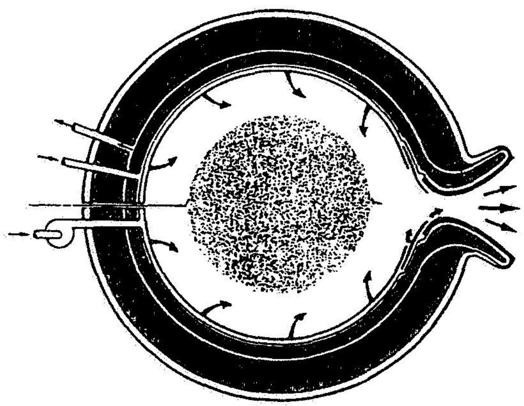
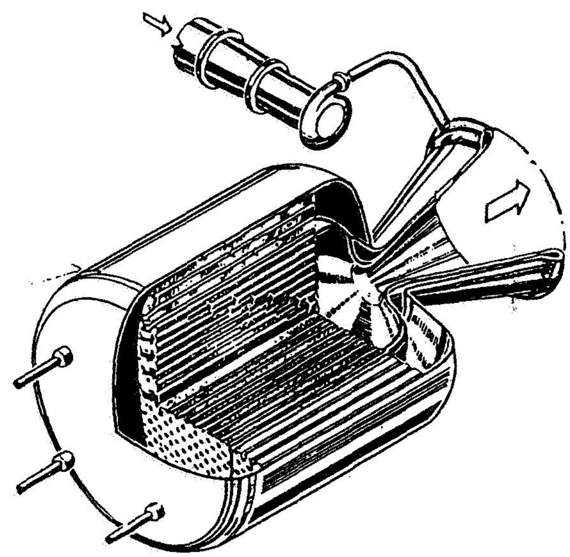
NASA Conference Publication 10079

Nuclear Thermal Propulsion

A Joint NASA/DOE/DOD Workshop





Proceedings of the Nuclear Thermal Propulsion Workshop held at the Holiday Inn Strongsville sponsored by NASA Lewis Research Center Cleveland, Ohio July 10–12, 1990

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NASA
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PREFACE

John S. Clark
NASA Lewis Research Center

The Nuclear Thermal Propulsion (NTP) Workshop, co-sponsored by NASA, DOE, and DOD, was held in Cleveland, Ohio on July 10-12, 1990. Over 200 people attended the workshop from government laboratories, industry, and academia. The purpose of the workshop was to review as many NTP concepts as possible, evaluate their current state-of-the-art, and discuss development requirements for these concepts - to provide a database from which to develop NTP Project Plans. A similar workshop was held on Nuclear Electric Propulsion in Pasadena, California on June 20-24, 1990.

An organizational meeting for the workshops was held in early May, 1990, so very little time was available for any new analyses. Therefore, most of the results and plans discussed were from earlier studies. In many cases the work was done during the ROVER/NERVA era (i.e. 1955-1972).

A Concept Focal Point (CFP) was selected to represent each concept at the Workshop. The CFP was asked to describe the concept, discuss its safety and performance characteristics, technology development activities required to advance the concept to Technology Readiness Level 6: (TRL-6 - full system ground testing complete), and present a "first-order" development cost and schedule for the concept.

Technical Review Panels (TRP) were established with recognized national NTP experts to:

(1) provide a consistent comparison of the concepts
(2) outline strengths/weaknesses, and
(3) provide a "first-order" ranking of the concepts compared to a NERVA reference engine system.

The presentations of each of the Concept Focal Points (as were the additional presentations) were transcribed and then edited for clarification for this Proceedings. No new material has been added to the resulting papers except a bibliography for each concept. Each author/CFP has reviewed the edited text and figures. I will take responsibility for any errors that may have crept in during this process, however.

The final presentations by the Technical Review Panels, while preliminary, were left mostly unedited, so as not to change the intent or content of their presentations.

I would like to acknowledge the help and support of a number of people that have

contributed to the success of this Workshop and Proceedings:

(1) Gary Bennett (NASA HQ/RP), our Nuclear Propulsion Program Manager, whose initial guidance and support provided a "roadmap" that was easy to follow,

(2) Tom Miller (NASA - LeRC), now the Manager of the Nuclear Propulsion Office at lewis, for giving me the opportunity to organize and coordinate the Workshop,

(3) the Technical Review Panelists, for allowing us to "pick their brains" again,

(4) the Concept Focal Points, who so eloquently described the concepts and technology requirements - on very short notice and with no funding!

(5) and finally to all the "behind-the-scenes" people that were so instrumental in making the Workshop and this Proceedings a success - especially James Graham and Karen Fandrich-Molnar, who have worked so diligently to get the Proceedings published.

WELCOMING REMARKS
Dr. Lawrence J. Ross
Director, NASA Lewis Research Center

Good morning, folks. This is a joint NASA/DOD/DOE workshop, but since it is in Cleveland, I get the honor of saying, "Hello," and "Welcome to Cleveland," and "Welcome to the Lewis Research Center."

I have a couple of things that I wanted to reflect on and share with you. One is that we are coming within exactly ten days now of a very important event. Last July 20th you may recall, the President, standing on the steps of the Smithsonian Air and Space Museum, described his vision for this country's civilian space program. He really charged us, and others, and began to engage the Congress in laying the foundation for what basically I think our children will be doing in this business after the turn of the century: the return to the moon to stay, and then on to Mars, with man.

In describing that vision, the President opened the door for a lot of us in the world of technology, to start searching for those things that will represent the enabling steps to making that vision a reality, for this country and for our children. One of them is what you are all here for this workshop: that is propulsion, and specifically, nuclear thermal propulsion.

During the period of time between last July 20th and now, we have done a lot of work. Everybody in the community has worked hard to try to find ways in which the program the President described in broad terms could be made to happen.

Lots of things came from that work, one of which, I should remark, is why this meeting is happening here, and why you see some NASA Lewis people around. The technology program for nuclear electric and nuclear thermal propulsion is something that Lewis has been asked by NASA to formulate, to lead, to establish the partnerships with DOD and DOE, to provide the program plan, and really get on with it because of things we (at Lewis) did many years ago.

But the other thing that came out of studies and other activities that have taken place is the realization that we are really lacking in propulsion, not only for manned Mars travel, but also the need for lifting things into space. Even to go back to the moon to stay, to do that correctly, we are limited in propulsion in this country. We have relied on very old technology in the liquid propulsion field for so long. Now we may be able to move forward to the next plateau in technology, because we have a need to do it, which we haven't had since the Appollo program.

So we have twin needs. One is the conventional booster technology, Advanced Launch System (ALS) type technology, and the new breakthrough stuff that will put on the table for the first time, in a real sense, the proposition of sending human crews to Mars. That requires nuclear thermal. To do that we must dust off the knowledge we had many years ago, reestablish partnerships, get focused on those roadblocks and enabling technologies that need our attention early on and get on with it. This is a workshop that can share the kind of information and create the kind of baselines that begins to do that for us.

I just want to say a quick word about the political scene. Fortunately, technologists' "time constant" or "compute cycles" are much longer than the vagaries of the political scene, so maybe not too many of our technologists are terribly concerned about this year's or next year's budget fights. SEI, the Space Exploration Initiative that I was talking about is currently out of the budget, and that jockeying around will go on ad infinitum.

I have no doubt, notwithstanding the labels given to the technology program, that NASA is going to be asked, and funded, to do it. We are going to find ways of putting our resources and manpower against the right things. We are not going to do the wrong things. So I would just caution, don't let your enthusiasm be diminished by paying too much attention to Space News and those other things that will tell you "the sky is falling." NASA's $14.7M budget is not bad. You can do a lot of things within that budget and we will certainly work hard to do the right things.

It is my honor to welcome you here. If there is anything we can do for you, you are very close to Lewis. So if we can do anything for the visitors from out of town, just grab a Lewis person and they will be more than happy to help you with anything.

Good luck and have a productive rest of the week. Thank you.

N92-11089

NUCLEAR THERMAL PROPULSION PROGRAM OVERVIEW

Gary L. Bennett
NASA Program Manager
Propulsion, Power and Energy Division
Office of Aeronautics, Exploration and Technology
NASA Headquarters, Washington, DC

Picking up on what Larry Ross said, we are coming up almost on the first anniversary of the President's speech (Figure 1) committing us to finishing space station, going back to the moon and then going on to Mars, and he has repeated that on a number of occasions over the past year, and the money was put in the fiscal 1991 budget to work on the Space Exploration Initiative.

Specifically, the President requested $179.4 million for exploration technology, and of that, $11 million was earmarked for nuclear propulsion, subdivided into $10 million for nuclear thermal and $1 million for nuclear electric propulsion. There was flexibility put in that we could do studies on either concept under the 10 million.

And the President, again in the speech that he gave last July (Figure 2) spoke of finishing Space Station, going back to the Moon and then the mission to Mars; that's really the focus of our exploration technology program: the return to the Moon and then going to Mars.

In one of the meetings that I attended with Frank Martin, who was head of the Office of Exploration before it was merged with the old Office of Aeronautics and Space Technology, Frank said he did not necessarily need nuclear propulsion to go to the Moon but he certainly felt it was almost enabling to go to Mars.

The President in a number of speeches has talked about going to Mars within the lifetime of the scientists and engineers (Figure 3) who are going to be brought onboard to work the program and also to have people on Mars by the time of the 50th anniversary of the Apollo 11 landing which says that we have to be there by 2019. Now, in February (Figure 4) he approved policy for the Space Exploration Initiative and he said that is going to include both lunar and Mars elements as well as robotic missions. He said the near-term focus is going to be on technology development. And you may be aware that we have an Outreach Program, which General Tom Stafford is heading, and which is in response to the Vice-President's request that NASA cast a wide net looking for innovative ideas. There will be meetings and so forth coming up on that. In fact, NASA and the AIAA are sponsoring a meeting in the first full week of September to look at some of the technology items for the Space Exploration Initiative. There is going to be a focus on high leverage innovative technologies and certainly I think nuclear propulsion, both NTP and NEP, are part of that.

Now, it is probably going to take several years to come up with the mission architectures for how we go back to the moon and go on to Mars, and that requires us to maintain a certain amount of flexibility. I think in the nuclear propulsion program we have to be able to adapt to whatever comes out of these studies and we have got to be able to provide the planners with the information they will need.

NASA is going to be the principal implementing agency, but we are going to be working with the Defense Department and the Department of Energy. Certainly, when we get into nuclear propulsion we recognize the capabilities of the DOE laboratories, and we have been advertising the current workshops as joint NASA/DOD/DOE meetings. And I am glad to see the attendance from those agencies here.

Last November, the President approved our current version of the national space policy, which updated the policy that was in effect during the Reagan administration (Figure 5). One of the key points in that was that our goal is expanding human presence and activity beyond Earth orbit and out into the solar system. There is a background on this policy that's been developing over several years, dating back to 1986 when the Congressionally-mandated National Commission on Space issued its report on "Pioneering the Space Frontier" and in this report there is discussion of nuclear propulsion. And then Dr. Sally Ride issued a report to the NASA administrator in 1987 and that laid out about four mission scenarios including going to Mars. Of course, one of her recommendations was that NASA create an Office of Exploration, which NASA did, and that office issued the first of a series of annual reports in 1988, also looking at the Mars mission.

There are a number of reasons why we should go to Mars, (Figure 6) and certainly technology and education are key parts of it because we need something, at least in my view, that inspires people to go into science and engineering. My personal view is we have enough lawyers; we need people who are going to go out there and give us the technology edge because, as all the commentators are pointing out, the battle in the early 21st century is not going to be military, it is going to be economic. Certainly, continuing our journey into space and to Mars gives us the chance to understand planetary evolution. Perhaps the most fascinating thing concerns life on Mars, if it ever got a chance to start, and if not, why not. So again, that's our long-range goal and that's our focus on the nuclear propulsion program that we are developing.

Now, during the last year following the President's July 20th speech, NASA set up an in-house group which did the "90 day study," and that study looked at going to Mars and identified a number of key technologies (Figure 7) that are needed for human exploration of the moon and Mars and nuclear propulsion was one of those key technologies. So that was the first highlight on it.

Then in response to that, we put together within OAET, now the Office of Aeronautics

Exploration and Technology, an Exploration Technology Program which is to develop a broad set of technologies (Figure 8) to enable future decisions on development of future space exploration missions.

The Exploration Technology Program is not a sandbox, it is to be a critically needed focused technology program and it includes these technology areas. And again, one of them is nuclear propulsion. And the explorative technology program is the one that was budgeted at $179.4M in the President's submittal for FY91.

Now, specifically for nuclear propulsion these are the words that went into our internal budget documents (Figure 9). And we said that the technology that will be developed under nuclear propulsion is to address multiple approaches (Figure 10) for applying space nuclear power systems to the improvement of nuclear performance for human missions to Mars. We said we would start work on a nuclear thermal rocket propulsion technology and at the time we said solid core and gas core systems would be looked at. And later I will mention also liquid core concepts and all of these concepts were to be considered for future piloted missions to Mars. We also said we would be working on nuclear electric propulsion technologies and that would include both the reactor and the electric propulsion system.

For those of you who have followed this, we had a previous program called Pathfinder which is the precursor, if you will, for the Exploration Technology Program. We did have an element in the pathfinder program called Cargo Vehicle Propulsion which unfortunately was not funded. That was focused strictly on the electric propulsion thrusters. Now, under NEP, we have the reactor plus the electric thrusters and we also have nuclear thermal propulsion. So when we talk nuclear propulsion it consists of two key elements, and we have put together a draft thrust plan, as we call it, for all of nuclear propulsion and that is a draft document coming out of Headquarters.

We have set up various roles on this. Lewis Research Center here in Cleveland is our lead center within the NASA complex on working nuclear propulsion. And they are helping us pull this whole activity together. In nuclear thermal propulsion, they are being assisted very ably from the people from Marshall Space Flight Center in Huntsville, Alabama, and in nuclear electric propulsion they are being assisted by JPL, Jet Propulsion Laboratory, in Pasadena.

At this point I should thank a whole lot of you because you are going to see charts up here from the various NASA centers and DOE laboratories and contractors. Bob Frisbee at JPL frequently reminds me the difference between plagiarism and scholarship is whether or not you acknowledge the sources, so I want to acknowledge a lot of you on this. Now under nuclear thermal propulsion (Figure 11), we are going to be looking at the whole system, the reactor, shielding, pumps, and all of that.

Larry Ross mentioned, in his opening remarks, the previous work done at Lewis managing the NERVA program and other activities. I think that there is a synergism between chemical propulsion activities and nuclear propulsion activities, and this is a message I have gotten in talking to people at Lewis and Marshall. The ROVER/NERVA program in many ways led the country in the 1960s on cryogenic technology, but the chemical people with the Space Shuttle main engines and so forth have since gone beyond; there are things that we can learn from them.

One thing I would like to do is not get into a chemical versus nuclear mode; rather I would like to adopt a view that nuclear is simply an extension of chemical. We are going to take the chemical technology for pumps and nozzles and so forth and just heat the propellant in a different way.

Within our thrust plan, we have a number of goals (Figure 12). These include developing the technologies to apply space nuclear power to improve the performance for human missions to Mars. Our focus is really on the piloted missions, and out of this we want to come up with at least one concept that alone or in combination with other systems can meet the requirements for piloted and cargo missions to Mars.

Now, in combination it could be something like nuclear thermal propulsion plus nuclear electric propulsion, the hybrid concept. I think there is at least one talk on that scheduled during this workshop. It also might end up being chemical plus nuclear. There are various ways perhaps to do it. Our objectives (Figure 13) include developing safe advanced nuclear propulsion systems that are responsive to the Space Exploration Initiative requirements, and we have to have a focus on safety.

Right now NASA has a court case pending on the Ulysses mission, which is a European Space Agency spacecraft that NASA is launching this fall, and which has one radioisotope thermoelectric generator. We have been taken to court to stop that launch. We have also been asked to not allow Galileo to fly by the Earth in December, and we may get the judge's ruling this week. We have to be ever mindful of safety whenever we get into this nuclear arena. As the cliche goes we have to be squeaky clean. In fact, as one fellow said, if out of all of these workshops one piece of paper finds its way into the gutter and somebody comes by and picks it up, that piece of paper had better have the word safety on it.

We are going to look at component subsystems and systems technology, and what we want is to come out with a validated base for moving on in nuclear propulsion. There are project level goals (Figure 14 & 15) and Lewis has taken the lead and will be working with Marshall and JPL in coming up with project plans on nuclear thermal propulsion and nuclear electric propulsion.

