instability
F1	1.5×10^6	Yes
H-1	188K	Yes
J-2	205K	Yes
Thor	150K	Yes
Titan II	215K	Yes
Titan II	100K	Yes
LMDE	10.5K	None
LM Ascent (Bell)	3.5K	Yes
Apollo SM	21.9K	Yes
Transtage	8K	Yes



LMDE TEST SUMMARY (THRU MARCH, 1967)

	NUMBER OF NEW BUILDS	NUMBER OF STARTS	FIRING DURATION (SECONDS)
INJECTOR TESTS		1,755	70,191
THROTTLING HEAD END ASSEMBLIES **	26	816	55,463
ABLATIVE ENGINES :			
● SEA LEVEL ($\epsilon = 2:1$)	20	43	4,357
● HIGH ALTITUDE ($\epsilon = 48:1$)	27	195	18,834
TOTAL TEST EXPERIENCE		2,809	148,845***

TRW SYSTEMS

LMDE PROGRAM SUMMARY

- ROCKETDYNE ENGINE PROGRAM STARTED _____ FEBRUARY 1963
- TRW SYSTEMS PROGRAM STARTED _____ JULY 1963
- FIRST ENGINE TEST IN HIGH ALTITUDE TEST STAND _____ AUGUST 1964
- FULL THROTTLING RANGE ENGINE TEST _____ NOVEMBER 1964
- TRW SYSTEMS SELECTION _____ JANUARY 1965
- INITIATION OF PHASE A QUALIFICATION _____ JULY 1966
- COMPLETION OF PHASE A QUALIFICATION _____ NOVEMBER 1966
- COMPLETION OF PHASE B QUALIFICATION SCHEDULED _____ JULY 1967
- 9 ENGINES DELIVERED TO CUSTOMER _____ MAY 1967

TRW