Now, there is a bit of a strategy behind this I would like to spend a few minutes on. In putting all of this together, again our focus has been on safety, reliability and high performance technology. As to reliability, we are of course, aware of the problems on

Hubble and other things and so this is going to be a challenge for the people working the panel on advanced planning. How do we test a nuclear propulsion system? That's going to be something we are really going to have to wrestle with, and certainly there are strong arguments for all-up testing on the ground if we can do it.

We certainly need to work with the public, with Congress, and the administration on developing a consensus on the safe use of nuclear propulsion, because now we are doing something a little different from say a Galileo or Ulysses, where the device is just sent out. We are talking about sending people out a nuclear system and bringing them back into, perhaps, a low Earth orbit. And in fact, there is a meeting scheduled today in the Pentagon to wrestle with the question of the effects of gamma rays and other particle emissions from reactors on scientific satellites. Congress mandated that the Defense Department would provide a report on how reactors in space might affect science satellites such as the Gamma Ray Observatory and so forth. We are going to have to be sensitive to that with nuclear propulsion.

Out of our work we have got a chance to strengthen and extend the propulsion technology foundation for the civil space program. Again I want to emphasize we are just taking chemical another step further. A key part of this effort has to be involving the universities, because that's where the people are coming from who are going to carry these programs into the 21st century. Also, the program really needs to be done with other agencies such as the Department of Energy, the Department of Defense; their labs and their contractors have expertise that we don't have at NASA, and I think this maximizes the use of existing resources. And obviously, in this country, if you do something it really ends up being done by industry and by laboratories. So it's got to be done as a team approach involving industry and the universities and laboratories.

Again, to emphasize, this is to be a phased and focused technology development program. We have been asked throughout the Exploration Technology Program to set up "wickets" through which these various ideas have to flow and we are going to have to make decisions as we go along. We cannot continue to work nuclear propulsion or we cannot work our life support or artificial intelligence or whatever indefinitely. We have to be focused on where we are going with them.

The last issue is maintaining a flexible design approach. If you go back and look at the ROVER/NERVA program, it started out when the Air Force went to the Atomic Energy Commission looking for a way to have an ICBM, and they wanted a nuclear rocket ICBM. Then, when NASA was created, it became a vehicle for going to Mars. Next it became a tug to go from low Earth orbit to lunar orbit; so that's part of the reason you see a multiplicity of nuclear thermal propulsion designs in the late 1950s and 1960s. The requirements keep changing, so we have to be flexible; but as a colleague of mine once said, "we have to be flexible but not limp."

There are a couple of things on "why nuclear propulsion" that are coming out of studies that Lewis and Marshall and others have done. If you look at an all-propulsion chemical system, the initial mass requirement in the low earth orbit is pretty humongous (Figure 16). Once you go down into an aerobrake system or a nuclear thermal rocket either at 900 to a 1000 seconds Isp and even nuclear thermal rocket with aerobrake, they all significantly improved (Figure 17).

And I might mention that we have had some discussions on what we need to know and I will start by saying when nuclear is compared against chemical plus aerobrake, the aerobrake mass fraction used is quite often an optimistic assumption of 13 to 15 percent or something like that, so that needs to be noted. These have been the typical measures of performance, but there are people in Headquarters who have asked me a different question, not so much about the required mass in the low Earth orbit but about the trip time. In this particular study (Figure 18), for example, the electric propulsion systems were of the order of 650 days, although with some sort of a boost either from nuclear thermal or chemical they can get that down to a time comparable with nuclear thermal.

During the 90 day study there was a lot of interest in nuclear gas cores (Figure 19), simply because of short trip times, and there are people out there who believe that this is the major selling point for nuclear propulsion, getting people to Mars quickly so we don't have excessive life support issues to deal with, we don't have to extend the time during which the astronauts might be exposed to a solar flare, and we minimize the radiation dose they get from galactic cosmic rays.

This is another chart from Lewis showing plots of relative mass in the low Earth orbit as a function of engine thrust/weight (Figure 20). These have always surprised me, but the message that comes out of these is above about six to ten, thrust-to-weight isn't as important as specific impulse. So things to think about as you go into these deliberations on going to Mars are, short trip times and high specific impulses.

This is a chart that was presented at the NEP workshop (Figure 21). Perhaps there is a clue here that, if we are willing to relax our mass in the low Earth orbit, we can start pushing for shorter trip times, and perhaps nuclear will get there more quickly than chemical plus aerobrake.

As something that I want to leave you with, I will quickly mention that these nuclear propulsion systems certainly give us versatility (Figure 22). In the ROVER/NERVA program basic modules were developed, and they can be stacked up depending on what the mission is. Nuclear thermal propulsion and even nuclear electric propulsion offers the possibility of using in-situ propellants (Figure 23) and Bob Zubrin will be talking about that later.

In the days of NERVA, and more recently in other studies, people have looked at using the reactor not only for direct thermal propulsion (Figure 24), but also to drive a turbine alternator so you could have both power have a nuclear electric propulsion system as well.

The nuclear rocket program as set up in the 1950s and 1960s runs roughly like this (Figure 25). The point I want to make is that Los Alamos was turned on in about 1955 and the KIWI test started about four years later; this was before the National Environmental Policy Act and was a classified program. Also, Westinghouse and Aerojet were turned on around 1961, and again it was several years before we get into the NRX series.

It's now going to take several years to get a ground facility built up and running and tests going, so we need to be realistic about that. We may be a little optimistic in some of our sales pitches, but I think we ought to not kid ourselves about its taking time to do this.

This just simply shows the evolution of the Los Alamos concepts (Figure 26) and this was the Aerojet/Westinghouse NERVA (Figure 27) and I won't dwell too much on that. These (Figure 28) are various ways of running the engine and this breaks out the individual tests (Figure 29), ending up with the nuclear furnace.

Now, the NERVA/ROVER program had a price tag in 1960 dollars of $1.4 billion; if you mention those kind of numbers today people get a little nervous; but I have been told by several people that the cost of developing and qualifying the chemical engine on the advanced launch system is about $4 billion. The chemical people historically have thought of at least a billion dollars to qualify a chemical engine, so I don't think we need to apologize in the nuclear community that we might spend more than a billion dollars to develop something that is at least twice as good as what we have today. Nor should we be apologetic about the fact it may take several years to do it.

Even though the ROVER/NERVA program ended about 1972, some people have continued to work on it. Las Alamos and INEL looked at small advanced nuclear rocket engines (Figure 30) and low pressure engines and Brookhaven looked at particle bed reactor design (Figure 31), which improves heat transfer. And recently I was made aware of the fact that Brookhaven has looked at a liquid annular reactor system (Figure 32), about which they will talk later, which is a step toward the gas core system and allows even higher temperatures. Also, of course, work was done under the ROVER/NERVA program on gas core systems (Figure 33), wherein you could push the uranium plasma up to 10,000 degrees Kelvin.

Additionally, there was a nuclear light-bulb, and we will be hearing about this over the next few days. On paper, these advanced concepts certainly offer the possibility of quick trip times, because they have the right combination of thrust and Isp.

Now then, what we want to do, given the fact that there are these various concepts both under solid core and gas core and liquid core (Figure 34) is study them, get into more detailed designs, do some component testing, with the idea that somewhere toward the end of the decade we would come up with a basic nuclear thermal propulsion concept and similarly, a basic nuclear electric propulsion concept. So, basically, these workshops are put together as a way to educate those of us who are working on the nuclear thermal propulsion, and as a quick way to find out where all these concepts are. We do not intend to use the workshops to make any sort of selection, however.

Ideally, we would like to carry a number of concepts along in the planning, and for those of you who were involved in the ROVER/NERVA program, that program did more than just NERVA and Phoebus and so forth. It also worked on things like gas core and so forth. So it kept alive even more advanced technologies and I would hope that we would be able to continue to do that, and that we would not look at any one of these concepts as the be-all for the rest of the duration of humanity's existence.

There are a lot of issues that we have got to look at (Figure 35). Again, chief among them will be safety and safeguards plus quality assurance, how we test these concepts and what sort of reliability program we come up with. Obviously we are not going to be able to test dozens of these systems, so we have got to come up with a test program that will enable us to calculate the reliability and still come up with good reliability. These are the kinds of issues that we must address in our programs.

The scope that we will be working on (and liquid core should also be in here) (Figure 38), will be going through the different reactor types and how we move the heat around. Radiation shielding is going to be a key aspect. I might mention that under the Exploration Technology Program we have a separate program thrust that deals with shielding. That's being managed by our Materials and Structures Division.

Now, given the fact that back in January the President submitted this budget that included the $179.4 million for exploration technologies with $11 Million for nuclear propulsion, what are we going to do, given that we have no money in FY90?

Well, we kicked it around in several meetings (Figure 37 & 38) and decided that we should at least assemble what we can of the requirements. We will go out and talk to the people at MASE, the Mission Analysis and Systems Engineering group at the Johnson Space Center, and find out what assumptions they and the supporting centers have made about nuclear propulsion, what the requirements were on NERVA, and what the requirements were on SP100, because that's the current ongoing space nuclear system in this country. Particularly, we should learn where we are in safety, because the safety philosophy is different from NERVA to SP100.

In the days of SP100 and the SNAP-10A system, the idea was "burnup on reentry." People were looking at things as far-out of shooting cannons up the nozzle of NERVA to blow it apart to ensure that it would burn up on reentry; now we are looking at "intact reentry."

So we decided to pull together these workshops (Figure 39) and to assemble a data base on the various concepts; but we wanted to do more than just simply bring everybody in and go through the advocacy. We decided to put together a technology review panel, which will try to evaluate these things, separate the facts from the advocacy, and try to get the advocates evenly weighed and on a level playing field. Again there is no intention of making any decisions in terms of concepts, but rather to determine what work is needed on each concept to bring them up to enough design maturity that we can make intelligent decisions later on in the decade.

Next, we will work on our program and project plans. We have a draft program plan (Figure 40 & 41) called the Thrust Plan, and we are now working on draft project plans. Our goal is to get ready so that, depending on what money comes in in FY91, we can hit the deck running with strategy, and statements of work, and we would have our plans in place. We are going to do that by assembling the data bases and by holding these workshops; and I think a key part of the process is developing advocacy charts and papers.

Realize that we are going to have to sell this program and sell it and sell it; there are going to be reviews on top of reviews. Immediately after the 90 day study was completed the National Research Council (NRC) met and reviewed it, and we will have our own internal reviews, and our own Space Systems and Technology Advisory Committee (SSTAC). The National Research Council (NRC) has an Aeronautics and Space Engineering Board (ASEB) that will be reviewing it, and there will probably be Congressional reviews.

So these workshops will help us, through meeting you and seeing your charts, to put together a coherent total story on nuclear propulsion. We are going to use these workshops to put together that data base, to help us identify the technical issues and to help us define our program. Again, we are using the technology review panels to evaluate the data and they will be meeting with us again in September. Then we hope later in the fall to have a meeting with all of you give you feedback on where all of this is going.

Just to recap, our philosophy is developing nuclear propulsion technology for space missions and that means going into the critical subsystem and components. We are going to look at real system performance and operating characteristics, and we are going to look at specific space missions such as going to Mars. And we are going to have to have a program that's environmentally acceptable, that is certainly innovative, and that is driven by the mission requirements. It's going to be focused on critical propulsion components, including the reactor and the rest of it. We are certainly going to have to spend a lot of time wrestling with the philosophy on how we verify the system, and there are a number of requirements that are going to have to met, chief among them being safety.

I always like the quote attributed to Glenn Seaborg (Figure 42). He said what we are attempting to make is a flyable compact reactor, not much bigger than a desk, which would produce the power of Hoover Dam from a cold-start in a matter of minutes. So I always thought Seaborg had that pretty well in focus.

I think there is enough in this to keep us all busy, and again I remind everybody the 11th commandment of the nuclear community, we are not making decisions in these workshops, so "Thou Shalt Not Speak Ill of Another Nuclear Program." And with that, let's go to Mars.

NASA

PRESIDENT BUSH AND THE SPACE EXPLORATION INITIATIVE

Speech - July 20, 1989	Commitment and Direction	Space Station, the Moon, then Mars
Speech - February 2, 1990	The Goal	To place American on Mars
Decision - February 21, 1990	Policy Guidance	Technology and mission architectures
Announcement - March 30, 1990	International Guidance	Exploratory dialogue with other nations
FY 1991 budget request to Congress	Budget Request	Funds to begin preliminary planning
Speech - May 11, 1990	Timetable	Mars: before July 20, 2019

Office of Aeronautics, Exploration and Technology

Figure 1

ORIGINAL PAGE IS
OF POOR QUALITY

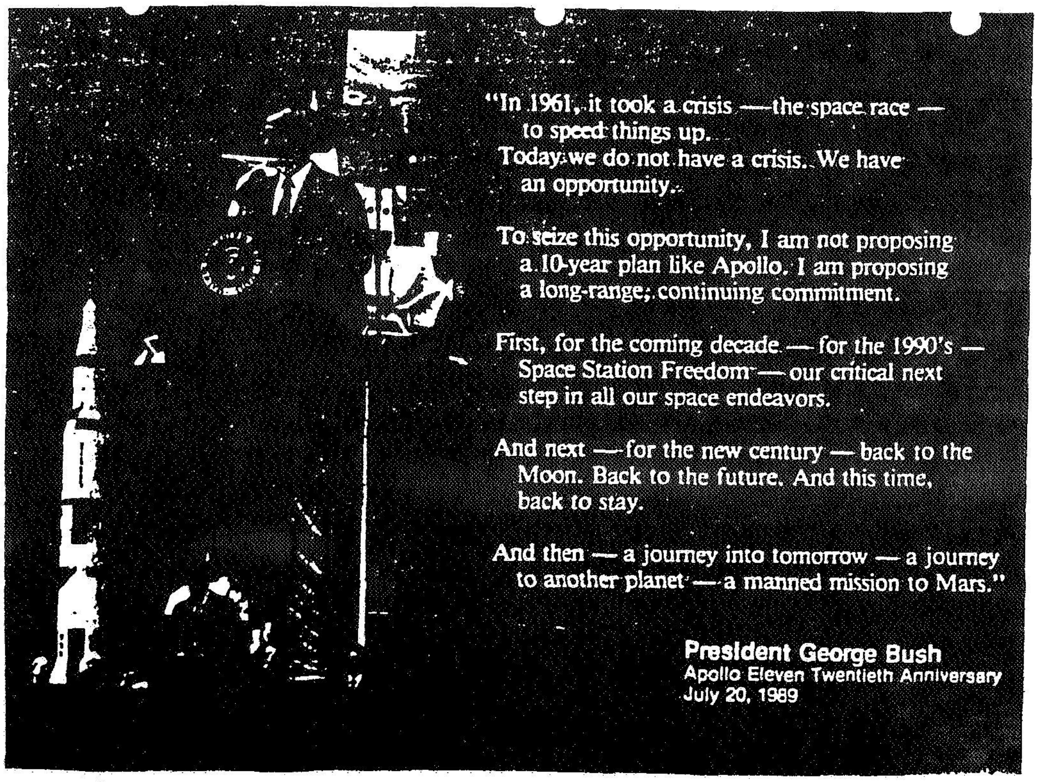


Figure 2

NASA

THE PRESIDENT STATES THE GOAL

"Our goal: To place Americans on Mars—and to do it within the working lifetimes of scientists and engineers who will be recruited for the effort today. And just as Jefferson sent Lewis and Clark to open the continent, our commitment to the Moon/Mars initiative will open the Universe. It's the opportunity of a lifetime—and offers a lifetime of opportunity."

President George Bush
Remarks at the University of Tennessee
February 2, 1990

Office of Aeronautics, Exploration and Technology

Figure 3

NASA

PRESIDENTIAL DECISION ON THE SPACE EXPLORATION INITIATIVE

On February 16, 1990 President Bush approved policy for the Space Exploration Initiative:

- Initiative will include both Lunar and Mars program elements, as well as robotic science missions
- Near-term focus will be on technology development
  - Search for new/innovative approaches and technology
  - Investment in high leverage innovative technologies with potential to make a major impact on cost, schedule, and/or performance
  - In parallel with mission, concept, and system analysis studies
- Selection of a baseline program architecture will occur after several years of defining two or more reference architectures while developing and demonstrating broad technologies
- NASA will be the principal implementing agency while DOD and DOE also will have major roles in technology development and concept definition. The National Space Council will coordinate the development of an implementation strategy by the three agencies

Office of Aeronautics, Exploration and Technology

Figure 4

NATIONAL SPACE POLICY – GOALS

On November 2, 1989, the President approved a national space policy that updates and reaffirms U.S. goals and activities in space.

- Strengthen the security of the United States
- Obtain scientific, technological, and economic benefits
- Encourage private sector investment
- Promote international cooperative activities
- Maintain freedom of space for all activities
- Expand human presence and activity beyond Earth orbit into the solar system

Figure 5

WHY ARE WE GOING TO MARS?

To fulfill the human imperative to explore

To understand planetary evolution

To enhance our understanding of life in the universe and find out if life once existed on Mars

To improve our country's technological competitiveness

To continue America's journey into space

Carry out the National Space Policy goal of expanding human presence and activity beyond Earth orbit into the solar system

Figure 6

TOP

KEY TECHNOLOGIES NEEDED FOR HUMAN EXPLORATION OF THE MOON AND MARS

- REGENERATIVE LIFE SUPPORT SYSTEMS
- AEROBRAKING
- ADVANCED CRYOGENIC HYDROGEN-OXYGEN ENGINES
- SURFACE NUCLEAR POWER SYSTEMS
- IN SITU RESOURCE UTILIZATION
- RADIATION PROTECTION
- NUCLEAR PROPULSION

Figure 7

NASA EXPLORATION TECHNOLOGY PROGRAM OAST

PROPULSION, POWER AND ENERGY

- The Exploration Technology Program is a program through which NASA will develop a broad set of technologies to enable future decisions on and development of future space exploration missions. The Exploration Technology Program is a critically-needed, focused technology program that will strengthen the technological foundation of the civil space program and the nation's leadership to go forward with ambitious future solar system exploration missions.

- The Exploration Technology Program is organized into eight technology areas:

Space Transporation	Lunar and Mars Science
In-Space Operations	Information Systems
Surface Operations	Automation
Human Support	Nuclear Propulsion

Figure 8

NASA **EXPLORATION TECHNOLOGY PROGRAM** OAST

PROPULSION, POWER AND ENERGY

NUCLEAR PROPULSION

The technology developed in the nuclear propulsion program area will address multiple approaches to applying space nuclear power systems to the improvement of mission performance for human missions to Mars

BASIS OF THE FY 1991 BUDGET ESTIMATE

Research will be started in nuclear thermal rocket propulsion technologies, including both solid core and gaseous core nuclear system concepts, capable of long-life and multiple starts, for future piloted mission to Mars applications, and in nuclear electric propulsion technologies, including both nuclear reactor systems technologies, advanced low-mass radiator and power management systems, and in high-power long-life electric thrusters for piloted missions to Mars.

Figure 9

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

THRUST BREAKDOWN STRUCTURE

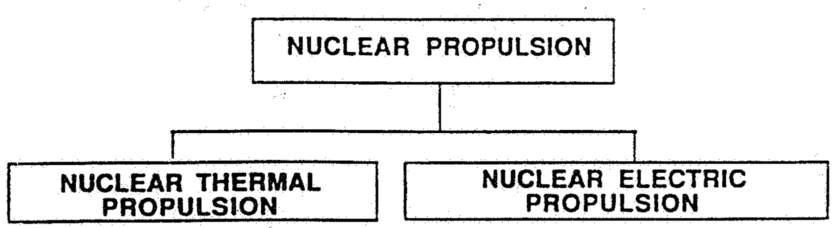
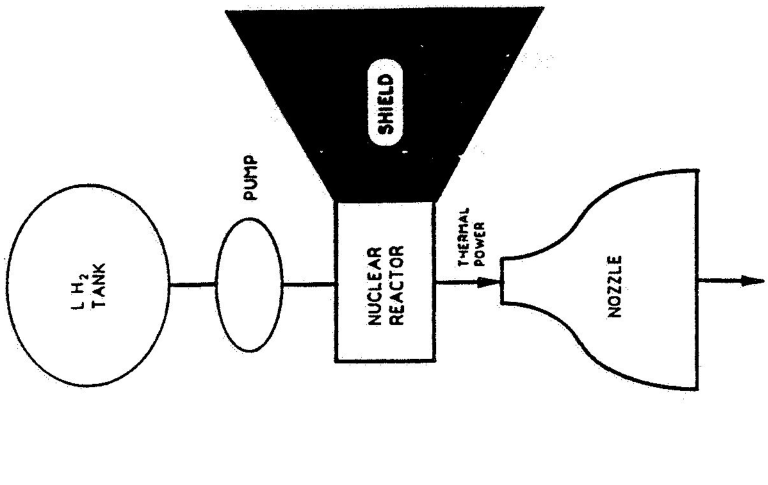


Figure 10



SCHEMATIC DIAGRAM OF A GENERALIZED NUCLEAR THERMAL PROPULSION SYSTEM

Figure 11

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

EXECUTIVE SUMMARY

THRUST GOALS

- **Develop the technologies required to apply space nuclear propulsion systems to improve the mission performance for human missions to Mars**
- **Identify and develop at least one space nuclear propulsion system that, alone or in combination with other propulsion systems meets the propulsion requirements for piloted and cargo missions to Mars (including unmanned precursor missions) and for which technical feasibility issues have been resolved**

Figure 12

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

EXECUTIVE SUMMARY

OBJECTIVES

- Develop safe advanced nuclear propulsion system concepts that are responsive to SEI requirements (including vehicle/stage considerations)
- Demonstrate component, subsystem, and systems technologies for advanced nuclear propulsion systems
- Validate design analysis techniques and develop a technology base in the required disciplines

Figure 13

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

EXECUTIVE SUMMARY

PROJECT-LEVEL GOALS

- Develop the nuclear thermal rocket propulsion technologies, capable of long-life and multiple starts, for future piloted and cargo missions to Mars, including unmanned precursor missions
- Develop the nuclear electric propulsion technologies, including nuclear reactor systems technologies, advanced low-mass radiator and power management systems, and high-power, long-life electric thrusters for piloted and cargo missions to Mars, including unmanned precursor missions

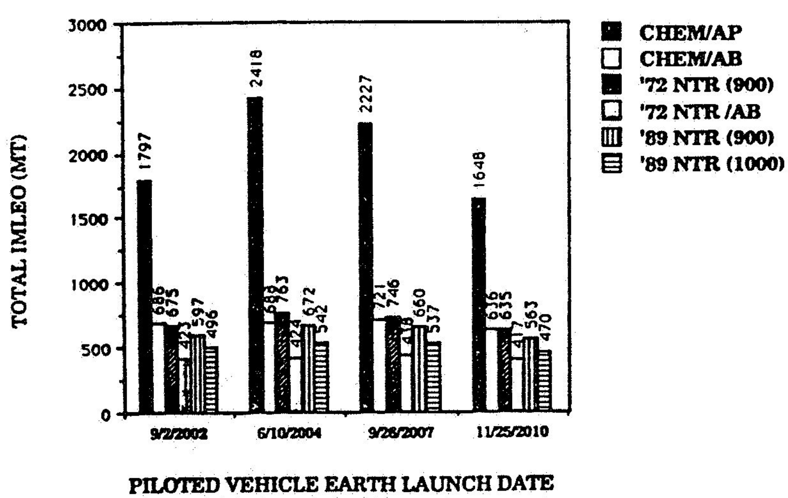
Figure 14

NASA **NUCLEAR PROPULSION THRUST PLAN** OAET

PROPULSION, POWER & ENERGY

- **Develop safe, reliable, high-performance nuclear propulsion technology for exploration of the Solar System**
- **Develop a consensus on the safe use of nuclear propulsion in order to achieve public acceptance**
- **Strengthen and extend the propulsion technology foundation of the civil space program so that a new, higher technology plateau will be established for future propulsion programs**
- **Broaden participation of universities to enhance the scientific and technical educational level of the U. S.**
- **Coordinate with DOE, DoD and their labs and contractors to minimize duplication and maximize use of existing resources**
- **Implement through a joint NASA/DOE/DoD/Industry/University team approach**
- **Carry out a phased and focused technology development program with clearly defined technical objectives in order to identify early the best approach(es)**
- **Maintain a flexible design approach to accommodate changes**

Figure 15



MARS EXPEDITION CASE - IMLEO SENSITIVITY TO LAUNCH OPPORTUNITY

Figure 16

NASA

IMLEO SENSITIVITY TO LAUNCH OPPORTUNITY
MARS EXPEDITION CASE

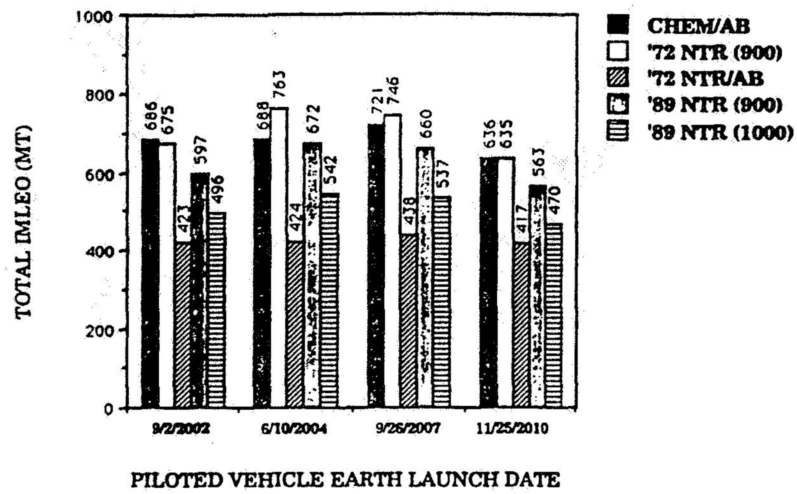
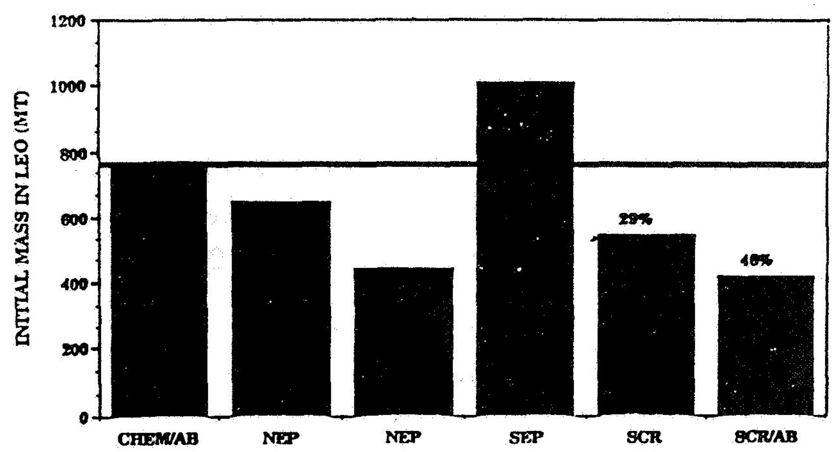


Figure 17

NASA

PROPULSION PERFORMANCE COMPARISON
NEP, SEP, AND SCR PILOTED MARS MISSION



	CHEM/AB	NEP	NEP	SEP	SCR	SCR/AB
Alpha (kg/kWe)	N/A	15	10	10	N/A	N/A
Isp (s)	475	5000	5000	5000	900	900
Trip Time (d)	434	650	650	650	434	434

Figure 18

NASA

PROPULSION PERFORMANCE COMPARISON
SCR AND GCR PILOTED MARS MISSIONS, QUICK TRIPS

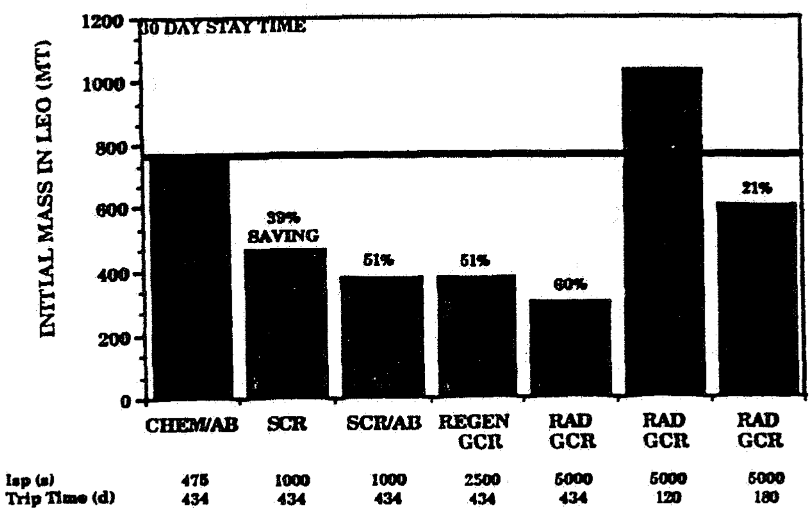


Figure 19

NTR MARS PERFORMANCE
THRUST/WEIGHT AND ISP VARIATIONS

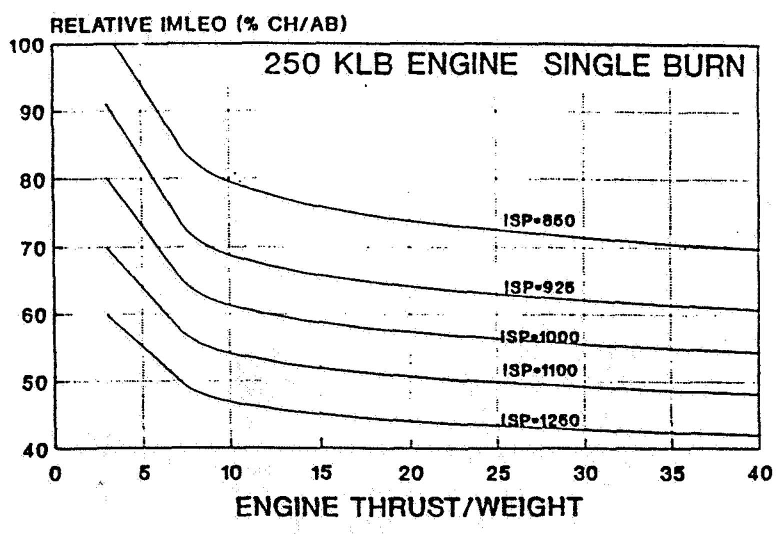


Figure 20

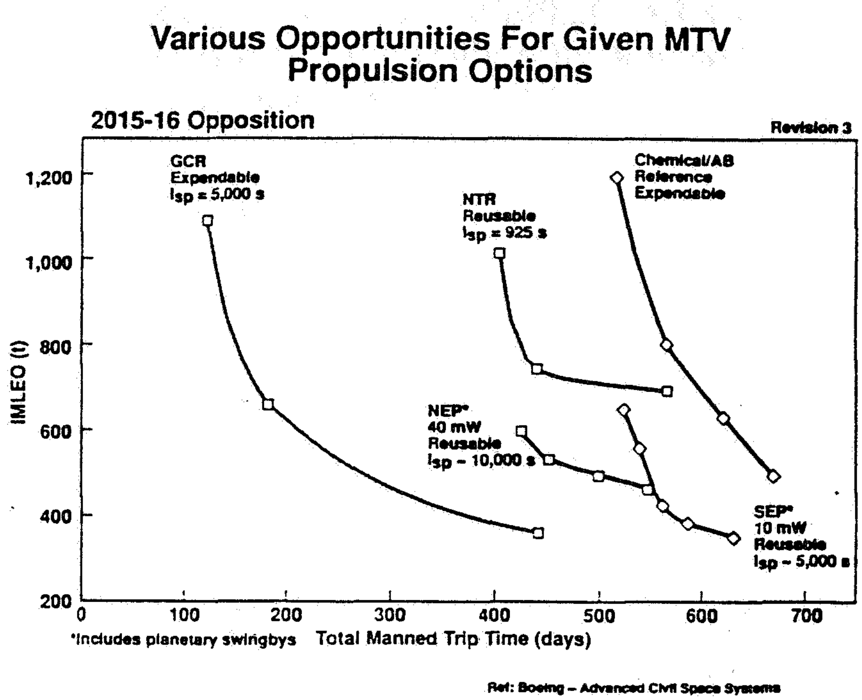


Figure 21

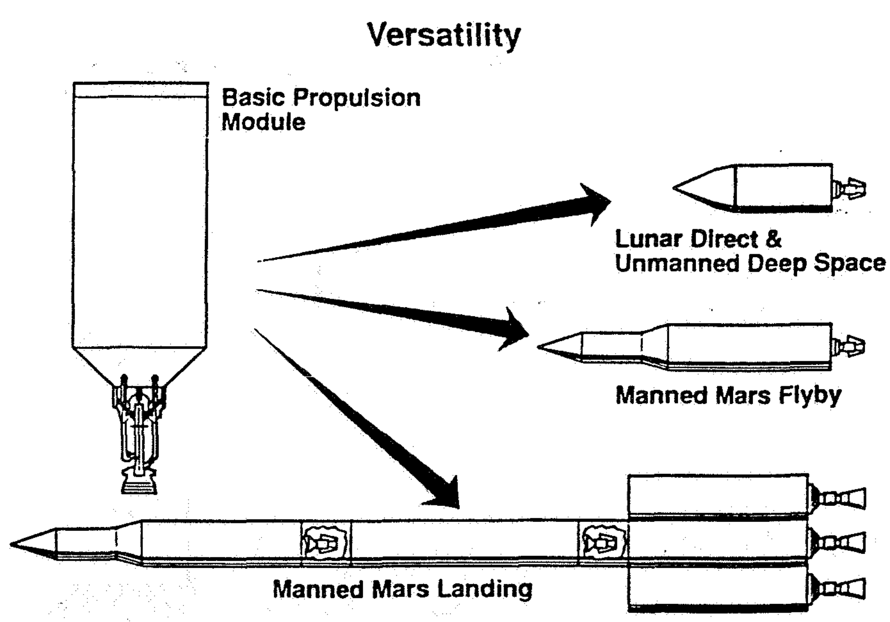


Figure 22

EXTRA-TERRESTRIAL PROPELLANT LANDER/HOPPER/ASCENT VEHICLE (DIRECT FISSION-THERMAL PROPULSION)

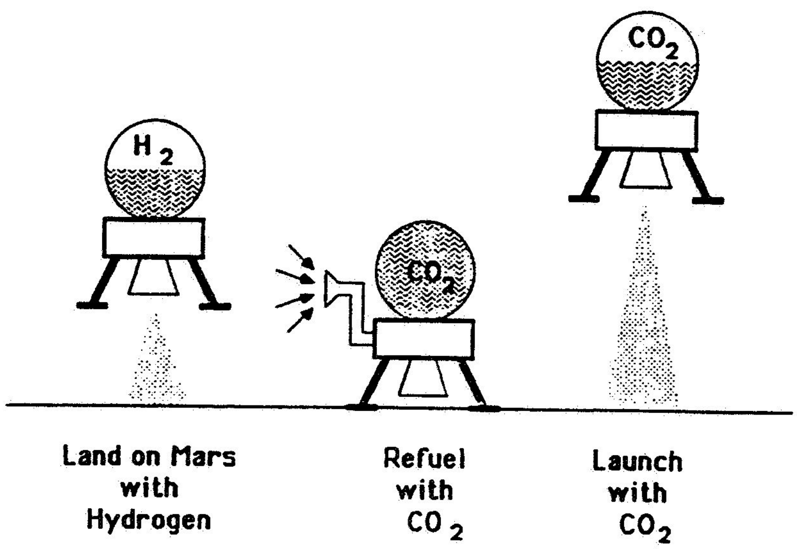


Figure 23

NERVA TECHNOLOGY HAS SYNERGISTIC APPLICATIONS

Steady-State Power

- 10's of MWe for electric propulsion

Direct thermal propulsion

- 15,000 to 250,000 pounds of thrust

Dual Power Systems

- High direct thrust (e.g., 75,000 pounds) plus low electric propulsion (e.g., 1MWe)

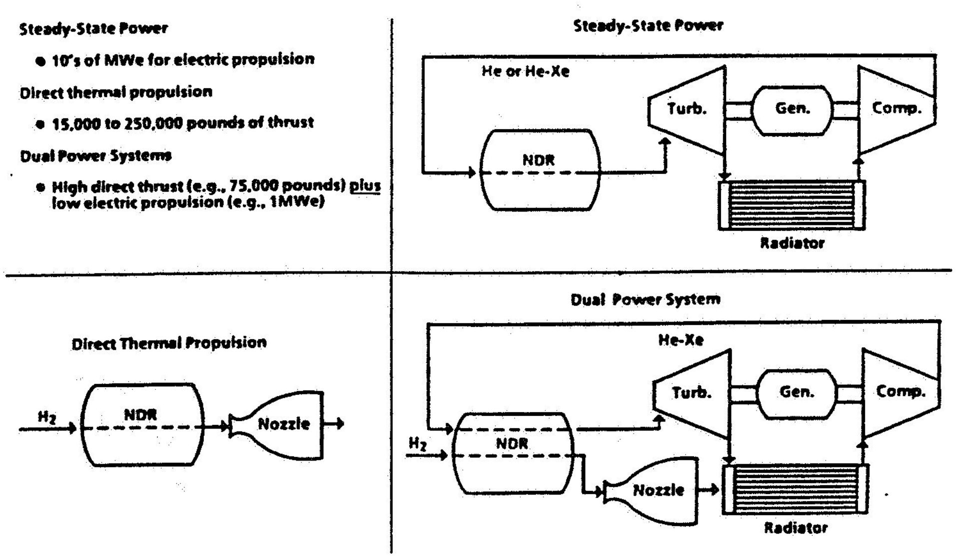


Figure 24

Nuclear Rocket Program

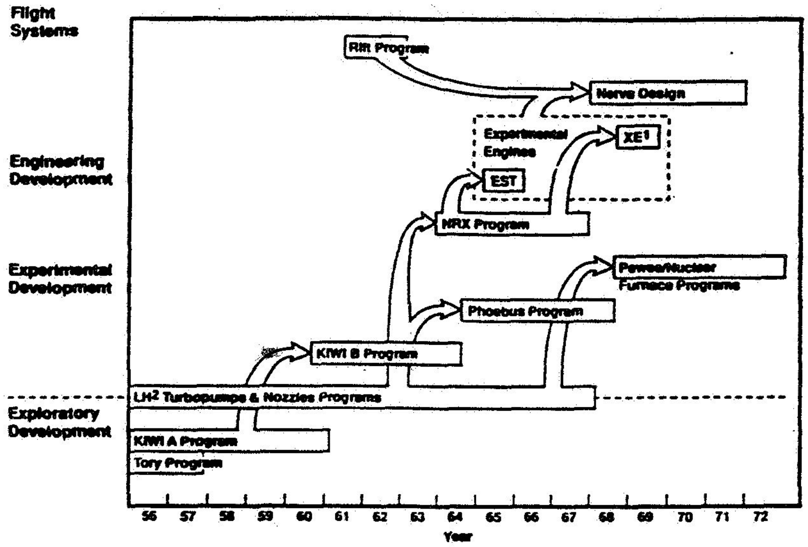


Figure 25

Evolution of Rover Reactors

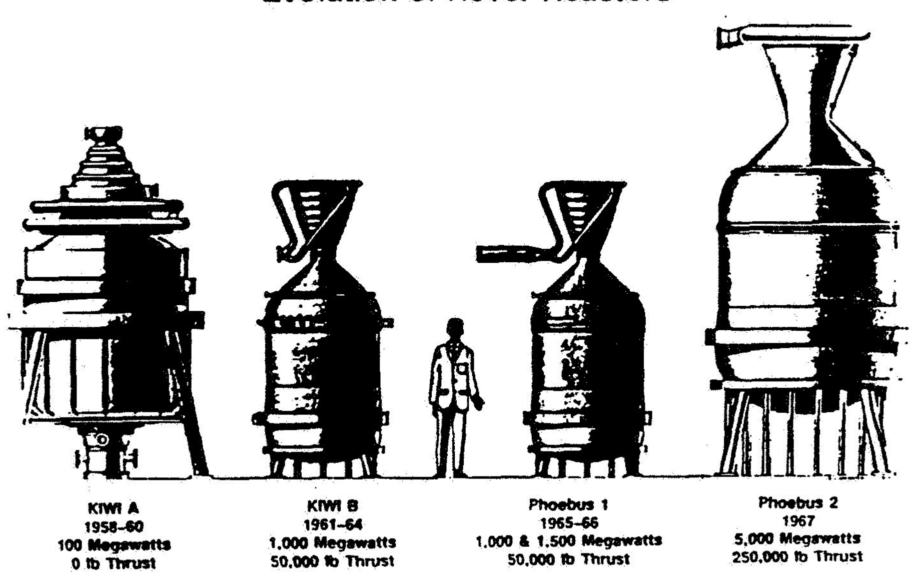


Figure 26

NERVA Flight Engine Configuration

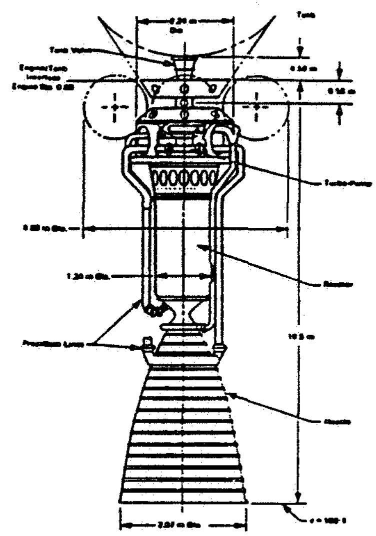
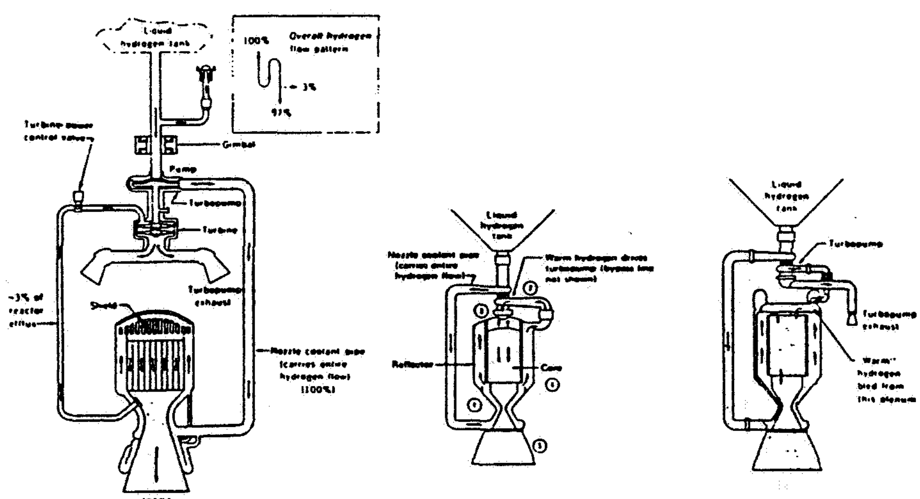


Figure 27

MODES OF NUCLEAR ENGINE OPERATION



Hot Bleed Cycle Full-Flow Cycle Cold Bleed Cycle

Figure 28

NERVA/Rover Reactor System Test Sequence

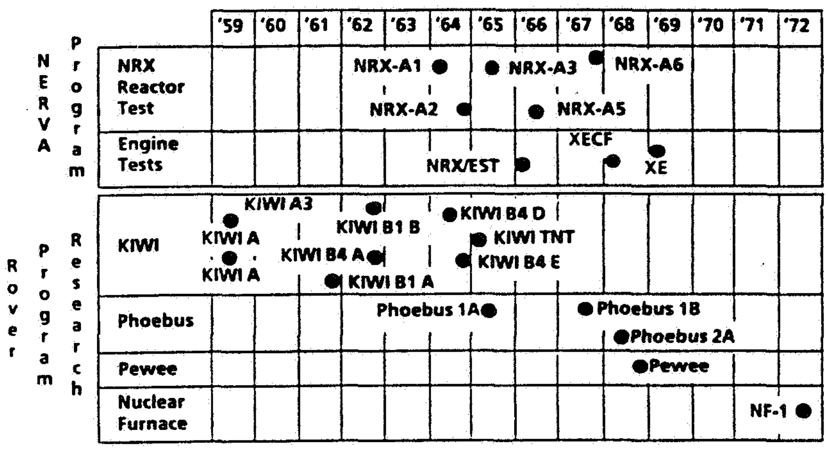


Figure 29

SMALL/ADVANCED NUCLEAR ROCKET ENGINE
(SNRE/ANRE - LANL/INEL)

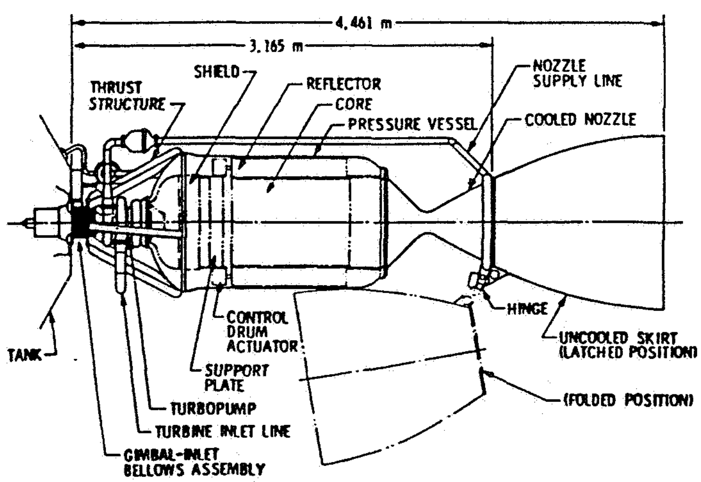


Figure 30

PARTICLE BED REACTOR DESIGN

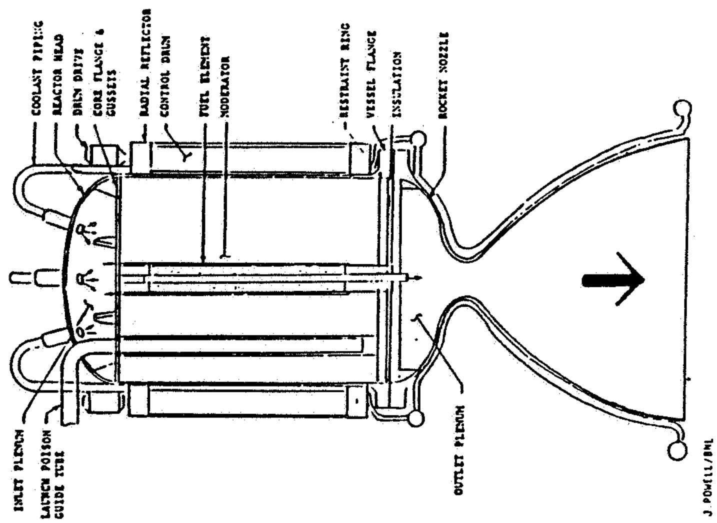
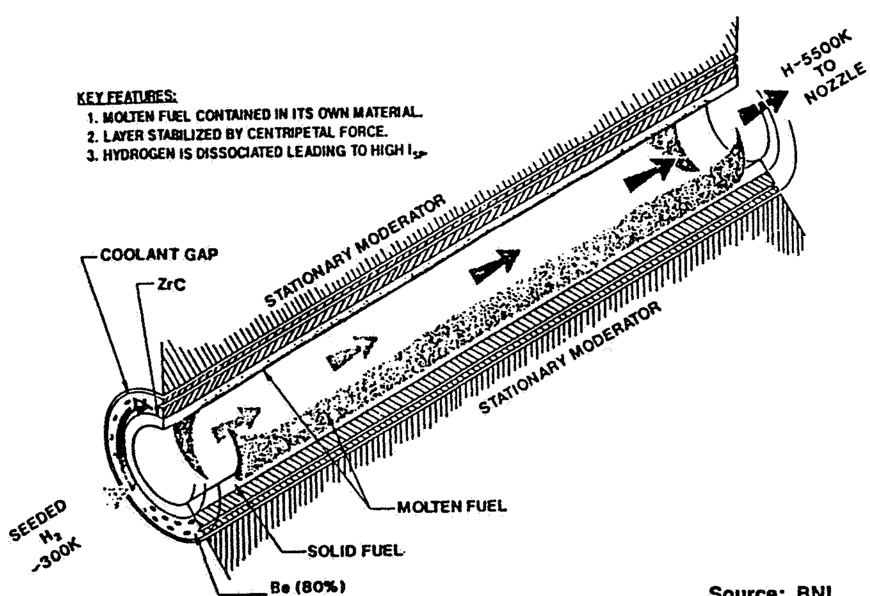


Figure 31

LIQUID ANNULAR REACTOR SYSTEM



Source: BNL

Figure 32

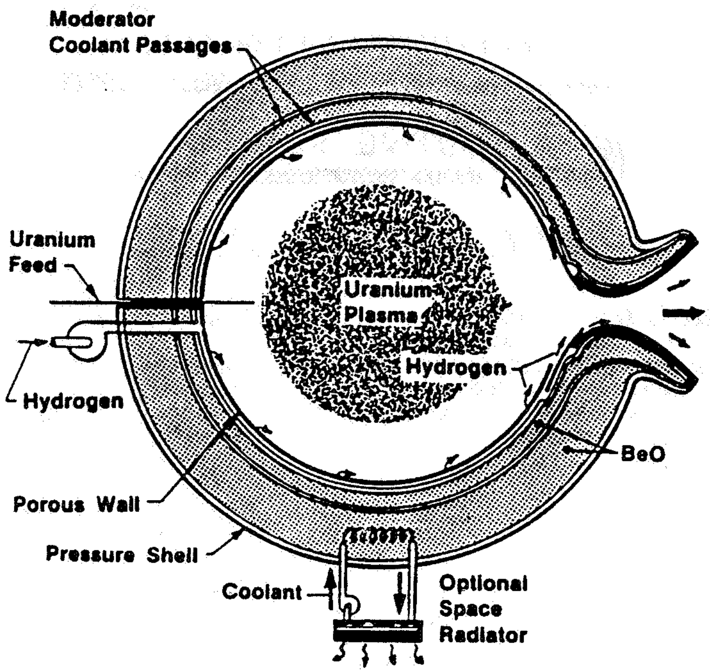


Figure 33

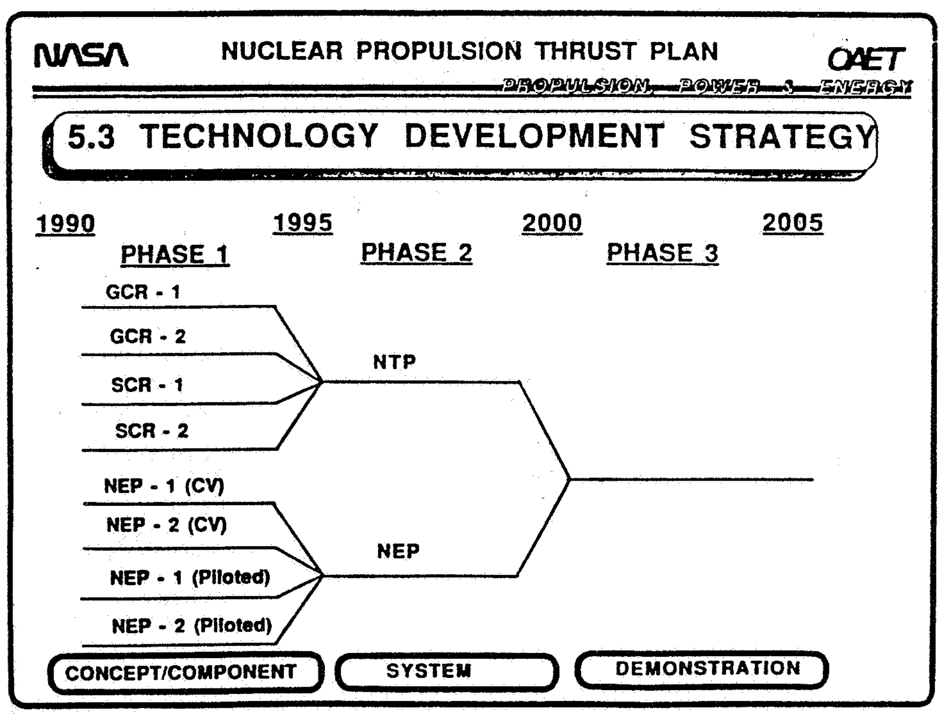


Figure 34

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

EXECUTIVE SUMMARY

KEY TECHNICAL ISSUES

Safety/safeguards/QA
  (during all program phases)
Qualification/acceptance test strat.
Reliability and fault tolerance
High Performance engines
  (including reactors)
Reusability/restart capability
Reactor Fuel
Structural Aspects
Turbomachinery
Vessels/Nozzles
Pumps/Valves
Diagnostic Capability
Control Systems (neutronics/
  I&C)

Power Processing Units
  (NEP)
Thrusters (NEP)
Space operations
  - radiation shielding
  - design criteria for in-space
    operation and
    maintenance
Propellants/Prop. handling
Thermal hydraulics
Thermal Management
Materials
Lifetime
Mass/Volume Limitations
In-situ Prop. Utilization

Figure 35

NASA NUCLEAR PROPULSION THRUST PLAN OAET

PROPULSION, POWER & ENERGY

4.2 CONCEPT DEVELOPMENT SCOPE

GAS CORE ROCKETS

Reactor type
Heat transport and rejection
Safety Systems
Radiation Shielding
Control
Pressure vessel
Turbopumps
Nozzle
Thrust Structure

SOLID CORE ROCKETS

Reactor type
Heat transport and rejection
Safety systems
Radiation shielding
Control
Pressure vessel
Turbopumps
Nozzle
Thrust structure

NEP

Reactor type
Heat transport and
  rejection
Power conversion
  unit
Safety systems
Radiation shielding
Control
Pressure vessel
Turbopumps
Power processing
  unit (PPU)
Thrusters
Thrust structure

Figure 36

NASA EXPLORATION TECHNOLOGY PROGRAM OAST

NUCLEAR PROPULSION

WHAT NEEDS TO BE DONE IN FY 1990

- ASSEMBLE "REQUIREMENTS"
  - Mission Study Assumptions (workshop)
  - NERVA Requirements
  - SP-100 Requirements (especially safety)
- ASSEMBLE DATA BASE ON CONCEPTS
  - Workshops on GCR, SCR and NEP
  - Publish report (data base)
- DEVELOP PROGRAM AND PROJECT PLANS
  - Prepare SOW for Contracts
  - Prepare procurement packages

Figure 37

NASA NUCLEAR PROPULSION OAET

FY 1990 PROGRAM STRATEGY

Objective: Develop the FY 1991 program, including
- procurement strategy
- statements of work
- thrust/project plans

Implementation: Assemble data base

Hold Workshop(s)

Figure 38

NASA **NUCLEAR PROPULSION** OAET

NUCLEAR PROPULSION WORKSHOP

Objectives

- Assemble data base
- Identify technical issues
- Provide input for FY 1991 studies

Approach

- Hold workshop covering
  - Mission studies
  - Safety
  - GCR/SCR/NEP
- Collect data and have technical "tiger team" evaluate data
- Issue evaluated data report and workshop summary

Figure 39

NASA **Nuclear Propulsion Thrust** OAET

PROPULSION, POWER AND ENERGY

NUCLEAR PROPULSION PROGRAM PHILOSOPHY

Objective:

- The development of nuclear propulsion system technology for space missions
- The development of critical subsystem and component technology
- Evaluation of real system performance and operating characteristics
- The evolution of a propulsion system concept that will meet the objectives of specific space missions when firm objectives are identified
- The development of a sound technical system verification approach which is environmentally and programmatically acceptable
- Pursuit of innovative and advanced technologies with significant mission advantages

IN SUMMARY, A MISSION/REQUIREMENTS DRIVEN TECHNOLOGY PROGRAM IS PLANNED

Figure 40

NASA **Nuclear Propulsion Thrust** OAET

PROPULSION, POWER AND ENERGY

NUCLEAR PROPULSION DEVELOPMENT PHILOSOPHY

Principal thrust directed to the development of critical propulsion components and subsystems that significantly affect propulsion system characteristics:

- Reactor subsystem
- Thrusters (for NEP)
- Nozzle (for NTP)
- Turbopump assembly (for NTP)
- Thrust vector control system (for NTP)
- Power system (for NTP)
- Power processor (for NEP)
- Control system

Development of a verification approach that includes components, subsystems and systems, and addresses:

- Analysis
- Simulation
- Test

Requirements priority in order:

- High reliability and ground/flight safety
- Development cost/risk
- Performance/Weight
- Remote maintenance (robotics)

Figure 41

What we are attempting to make is a flyable compact reactor, not much bigger than an office desk, that will produce the power of Hoover Dam from a cold start in a matter of minutes

– Dr. Glenn T. Seaborg
Chairman
Atomic Energy Commission

Figure 42

DOE NUCLEAR PROPULSION PROGRAM
OPENING REMARKS

Earl Walquist
Acting Associate Deputy Assistant Secretary
for Space Defense Energy Projects
Department of Energy

I just want to add just a couple of thoughts to what Gary has said. I want to reemphasize that DOE intends to be an important supporter of NASA in this endeavor of nuclear propulsion. Looking back at the 1991 budget time-frame, when the President gave his speech, it was too late for DOE to respond in the 1991 cycle. But there is a major issue going in the 1992 cycle, for which the budgets are already in process. If NASA is successful in maintaining SEI in their 1991 budget, then I think DOE will become an important participant in funding in the 1992 cycle.

DOE's view is that this is a national priority, and if the Congress adopts it and the President continues to support it, DOE's intent is to support NASA and to be a co-equal player in making it happen.

There are lots of issues that have to be sorted out, and technologies: what to do and how to accomplish it. But when it comes to these kinds of programs for other agencies, DOE does not view itself as the technology pusher, but rather as a supporter, and to be active in the development through the use of its laboratories, and funding support. So the major lead for the requirements comes from NASA, and it will not come from DOE, though DOE will lay on certain safety requirements, as issues that they will want to see adopted and pursued.

As many of you are aware, when you talk about nuclear electric propulsion, at low power end, one of the primary missions considered for SP-100 was nuclear electric propulsion. In fact, that sort of reference mission for SP-100 has been talked about for some time, since SEI came up with nuclear electric propulsion as an option. When you move SP-100 into the higher megawatts, it has to be looked at by viewing other technology. I guess the thought that I would like to leave here, is that we are excited about the program in DOE, and some of us are pushing our management harder than they want to be pushed, but there are major decisions going on.

As you are aware, Congress wasn't too kind to SEI in the initial 1991 budget; they tried to zero everything that had anything to do with SEI. Probably it's a bargaining position with the President. And they also did it in NASA. But Congress didn't go to the other agencies and zero everything in relationship to SEI. So when you get into joint funding programs, sometimes it helps you to maintain joint programs because if you get an enemy in one place, you can sort of overcome them, with a friend in another place. So we are excited about it and we are hopeful that we can we, in its wisdom, doesn't know where other things are, they work closely with Gary in trying to support him in laying out the development plans.

How it all comes out, none of us knows for sure, but I guess I am optimistic that we will see something go forward in the near term. And, as Gary commented, one of the exciting things to me about the SEI is that it probably could do more to excite or rejuvenate technology and the interest of the young people growing up than anything else that we could start on.

I was at a community meeting one evening. I came in late and commented that I had been in a meeting talking about going to Mars and they didn't want to talk about anything else. They wanted to talk about how are we going to do it and how it is going to work.

Young people can be excited about being scientists and engineers with this kind of an endeavor. They don't get so excited about technology that is just for weapons or other things, but there is something about this mission to Mars. I think we have an opportunity here to create something that will have a legacy in many respects for mankind. So let's work together and make it happen.

N92-11090

NUCLEAR THERMAL PROPULSION WORKSHOP OVERVIEW

John S. Clark
Workshop Chairman
NASA Lewis Research Center
Cleveland, OH

In the October/November issue of Air and Space Magazine (the quarterly magazine of the Smithsonian Air and Space Museum) the cover story was "Destination Mars, What Kind Of Rockets Will Get Us There." I think this article talks about why we are here today (Figure 1). We are here to try to figure out how to use nuclear propulsion to accomplish that mission, and we appreciate the help that we will be receiving from all of you.

I have a very detailed purpose statement in the handout (Figure 2). I am not going to read the words for you, but the bottom line is included in the last paragraph, to assess the state-of-the-art, to try to identify which of those concepts that have been proposed have the most benefit for the manned mission to Mars, to identify the technologies that need to be developed, to lay out some first-order plans for those technologies, and to try to get a first-order cost estimate, and from there to put together our project plan.

There is also included in the handout a listing of the members of the steering committee (Figure 3). You have met Gary Bennett, Earl Wahlquist, and Tom Miller, and Roger Lenard will be joining us. There are also a number of ex-officio members of the steering committee, including Franklin Chang-Diaz, who is an active astronaut at Johnson at this time; he has been included to bring in the astronaut safety aspects.

Figure 4 tries to show what we are trying to accomplish, and how we are going to do it. Back about the first of May, we got together in Washington and agreed upon an approach that looks very similar to the final approach that we are using for these workshops. We identified a large number of concepts that are candidates for this kind of a mission to Mars, and we tried to identify an appropriate person who could be a spokesperson (or Concept Focal Point - CFP) for that concept at these workshops. At the same time, we tried to define some requirements for the mission; Stan Borowski will talk about that baseline reference mission to Mars in his presentation, which will follow this one.

Based on those common requirements then, each of the concept focal points were to address their concept and how to do the mission, the kinds of technologies that would be required to perform that kind of a mission in terms of lifetime, endurance, reliability, safety and all of those things.

We put together Technology Review Panels (Figure 8-12) that are a national community of experts, if you will; they are here and will be sitting in on the parallel sessions, evaluating each of the concepts based on the four criteria: cost, safety, benefit to the mission, and technical risk.

Each of the concept focal points will present a brief summary of their concept, something on how that concept would perform on the mission, what the critical tests are, schedule, milestones, costs, and facilities.

The technology review panels then are going to use that information, prepare recommendations, and make a final presentation to the steering committee in September.

This is a quick summary of how we are going to get through the next three days (Figure 5). All day today and through 9:15 a.m. tomorrow, we will be meeting in this plenary session, where each of the concept focal points will give a brief summary of their concepts. We will then break into parallel working sessions starting at 9:30 a.m. tomorrow and running through about 10:30 a.m. on Thursday. From 10:30 through lunch the panels will caucus and put together their remarks for a plenary feedback session in the afternoon on Thursday; we should break about 3:30 p.m. on Thursday.

We also have a number of special information presentations (Figure 6) that I included in the agenda and I want to just mention some of them. The first one I have already talked about; Stan Borowski is going to talk about the reference mission from which we will "Delta" each of the other concepts. All of the evaluations will be performed compared to that baseline "reference" mission design.

Stan Gunn from Rocketdyne is going to talk a little bit about some of the things that we can do to NERVA that will upgrade that system for changes in the past 20 years.

Press Layton is going to talk about some dual mode concepts. Tonight at the banquet, Peter Worden will have some remarks. Peter is on the National Space Council. Then Brian Pritchard from NASA Langley will be here. He is involved with some of the Space Station Freedom studies and so he is going to talk about the work that is planned to get us from the space station, in its initial configuration, to the Space Exploration Initiative.

There are a number of other special presentations that I am not going to describe, but they will be of special interest to the panels, but that we felt might not be covered in as much detail by the concept focal points.

Figure 7 is a list of the concept focal points as you have in your agenda . I want to point out on the agenda that Dick Dahlberg from GA called me yesterday and he will not be able to attend.

Dilip Darooka from GE has worked on hybrid propulsion systems and he asked for about ten minutes in our plenary session this morning to present some of that material, so we will do that in place of the "pulsed nuclear" presentation. We also have, in addition to the solid core concepts, some liquid core, gaseous core and one paper by Bruce Reid on the NTP/NEP hybrid systems.

I would like to highlight some the members on the technology review panels. In the mission analysis area (Figure 8), Tim Wickenheiser from NASA Lewis is the panel chairman and Mike Stancati from SAIC is the executive secretary.

Ned Hannum is the chairman of propulsion panel (Figure 9) and the executive secretary for this panel will be Stan Borowski, both from NASA Lewis.

The reactor panel (Figure 10) is chaired by John Dearian from INEL and the executive secretary is Harvey Bloomfield from NASA Lewis.

The advanced development plans panel (Figure 11) is chaired by Steve Howe from Los Alamos and Darrell Baldwin is the executive secretary.

The safety panel (Figure 12) is integrated with the other four panels, with members from the safety panel distributed among the other four. They will be addressing the safety issues in each of those panels and then will caucus at the end of the workshop and will put together their separate report. Buzz Sawyer from NASA Headquarters is the chairman of that panel and Marland Stanley from INEL is the executive secretary.

I would like to emphasize the expected output from the workshop (Figure 13). For each of the concepts, we are looking for the critical test requirements, what needs to be done to develop that concept to a technology readiness level six. As indicated, we are working to technology readiness level six (TRL-6)-full system ground testing complete. We want to identify any safety issues with each of those concepts and we would certainly want to identify the facility requirements. And then once we have looked at all of the different concepts, we will be making a first order comparison based on their performance, the mission benefits, technical risk and a first cut at the development cost to TRL-6. Again, it's not a selection process, we are not trying to "down-select" and we are not trying to eliminate any concepts. We are simply trying to identify technology needs so that we can then put together our project plans.

In the assessment procedure (Figure 14) that's to be used, each of the five panels will be addressing the criterion that are identified. The output from the panels will be a written narrative from everyone in the workshop as well as the technology review panel members. That narrative should include discussions of strengths and weaknesses. And then the technology review panel will be doing a relative ranking and a comparison of each concept to the reference system.

Each of you have in your folder an evaluation worksheet (Figure 15) that we would like you to fill out. I encourage you to start filling those out during the summary sessions today and then to transfer them into your three-ring binder to keep the evaluation sheets together with the proper presentation; otherwise if you wait until the end of the session, at the end of all of the presentations you will not remember your comments. So I encourage you to write your comments as we are going along. Then after each concept has been presented in the parallel session, turn those sheets into the Executive Secretaries. He will collect them and we then will have that information available to us.

I want to talk very briefly about some of the factors that each panel will be looking for. In the mission analysis panel (Figure 16), they are looking for the benefit to the mission, how does it accomplish the mission better than the baseline system. Some of the factors are indicated here, initial mass in lower earth orbit or trip time, and they trade-off against each other. Specific impulse is an important measure of performance, and they will be looking at all of the mission safety and operations aspects. They will also address commonality, if that's appropriate, and we'll need to be thinking about whether or not the concept can be ready for TRL-6 in the 2006 time period. And of course they will be looking for inherent design reliability and/or complexity.

Indicated on the right side of the chart is the very simple scoring system that we will use. We have developed a consistent scoring system where the score of (3) represents the same (in this case) mission benefit or performance as the baseline system. There are two levels of performance less than the baseline, and two levels of performance better than the baseline, so the panel will be making an initial first cut at those kinds of discriminators.

The same approach is to be used in the propulsion technology panel (Figure 17). They will be looking at technical risk for developing the concept, and will ask the concept focal points to try to rate the concept on the technology readiness level scale; I will talk about what that means on my next chart. And then each evaluator will have a chance to decide whether he agrees with that rating or not. The factors that they will be considering are: where the concept really is - how mature is the technology. It's probably a pretty good measure of how much money is needed to develop it to TRL-6.

They will certainly be trying to identify the key feasibility issues and the testing requirements for that concept and this is the primary output that we expect from this panel. They will also be addressing integration issues.

NASA Technology readiness levels are defined in Figure 18. Again, this project is intended to go through Technology Readiness Level-6, which is a system demonstrated in a simulated environment, including lifetime, performance, and system interactions. Level-7 is a flight test of that qualified system, so we are trying to determine what needs to be done to get to this point and how much it will cost. You can see the intermediate levels that get us there.

The reactor technology panel evaluation (Figure 19) is very similar to the propulsion panel, in that they will be assessing technical risk and trying to determine where that concept is on the technical readiness level scale. The same kinds of factors will be considered, but primarily inside the reactor, as opposed to outside the reactor in the other components of the propulsion system. The same scoring system is used.

The Advanced Development Planning panel (Figure 20) has the tough part; they have to figure out how much this is all going to cost. This really is a tough one, because the numbers that we have seen so far are all over, and it's pretty much a guessing game. As a first cut, we have asked the CFP's to try to come up with estimates.

Stan Borowski will talk about an initial estimate for the baseline system and we'll try to make our comparisons to that; if it looks like a concept is going to cost more, or a concept is going to cost less, and so forth. The factors that this panel will be considering, are the technology readiness level, the key testing, key feasibility issues, and the testing requirements, the verification issues, safety performance, how we do the simulation, and how we do the testing. A big part of the cost is certainly going to be wrapped up in facilities. Last, but not least, they will develop an overall estimated development cost for that system.

The safety panel, as mentioned (Figure 21), is distributed among the other panels and will be addressing hazard identification and mitigation, safety verification issues, launch safety, inherent control and stability, system refurbishment and disposal (which is certainly an important aspect), orbital assembly, and startup considerations, crew radiation protection (which will be a necessity), redundancy, reliability, and so forth. Also, any other safety issues that need to be considered.

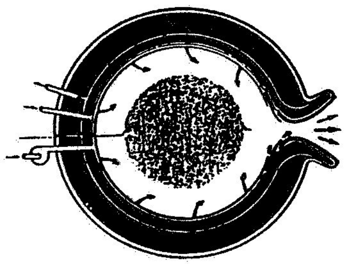
Finally, after we get through with the workshops (Figure 22), the technology review panels or (some smaller subgroup of those technology review panels) will get together to try to clarify some of the issues that have been identified for each of these concepts, and for the nuclear thermal propulsion technology as a whole, and to try to verify some claims that are made by the advocates. We will then do a collation of the written evaluations, and maybe do some simple calculations if that's appropriate. Finally, we will prepare recommendations that will go to the steering committee in the September time period. There will be a workshop proceedings published. And we do intend to provide some feedback to the concept focal points after the steering committee has met.

NASA – Lewis Research Center

WORKSHOP OVERVIEW

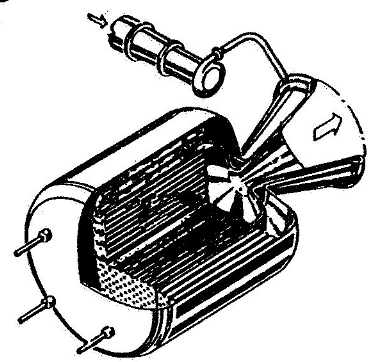
by John S. Clark
Workshop Chairman

NUCLEAR THERMAL PROPULSION WORKSHOP



JULY 10 – 12, 1990

CLEVELAND, OHIO



HOSTED BY: *NASA – Lewis Research Center*

NUCLEAR PROPULSION PROJECT

NPWS1/7-5-90/JSC

Figure 1

NASA – Lewis Research Center

NASA/DOE/DOD
NUCLEAR THERMAL PROPULSION WORKSHOP

STATEMENT OF PURPOSE:

NASA is planning an Exploration Technology Program as part of the Space Exploration Initiative to return U.S. astronauts to the moon, conduct intensive robotic explorations of the moon and Mars, and to conduct a piloted mission to Mars by 2019.

Nuclear Propulsion is one of the key technology thrusts for the human mission to Mars. This workshop will address NTP tech-nologies; a similar workshop was hosted earlier by JPL for NEP technologies.

The purpose of the workshops is to assess the state of the art of nuclear propulsion concepts, assess the potential benefits of the concepts for the mission to Mars, identify critical, enabling technologies, lay-out (first order) technology development plans including facility requirements, and estimate the cost of developing these technologies to "flight-ready" status. The output from the workshops will serve as a data base for nuclear propulsion project planning.

NUCLEAR PROPULSION PROJECT

NPWS2/7-1-90/JSC

Figure 2

NASA LEWIS RESEARCH CENTER

JULY 10, 1990

JOINT NASA/DOE/DoD
NUCLEAR PROPULSION WORKSHOP STEERING COMMITTEE

Members	Organ.:	Loc.:	Responsibilities:
BENNETT,GARY L.	NASA	RP	CHAIRMAN
WALQUIST,EARL J.	DOE	NP-50	
LENARD,ROGER LT.COL.	DoD	SDIO	
MILLER,THOMAS J.	NASA	LERC	EXEC. SEC.
Ex Officio Members:			
VERGA,RICHARD	DOD	SDIO	PROPULSION
BURICK,RICHARD J.	DOE	LANL	NUCLEAR SYSTEMS
COOPER, ROY	DOE	ORNL	NUCLEAR FUELS
MARTINELL,JOHN	DOE	INEL	NUCLEAR SYSTEMS
WALKER,JACK V.	DOE	SNL	NUCLEAR/SAFETY
WIDRIG,DICK	DOE	PNL	SPACE ENERGY SYSTEMS
CHANG-DIAZ, FRANKLIN	NASA	JSC	ASTRONAUT
KELLY,JIM	NASA	JPL	SPACE SYSTEMS
McDONOUGH, GEORGE	NASA	MSFC	DIR.-SCI & ENGR.
WEYERS,VERN	NASA	LERC	DIR,SPACE FLIGHT SYS.
Workshop Chairmen:			
BARNETT,JOHN	NASA	JPL	NEP WORKSHOP
CLARK,JOHN S.	NASA	LERC	NTP WORKSHOP

NUCLEAR PROPULSION PROJECT

Figure 3

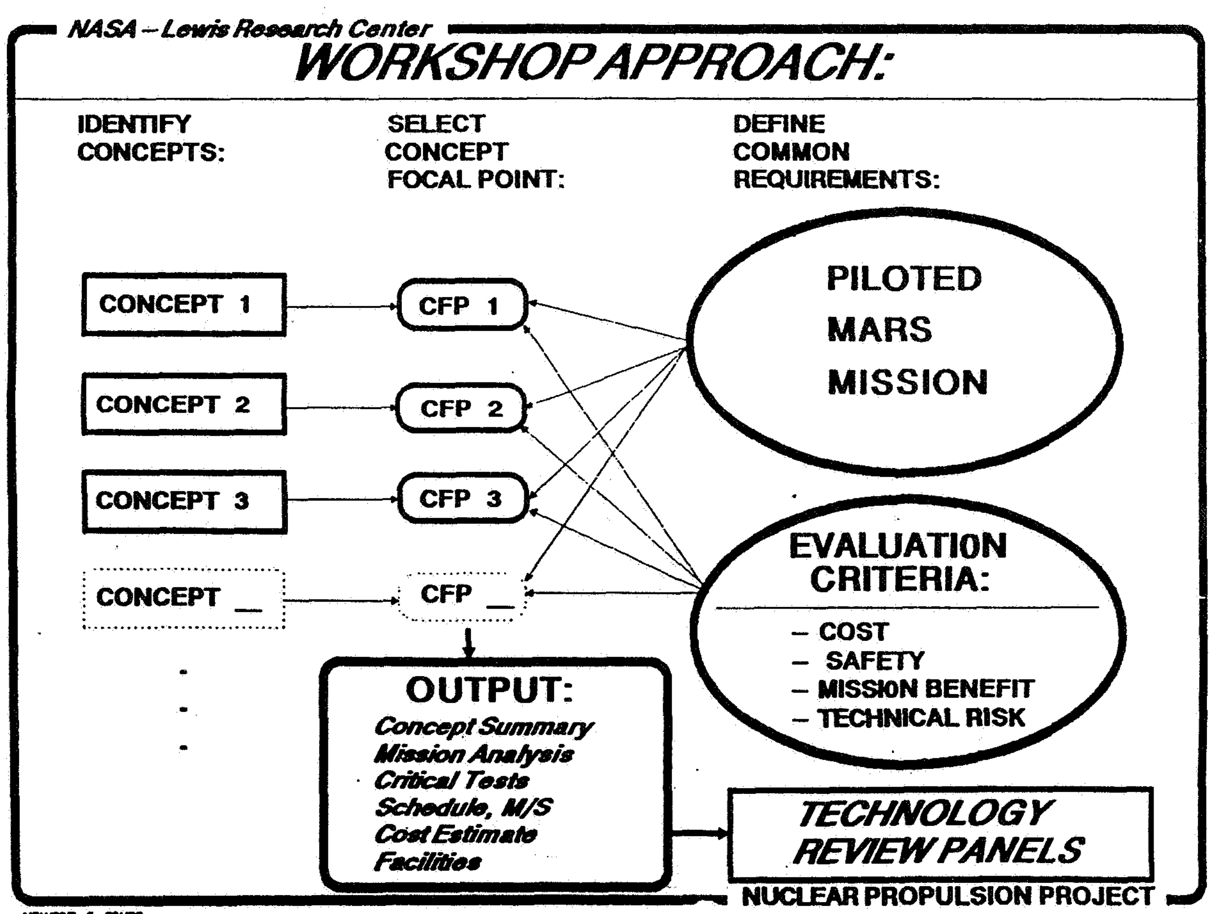


Figure 4

NASA – Lewis Research Center

NTP WORKSHOP MODUS OPERANDI

SUMMARY OF CONCEPTS

TUESDAY: ALL DAY – PLENARY SESSION

WEDNESDAY: UNTIL 9:15 – " "

PARALLEL WORKING PANELS

WEDNESDAY: 9:30 – 5:00

THURSDAY: 8:00 – 10:30 am

FEEDBACK FROM PANELS

THURSDAY: 1:00 – 3:30 pm

NUCLEAR PROPULSION PROJECT

US/7-6-90/JSC

Figure 5

NASA LEWIS RESEARCH CENTER

SPECIAL INFORMATION PRESENTATIONS:

TITLE:	ORG:	PRESENTER:
Tuesday:		
NTP BASELINE DESIGN	LeRC	STAN BOROWSKI
NERVA UPGRADE	ROCKETDYNE	STAN GUNN
DUAL MODE CONCEPTS	CONSULTANT	PRESTON LAYTON
NATL SPACE COUNCIL–REMARKS		PETER WORDEN
SSF TO SEI	NASA–LaRC	BRIAN PRITCHARD
Wednesday:		
UPDATED NERVA TRADE	ROCKETDYNE	MIKE NORTH
NTR MISSION APPLIC.	BOEING	BEN DONAHUE
PROPULSION SYS. NEED	ROCKETDYNE	STAN GUNN
NUCLEAR FUELS STATU	ORNL	ROY COOPER
REACTOR MATERIALS	MSFC	BILL EMRICH
SYSTEM TESTING ISSUE	SVERDRUP	DARRELL BALDWIN
NUCLEAR SAFETY	INEL	DAVE BUDEN
Thursday:		
QUICK TRIPS TO MARS	BOEING	DICK HORNUNG
DISPOSAL METHODS	SAIC	ALAN FRIEDLANDER
SAFETY ISSUES	LeRC	BOB ROHAL
FEEDBACK FROM PANELS		PANEL CHAIRMEN

NUCLEAR PROPULSION PROJECT

Figure 6

CONCEPT FOCAL POINTS: NTP WORKSHOP

Solid Core:			
PIERCE,BILL	WESTINGHOUSE		NERVA DER. – ENABLER II
RAMSTHALER,JACK	DOE	INEL	LOW PRESSURE CORE
LUDEWIG,HANS	DOE	BNL	PBR
KRUGER,GORDON	GE		CERMET
DAHLBERG,RICHARD	GA		PULSED NUCLEAR
ZUBRIN,ROBERT	MARTIN-MARR.		NIMF
HARDY,DICK	ROCKWELL INTL		WIRE CORE
KIRK,BILL	DOE	LANL	ADV. DUMBO
EL-GENK, MOHAMED	SAIC		PELLET BED
WRIGHT,STEVE	DOE	SNL	FOIL REACTOR
Liquid Core:			
LUDEWIG,HANS	DOE	BNL	LIQUID ANNULUS
ANGHAIE,SAMIM	U. FLORIDA		LIQUID CORE
Gaseous Core:			
RAGSDALE,BOB	NASA	LERC	OPEN CYCLE A
LATHAM,TOM	UTRC		LITE BULB – GASEOUS CORE
DIAZ,NILS	U. FLORIDA		OPEN CYCLE B
NTP/NEP Hybrids:			
REID, BRUCE	DOE	PNL	NTP/NEP HYBRIDS

Figure 7

Technology Review Panels: (Both NEP and NTP Workshops)

Mission Analysis:			
WICKENHEISER,TIM	NASA	LERC	PANEL CHAIR
SAWYER,BUZZ	NASA	HQ/QS	CREW SAFETY
DANDINI,VINCE	DOE	SNL	NUCLEAR SAFETY
PERKINS,DAVE	DOD	AFAL	PROPULSION SYSTEMS
COOMES,ED	DOE	PNL	POWER SYSTEMS
AUSTIN,GENE	NASA	MSFC	SPACE TRANS. & EXPLOR.
EVANS, DALLAS	NASA	JSC	LUNAR/MARS EXPLORATION
GEORGE,JEFF	NASA	LERC	NEP SYSTEMS ENGRG
GILLAND,JIM	NASA	LERC	NEP STUDIES, THRUSTER TECH.
HACK,KURT	NASA	LERC	TRAJECTORY ANALYSIS
SAUER,CARL	NASA	JPL	TRAJECTORY ANALYSIS
SUMRALL,PHIL	NASA	MSFC	SPACE EXPLORATION
STANCATI,MIKE	SAIC	ILL	EXEC.SEC.

Figure 8

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Propulsion:			
HANNUM,NED	NASA	LERC	PANEL CHAIR
STANLEY, MARLAND	DOE	INEL	SAFETY
ERCEGOVIC,DAVE	NASA	LERC	SAFETY ASSURANCE
MCDANIEL,PAT	DOD	AFST	SNL-PBR
SCHMIDT, WAYNE	DOD	AFAL	NEP
SULLIVAN,GREG	DOD	SDIO	NT-PBR
GERSTEIN,NORM	DOE	HQ	FORMER NERVA
JOHNSON,BEN	DOE	PNL	INTEGRATION
MERRIGAN,MICHAEL A.	DOE	LANL	HEAT PIPE/HEAT TRANSFER
SCHOENBERG,KURT F.	DOE	LANL	PLASMA PHYSICS
BARNETT,JOHN	NASA	JPL	PROPULSION SYSTEMS
BOROWSKI,STAN	NASA	LERC	NUCLEAR SYSTEMS,EX SEC
CALAGEROUS,JIM	NASA	LERC	HEAT REJECTION
DUDENHOEFER,JIM	NASA	LERC	POWER SYSTEMS
NAININGER,JOE	NASA	LERC	POWER CONV. SYS.
RAGSDALE,BOB	NASA	LERC	GAS CORE
SOVEY,JIM	NASA	LERC	NEP
WINTER,JERRY	NASA	LERC	CSTI SYSTEMS
GERRISH, HAROLD	NASA	MSFC	PROPULSION

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Figure 9

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Reactor:			
DEARIEN,JOHN	DOE	INEL	MMW - PANEL CHAIR
LEE,JIM	DOD	SDIO	SAFETY-MMW
GALLUP,DON	DOE	SNL	REACTOR CONCEPTS,SAFETY
NIEDERAUER,GEORGE	DOE	LANL	SAFETY SP-100
REMP,KERRY	NASA	LERC	SAFETY
HELMS,IRA	CONS.		FORMER NERVA
BHATTACHARYYA,SAM	DOE	ANL	FUEL DEV.
MATTHEWS,R.BRUCE	DOE	LANL	FUEL DEV.
OLSEN,CHUCK	DOE	INEL	FUELS
POWELL,JIM	DOE	BNL	FUELS DEV.
RANKEN,WM.A.	DOE	LANL	THERMIONICS
WALTER,CARL	DOE	LLL	FUELS
BLOOMFIELD,HARVEY	NASA	LERC	SP-100, EXEC. SEC.
EMRICH,BILL	NASA	MSFC	PROPULSION SYS. DESIGN
MONDT,JACK	NASA	JPL	SP-100
SMITH,JOHN M.	NASA	LERC	SP-100
WHITAKER,ANN	NASA	MSFC	ENGR. PHYSICS
KLEIN,ANDY	OREGON	ST.	NUCLEAR ENGR.

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Figure 10

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Advanced Development Plan:			
HOWE,STEVE	DOE	LANL	PANEL CHAIR
ECKART,TED	CONS	AF	LAUNCH SAFETY,VANDENBERG
ALLEN, GEORGE	DOE	SNL	PROJ.MANAGE.,SAFETY
BOHL,DICK	DOE	LANL	SAFETY
BUDEN,DAVE	DOE	INEL	SAFETY, NERVA
KATO,WALTER	DOE	BNL	SAFETY
MARSHALL,AL	DOE	NP-50	SAFETY
RICE,JOHN	DOE	INEL	MMW-SAFETY
ROHAL,BOB	NASA	LERC	SAFETY
WARREN,JOHN	DOE	NP-50	MMW, NUCLEAR SYSTEMS
KIRK,BILL	DOE	LANL	NUCLEAR SYSTEMS,TESTING
BALDWIN,DARRELL	NASA	LERC	FACILITIES, EXEC. SEC.
BRANTLEY,WHIT	NASA	MSFC	PRELIM. DESIGN
BYERS,DAVID	NASA	LERC	NEP TECHNOLOGY
MARRIOTT,AL	NASA	JPL	SP-100
MILLER,TOM	NASA	LERC	NP PROJECT MANAGER
RICHMOND,BOB	NASA	MSFC	OAET R&T OFFICE
ROBBINS,RED	ANALYTICAL ENG		FORMER NERVA

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Figure 11

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Safety (Integrated with all Panels – Separate Report)			
SAWYER,BUZZ	NASA	HQ/QS	CREW SAFETY,PANEL CHAIR
ECKART,TED	CONS	AF	LAUNCH SAFETY,VANDENBERG
LEE,JIM	DOD	SDIO	SAFETY-MMW
ALLEN,GEORGE	DOE	SNL	NUCLEAR SAFETY
BOHL,DICK	DOE	LANL	SAFETY
BUDEN,DAVE	DOE	INEL	SAFETY, NERVA
DANDINI,VINCE	DOE	SNL	NUCLEAR SAFETY
GALLUP,DON	DOE	SNL	REACTOR CONCEPTS,SAFETY
KATO,WALTER	DOE	BNL	SAFETY
MARSHALL,AL	DOE	NP-50	SAFETY
NIEDERAUER,GEORGE	DOE	LANL	SAFETY SP-100
RICE,JOHN	DOE	INEL	MMW-SAFETY
STANLEY, MARLAND	DOE	INEL	SAFETY,EXEC.SEC.
ERCEGOVIC,DAVE	NASA	LERC	SAFETY ASSURANCE
REMP,KERRY	NASA	LERC	SAFETY
ROHAL,BOB	NASA	LERC	MISSION SAFETY

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Figure 12

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EXPECTED OUTPUT FROM WORKSHOPS:

- FOR EACH CONCEPT:
  - CRITICAL TEST REQUIREMENTS
  - SAFETY ISSUES IDENTIFIED
  - FACILITY REQUIREMENTS IDENTIFIED
- FIRST – ORDER COMPARISON:
  - MISSION BENEFIT
  - TECHNICAL RISK
  - DEVELOPMENT COST TO TRL – 6

NUCLEAR PROPULSION PROJECT

Figure 13

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ASSESSMENT PROCEDURE:

PANEL:	*CRITERIA:*
MISSION ANALYSIS	MISSION BENEFIT
PROPULSION	TECHNICAL RISK
REACTOR	TECHNICAL RISK
ADVANCED DEVEL. PLANS	DEVELOPMENT COST
SAFETY	SAFETY

OUTPUT:

→ WRITTEN NARRATIVE, STRENGTHS/WEAKNESSES

→ RELATIVE RANKING – COMPARISON TO BASELINE

NUCLEAR PROPULSION PROJECT

NPASSESS / 7-6-90 / JSC

Figure 14

NASA/DOE/DOD NUCLEAR THERMAL PROPULSION WORKSHOP

EVALUATION WORKSHEET

SESSION (PANEL NAME) ____________

EVALUATOR ____________

EVALUATOR PHONE NUMBER ____________ DATE ____________

CONCEPT ____________

PRESENTER (CFP) ____________

GENERAL COMMENTS ____________

CONCEPT STRENGTHS ____________

CONCEPT WEAKNESSES ____________

DESIRED CLARIFICATION OR SUBSTANTIATION ____________

Figure 15

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MISSION ANALYSIS PANEL

CONCEPT EVALUATION CRITERIA: MISSION BENEFIT

FACTORS:	*SCORES:*
1. IMLEO OR TRIP – TIME 2. SPECIFIC IMPULSE 2. INHERENT MISSION SAFETY/OPERATIONS – LAUNCH – ASSEMBLY – REUSEABILITY – DISPOSAL – etc. 4. COMMONALITY (OTHER MISSIONS) 5. AVAILABILITY IN 2006 ? 6. INHERENT RELIABILITY (COMPLEXITY)	1 PERFORMANCE MUCH LESS THAN BASELINE SYSTEM 2 PERFORMANCE LESS THAN BASELINE 3 SAME PERFORMANCE AS BASELINE 4 EXCEEDS THE PERFORMANCE OF THE BASELINE SYSTEM 5 PERFORMANCE SIGNIFICANTLY EXCEEDS BASELINE SYSTEM *CONFIDENCE IN SCORE:* *LOW MEDIUM HIGH*

NUCLEAR PROPULSION PROJECT

NEPEVALM7-8-90JSC

Figure 16

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PROPULSION TECHNOLOGY PANEL

CONCEPT EVALUATION CRITERIA: TECHNICAL RISK

TECHNOLOGY READINESS LEVEL:
CFP SELF – RATING: ________ (1 – 6)
EVALUATOR'S RATING: _______ (1 – 6)

FACTORS:

1. CONCEPT MATURITY (TRL)
   - FEASIBILITY DEMONSTRATED
   - CONCEPTS DEMONSTRATED
   - SCALING DEMOS/ RULES
   - SYSTEM DEMONSTRATIONS
2. KEY FEASIBILITY ISSUES/ TESTS REQ'D
3. FACILITY REQUIREMENTS
   - EXISTING / MODS REQ'D
   - NEW
4. INTEGRATION
   - PROPELLANT TANKS, LINES
   - TURBOPUMPS
   - NOZZLES
   - REFLECTORS, CONTROLS
   - REACTOR, SUPT. STRUCTURE
   - THERMAL MANAGEMENT
5. OTHERS

SCORES:

5 – MUCH LESS RISK THAN BASELINE

4 – LESS RISK THAN BASELINE

3 – SAME RISK AS BASELINE

2 – MORE TECHNICAL RISK THAN BASELINE SYSTEM

1 – MUCH MORE RISK THAN THE BASELINE SYSTEM

CONFIDENCE IN SCORE:

LOW *MEDIUM* *HIGH*

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NE' 'NLP/7-8-90USC

Figure 17

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TECHNOLOGY READINESS LEVEL:

LEVEL 1: BASIC PRINCIPLES OBSERVED AND REPORTED

LEVEL 2: CONCEPT FORMULATED INTO APPLICATION

LEVEL 3: PROOF – OF – CONCEPT PROVEN

LEVEL 4: COMPONENT/BREADBOARD VALIDATION IN LAB

LEVEL 5: COMPONENT/BREADBORD DEMO IN RELEVANT ENVIRONMENT

LEVEL 6: SYSTEM VALIDATION MODEL DEMONSTRATED IN SIMULATED ENVIRONMENT, INCLUDING LIFE, PERFORMANCE,AND SYSTEM INTERACTIONS

LEVEL 7: FLIGHT TEST OF A QUALIFIED SYSTEM

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TRLEVEL7-8-90USC

Figure 18

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REACTOR TECHNOLOGY PANEL

CONCEPT EVALUATION CRITERIA: TECHNICAL RISK

TECHNOLOGY READINESS LEVEL: CFP SELF – RATING: ________ (1 – 6)

EVALUATOR'S RATING: ________ (1 – 6)

FACTORS:

1. CONCEPT MATURITY (TRL)
   - FEASIBILITY DEMONSTRATED
   - NUCLEAR FURNACE
   - REACTOR TESTS/VERIF.
   - MODELS VERIFIED
2. KEY FEASIBILITY ISSUES/ TESTS REQ'D
3. FACILITY REQUIREMENTS
   - EXISTING / MODS REQ'D
   - NEW
4. PROPULSION SYSTEM INTEGRATION
   - FAILURE MODES
   - THERMAL MANAGEMENT
   - CONTROLS/INSTRUMENT.
5. VEHICLE OPERATIONS/ SAFETY
   - ORBITAL ASSEMBLY
   - LAUNCH/REENTRY/DISPOSAL
   - RESTART/COMMONALITY

SCORES:

5 – MUCH LESS RISK THAN BASELINE

4 – LESS RISK THAN BASELINE

3 – SAME RISK AS BASELINE

2 – MORE TECHNICAL RISK THAN BASELINE SYSTEM

1 – MUCH MORE RISK THAN THE BASELINE SYSTEM

CONFIDENCE IN SCORE:

LOW MEDIUM HIGH

NUCLEAR PROPULSION PROJECT

NEPEVALRF-8-90JSC

Figure 19

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ADVANCED DEVELOPMENT PLAN PANEL

CONCEPT EVALUATION CRITERIA: DEVELOPMENT COST

CFP ESTIMATED COST, ________ , $M

EVALUATOR'S ESTIMATED COST, ________ , $M

FACTORS:

1. CONCEPT MATURITY (TRL 1 – 6)
   - FEASIBILITY DEMONSTRATED
   - COMPONENT VALIDATION
   - SYSTEM BREADBOARD DEMO.
   - SYSTEM VALIDATED
2. KEY FEASIBILITY ISSUES/ TESTS REQ'D
3. VERIFICATION ISSUES (SAFETY/PERF.)
   - SIMULATION
   - TESTING
4. FACILITY REQUIREMENTS
   - EXISTING / MODS REQ'D
   - NEW
5. ESTIMATED DEVELOPMENT COST (COMPARED TO BASELINE SYSTEM)

SCORES:

5 – MUCH LESS COST THAN BASELINE

4 – LESS COST THAN BASELINE

3 – SAME COST AS BASELINE

2 – MORE COST THAN BASELINE SYSTEM

1 – MUCH MORE COST THAN THE BASELINE SYSTEM

CONFIDENCE IN SCORE:

LOW MEDIUM HIGH

NUCLEAR PROPULSION PROJECT

NEP 17-8-90JSC

Figure 20

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SAFETY PANEL

CONCEPT EVALUATION CRITERIA: SAFETY

FACTORS:

1. HAZARD IDENTIFICATION & MITIGATION
2. SAFETY VERIFICATION ISSUES
3. LAUNCH SAFETY COMPATIBILITY
4. INHERENT CONTROL/ STABILITY
5. SYSTEM REFURBISHMENT/ DISPOSAL
6. ORBITAL ASSEMBLY / STARTUP
7. CREW RADIATION PROTECTION
8. REDUNDANCY / RELIABILITY
9. ETC.

SCORES:

1 – UNACCEPTABLE
2 – NOT AS SAFE AS BASELINE SYSTEM
3 – ABOUT THE SAME AS BASELINE
4 – SAFER THAN BASELINE
5 – MUCH SAFER THAN BASELINE

CONFIDENCE IN SCORE:

LOW MEDIUM HIGH

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NEP 7-8-80/JSC

Figure 21

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AFTER THE WORKSHOPS:

TECHNOLOGY REVIEW PANEL / (SUB – GROUPS)

TECHNICAL INPUT:

CLARIFY ISSUES
VERIFY CLAIMS
COLLATE EVALUATIONS
SAMPLE CALCS.
QUESTIONS TO CFP
RECOMMENDATIONS

NASA

DOE

DOD

STEERING COMMITTEE
WORKSHOP PROCEEDINGS
FEEDBACK TO CFP'S

NUCLEAR PROPULSION PROJECT

NPAFTER / 7-8-80 / JSC

Figure 22

N92-11091

NUCLEAR THERMAL ROCKET WORKSHOP REFERENCE SYSTEM -ROVER/NERVA-

Dr. Stanley K. Borowski
NASA Lewis Research Center

INTRODUCTION

The Rover/NERVA engine system is to be used as a "reference," against which each of the other concepts to be presented in this workshop will be compared. In this presentation I'll review the operational characteristics of the nuclear thermal rocket (NTR), the accomplishments of the Rover/NERVA programs, and performance characteristics of the NERVA-type systems for both Mars and lunar mission applications. I'll also briefly touch on the issues of ground testing, NTR safety, NASA's nuclear propulsion project plans, and NTR development cost estimates before concluding my presentation.

NERVA REFERENCE ENGINE

The NTR is basically a monopropellant liquid rocket system which utilizes a nuclear reactor core for power generation and propellant heating (Figure 1). High pressure hydrogen from a turbopump assembly passes through a high power reactor core where it is heated to high temperatures and then exhausted through a convergent-divergent nozzle at high speeds to produce thrust. Before entering the reactor core, hydrogen flowing from the pumps is first "preheated" by cooling the nozzle, reflector, control rods, peripheral shield, and core support structure.

In the "hot bleed cycle" (see Figure 2), this preheated hydrogen is routed down though the reactor core for heating to design temperatures and subsequent nozzle expansion. Approximately 3% of the heated hydrogen is diverted from the nozzle plenum chamber, cooled, and then used to drive the turbopumps with the exhaust being utilized either for roll control or readmitted into the diverging portion of the nozzle for additional thrust generation. In the "full flow topping" or "expander cycle" engine, the preheated hydrogen is routed to the turbopumps and then through the reactor core with the entire propellant flow being heated to design temperatures (Figure 2) providing more optimum performance in terms of higher engine specific impulse (I_{sp}).

The accomplishments of the Rover/NERVA program are summarized in Figures 3, 4, and 5. As Figure 3 indicates, the achievements were quite impressive with a total of 20 rocket reactors designed, built, and tested between 1955 and 1973 at a cost of $1.4 Billion. From program start in 1955 to testing of the first KIWI-A reactor was only 4 years which is pretty impressive in itself. Major performance accomplishments were demonstrated in the areas of power and thrust levels, peak and fuel exit temperatures and equivalent specific impulse, and full power burn duration. Most notable was the NERVA program's NRX-A6 test in which the system operated for 62 minutes at a thrust level of about 55,000 pounds-force (55klbf) and a thermal power level of about 1125 megawatts (MWt).

The NERVA program's NRX series of reactors culminated in the downward test firing of the Experimental Engine Prototype (the XE-P) in 1969. The NRX-XE underwent 28 startup/shutdown cycles and demonstrated rather convincingly the practicality of the NTR systems. In addition to these "full scale" integrated engine tests, electric and nuclear furnace (NF-1) tests were also conducted in an effort to develop higher temperature/longer life reactor fuels. Anticipated performance for the "composite" and "carbide" fuel forms, which you will be hearing about at this workshop, is about 10 hours at I_{sp} values of about 925 seconds and 1020 seconds for the composite and carbide fuel forms, respectively.

Again, 20 reactors were tested in the Rover/NERVA programs and the chronology of system tests for both programs is shown in Figure 4. After demonstrating feasibility of the basic KIWI-B series concept, the Los Alamos Rover program concentrated its efforts on fuel research and higher power density systems. The Phoebus-1B system, tested in 1967, was approximately the same physical size as KIWI-B (see Figure 5) but was operated at 1500 MWt. Phoebus-2A (shown in Figures 5 and 6), was designed for 5000 MWt and 250 klbf. It was operated at about 80% of its rated design conditions for about 12.5 minutes in July 1968 and was the most powerful nuclear rocket reactor ever built. It was to be the prototype for the 200-250 klbf-class NERVA II engine contemplated by NASA at that time. Figure 6 is a picture of Phoebus-2A being transported to "Test Cell C" (Figure 7) on the Jackass & Western Railroad for full power testing.

A final noteworthy reactor system was the Nuclear Furnace (NF-1). It was operated in 1972 at about 44 MWt and was utilized primarily as a inexpensive "test bed" system for screening advanced fuels and reactor structural materials. A special feature of the NF-1 reactor was its "effluent cleanup system" which effectively removed radioactive contaminants from effluent reactor gas. The database provided by the Nuclear Furnace is of particular interest today because of environmental restrictions which would prevent open-air testing.

Figures 8, 9, and 10 show three of the six NRX-series reactor systems developed by Aerojet and Westinghouse for NASA and the AEC during the Nuclear Engine for Rocket Vehicle Application (NERVA) program. Figure 8 shows the NRX-A3 being prepared for test firing at Test Cell C at the Nuclear Rocket Development Station (NRDS) at Jackass Flats, Nevada. Figure 9 shows the 62 minute "continuous full-power burn" of the NRX-A6 system in December 1967 with its two large 500,000 gallon liquid hydrogen tanks off to the right. Last, Figure 10 shows the XE prototype engine installed for downward test firing at the ETS-1 test facility also at the NRDS.

The very large database accumulated in both the Rover/NERVA programs was integrated into a reference NERVA engine design in 1972. A mockup of the 1972 NERVA is shown in Figure 11. The fuel form was coated UC_2 particles in a graphite matrix, the chamber pressure was 450 psia, and hydrogen exhaust temperatures from the reactor ranged from 2,350 to 2,500 K. Both hot bleed and expander cycle versions of the 1972 NERVA were examined with I_{sp} values ranging from 825 to 870 seconds. The engine shown in Figure 11 had an overall length of about 10.5 meters with a 100-to-1 nozzle expansion ratio; it weighed a little over 11 metric tons, resulting in an engine thrust-to-weight ratio of 3. In terms of NASA's technology maturity ranking, the XE engine was rated at an overall system technology readiness level of about 6 (TRL=6 is the prelude to the next development step, which is the "flight engine"). Some of the NRX components were rated at about the TRL=5 level and requirea some further development (see Figure 12).

On the "non-nuclear" subsystem side, there have been major advances in chemical rocket technology in the 17 years since termination of the NERVA program. Of particular note are the significant performance improvements and accompanying weight reductions in the turbopump and nozzle areas. Figure 13 compares the Space Shuttle Main Engine (SSME) and the 1972 NERVA. You can see that the SSME nozzle is lighter and is capable of handling exhaust gas temperatures in excess of 3,100 K (equivalent to those anticipated from the advanced carbide fuels). It also operates with heat fluxes four times greater than those encountered in the NERVA program. Pump discharge pressures from the SSME hydrogen turbopump are also a factor of 5 greater than those of the 1972 NERVA. Chemical propulsion system development has therefore provided us with a significant database for use in the design of current day NERVA-type engine systems. Performance projections for "state-of-the-art" NERVA derivative reactor systems are shown in Table 1. Assuming a full-flow expander cycle engine operating at about 1000 psia, the I_{sp} values for a 500-to-1 nozzle expansion ratio vary from about 850 to 885 seconds for graphite fuel, about 925 seconds for the composite fuel, and about 1020 seconds for the pure carbide fuel form. Higher performance/lower weight non-nuclear components also result in a 2 to 3 metric ton savings in overall engine mass and the improved engine thrust-to-weight ratios shown.

REFERENCE MARS MISSION ANALYSIS

I would now like to review with you the results of trajectory and mission analysis work performed at the Lewis Research Center for the reference Mars mission. Both 1972 vintage and "state-of-the-art" NERVA-type systems were examined. But first I'd like to briefly show you some previous NASA work in this area from the 1960-1970 time frame to set the stage for the current results I will be showing you shortly. I'll also point out the many similarities that exist between these earlier studies and our current day results. In August of 1969, just one month after the Apollo 11 moon landing, Werner von Braun described NASA's proposal for a piloted mission to Mars (around 1981) at a hearing of the Senate Committee on Aeronautics and Space Science. The mission would be accomplished using two spacecraft, each carrying a 6-person crew and having an initial mass in low Earth orbit (IMLEO) of about 727 tons. Each spacecraft would carry three 445 kilonewton (about 100klbf) NERVA-class engines (with an I_{sp} of 850 seconds) of which two would be used only for departing Earth orbit for the 270-day journey out to Mars. After this trans-Mars insertion (TMI) burn, the two strap-on NERVA-powered booster stages would separate, retrofire, and return to Earth for liquid hydrogen refueling and reuse (see Figure 14). Subsequent mission maneuvers would be accomplished by the remaining NERVA engine on the core spacecraft. Later mission studies assumed a single 75klbf-class NERVA engine for spacecraft propulsion (see Figure 15), and a multiple perigee burn Earth departure scenario was adopted. Two large tanks attached to the core spacecraft would carry the TMI propellant and would be jettisoned after completion of the TMI maneuver. The remaining propellant would be accommodated in the central core tank(s).

The mission profile proposed by von Braun was a 640-day opposition class mission with an 80-day stay at Mars and inbound Venus swingby. Twenty-one years later, NASA's reference Mars mission scenario is a 2016 opposition class mission with 30-day surface stay and an inbound Venus swingby (see Figure 16). For this particular opportunity, the overall mission duration is attractive--on the order of 434 days. Most opposition class missions have mission durations somewhere in the 420- to 650-day ballpark.

The 2016 reference NTR mission profile originally assumed for the workshop is shown in Figure 17. The "all propulsive" NTR vehicle features expendable TMI and Mars orbital capture (MOC) tanks attached to an optional central truss structure. Trans-Earth injection and Earth orbital capture (EOC) propellant would be contained in a common core propellant tank in the vehicle "reuse" mode. In the "expendable" vehicle mode, the return of the crew to Earth could be accomplished utilizing an Earth Crew Capture Vehicle (ECCV).

The mission assumption and ground rules are shown in Table 2 and the propulsion system, boil off, and tankage assumptions are summarized in Table 3. Because our principle "figure-of-merit" for this analysis is IMLEO, a single 75klbf NERVA-class engine has been assumed as the baseline engine thrust level, along with perigee propulsion. By utilizing a multi-perigee burn departure scenario, we can more effectively impart propulsive energy to our spacecraft while reducing gravity losses associated with the finite burn durations accompanying lower thrust-to-weight ratio vehicle designs.

The motivation for going to multiple perigee burns with lower thrust engine systems is illustrated quite dramatically in Figure 18. If we tried a "one burn" Earth departure maneuver using a single 75klbf engine with a vehicle thrust-to-weight ratio of about 0.05, gravity losses ("g-losses") would add 1500 meters per second (m/s) to the ideal TMI Delta-V requirement. By going to the "3 perigee burn" approach, g-losses are reduced to about 350 m/s. The actual g-loss value will vary, of course, depending on the mission C_3 requirement, the I_{sp} of the NTR, the orbital departure altitude, and the vehicle thrust-to-weight ratio. By using a single higher thrust engine or by clustering several lower thrust engines, the vehicle thrust-to-weight ratio can be increased, and single burn departure scenarios are possible with acceptable g-loss. As will be shown later in this talk, a single 250klbf Phoebus-2A class NTR can perform the 2016 Mars mission opportunity for an IMLEO of about 750 tons using a single burn Earth departure. With a thrust-to-weight ratio of about 0.15, the g-losses incurred during TMI are on the order of 400 m/s.

The "reference trajectory" assumed for this workshop (and shown in Figure 16) was originally established during the "90-Day Study" for the aerobrake chemical vehicle that was baselined at that time. The trajectory was subsequently adjusted somewhat for the NTR analysis purposes, although it was by no means optimum. An aerobrake-optimized trajectory weights both the arrival velocities at Mars and Earth more heavily since it assumes that a lightweight, high, heat-flux-resistant aerobrake will be developed in the future. By weighting the MOC and EOC velocities more heavily, the TMI and TEI Delta-V requirements can be reduced, thereby compensating for the limited capability of the chemical propulsion system. Table 4 summarizes trajectory data and associated IMLEO estimates for both the "doctored-up" NTR reference trajectory and a new "all propulsive optimized" NTR trajectory recently developed by Lewis Research Center's Advanced Space Analysis Office. The NTR optimized trajectory weights the departure maneuvers from Earth and Mars more heavily than the capture maneuvers thereby exploiting more fully the high I_{sp} capability of the NTR system.

Estimates of IMLEO from Marshall Space Flight Center's contractor, Boeing, and from the Lewis Research Center (LeRC) are shown for the reference trajectory and a "state-of-the-art" composite fuel NERVA derivative system operating at an I_{sp} of about 925 seconds. The Boeing estimate for IMLEO is about 735 tons and is based on the assumption of a fixed 200 m/s g-loss value and use of advanced composite cryogenic tanks. The LeRC IMLEO estimate is somewhat higher because of a more accurate g-loss estimate and different tankage assumptions. What is most impressive, however, is the impact on IMLEO of using the "all propulsive optimized" trajectory that results in a 150-ton mass savings!

A comparison of vehicle size for the 2016 Mars mission using the optimized and non-optimized trajectories of Table 4 are shown in Figure 19. The two TMI drop tanks are limited in size to the payload shroud dimensions of anticipated heavy lift launch vehicles currently under study and are approximately 10 meters in diameter by about 30 meters in length.

The performance potential of different 75klbf-class NERVA engines of the type shown in Table 1 were examined and compared in terms of IMLEO and total engine burn time requirements for the "all propulsive optimized" 2016 Mars trajectory described in Table 4. The results for "state-of-the-art" NERVA derivative reactor (NDR) systems using an expander engine cycle and a variety of fuel forms (graphite, composite, and carbide) are shown in Figure 20. At a 1000 psia chamber pressure and a 500-to-1 nozzle expansion ratio, a "current day" graphite NERVA system operating at 2,350 K (a temperature routinely demonstrated in the NERVA program) would deliver an I_{sp} of 850 seconds. The associated IMLEO and engine burn time for this system is 725 tons and 3.38 hours, respectively. Going to the higher performance composite and carbide fuel forms, the IMLEO and burn time requirements decrease to 613 tons/2.99 hours and 518 tons/2.64 hours, respectively. These values are to be compared to the reference aerobrake chemical vehicle from NASA's "90-Day Study" which had an IMLEO of about 752 tons for the expendable ECCV Earth return option, and about 830 tons for the reusable propulsive return option. The aerobrake mass fraction assumed for the MOC aerobrake was about 13 percent, which is also somewhat optimistic.

A "state-of-the-art," graphite fuel NDR engine propulsively returning the basic core spacecraft to LEO can therefore outperform the best aerobraked chemical vehicle design currently on the "drawing boards" by 27 tons when the chemical/aerobrake vehicle is operated in the expendable ECCV recovery mode, and by 105 tons in the vehicle reuse mode. Even the 1972 graphite fuel NERVA design outperforms the aerobraked chemical vehicle in the reuse mode with an IMLEO and engine burn time of about 755 tons and about 3.75 hours, respectively.

The relative vehicle size comparison for the graphite, composite, and carbide fuel NDR systems is shown in Figure 21. The individual burn duration for both 75klbf and 250klbf-class NTR systems are summarized in Table 5, and the relative vehicle sizes for the "3 perigee burn" 75klbf and "one burn" 250klbf-class NTR systems are shown in Figure 22. The 75klbf and 250klbf engines both assume a 1000 psia chamber pressure and a 500-to-1 nozzle expansion ratio, and utilize a composite fuel capable for delivering 925 seconds of I_{sp}.

In contrast to the approximately 3-hour total engine burn duration for the composite fuel 75klbf NDR system, the 250klbf engine burn time totals a little over one hour at 65.3 minutes. The IMLEO requirement of 749 tons is comparable to that of the expendable aerobrake chemical vehicle due to the higher g-loss accompanying the "one burn" departure scenario and the heavier weight (about 21.8 tons) of this higher thrust engine. Perigee propulsion can reduce the IMLEO requirements further, at the expense of the more complex "3 burn" departure scenario.

Other Mars mission opportunities have been examined besides the 2016 opportunity in order to assess the magnitude of IMLEO variation across a synodic period. Figure 23 shows the sensitivity of IMLEO to mission roundtrip time (for a 925-second NTR system with multiple perigee burns) for a variety of mission modes and two different opportunities--an easy one (2018) and a tough one (in 2014). The mission modes examined include a reusable, all propulsive mode, one with an ECCV for Earth return, and a split mission in which cargo is carried on a "minimum energy" conjunction-class trajectory while the piloted portion of the mission travels a faster, higher energy opposition-class trajectory. Stay times at Mars are in all cases assumed to be 30 days. This split-type mission is often referred to as the "split-sprint." A more advanced (but potentially greater risk) variation of the split mission involves having the cargo vehicle also carry the "return propellant" for the piloted vehicle. This variation was referred to during the 1960's as the "Hohmann tanker/dual vehicle" mission mode.

As we push from 434 days to round trip times on the order of one year, the IMLEO for the all-propulsive single vehicle case in 2018 almost doubles increasing from about 700 tons to about 1350 tons. By utilizing an ECCV for Earth return, one can shave off about 300 tons from the IMLEO requirement for the one-year mission. In the split-sprint mission mode the piloted vehicle IMLEO is on the order of 375 tons for the one-year mission although the total IMLEO requirement including the cargo vehicle is on the order of 750 tons. Even in the most difficult mission year of 2024, trip times from 400 to 500 days are possible with the various mission modes available. This is an important operational advantage of the NTR system over NEP systems--the ability to shorten trip times across the entire spectrum of Mars mission opportunities using a technology with a proven experimental database.

LUNAR MISSION ANALYSIS

Lewis Research Center has also been conducting "in-house" and contracted study efforts aimed at assessing the benefits of using NTR technology for lunar mission applications. During the "90-Day Study" the establishment of a lunar outpost was considered a prelude to undertaking missions to Mars. The flight schedule for the proposed lunar outpost scenario covered a 15-year period and required 30 separate flights involving either cargo, piloted, or combination missions (see Figure 24). The base line piloted Lunar Transportation Vehicle (LTV) in the 90-Day Study utilized chemical propulsion and required an aerobrake for Earth return to keep the IMLEO within a reasonable range (see Figure 25). The IMLEO for the first piloted lunar missions, which was used to size the system, was about 194 tons.

In the next several vugraphs you'll see some of the findings resulting from our contracted effort with SAIC. The specific mission and NTR system definition assumptions used in the SAIC study are shown in Figure 26 and 27, respectively, and a comparison of the IMLEO requirements for the first piloted mission using aerobraked chemical and NTR technologies is summarized in Figure 28. Figure 28 shows a mass savings of about 32 tons using an NTR-powered LTV in a "4 burn" all-propulsive lunar mission profile. By "4 burn" we refer to the four major propulsive maneuvers of trans-lunar injection, lunar orbit capture, trans-Earth injection, and Earth orbit capture.

In the SAIC study, the mass penalty associated with disposing of "end-of-life" NTR systems was also assessed and included in the IMLEO comparisons. A number of disposal modes were examined using 1-, 2-, 3-, and 4-burn lunar NTR scenarios, and the results are shown in Figure 28. One can see that disposing of the spent NTR propulsion module (consisting of a small propellant capacity run tank, an avionics package, and the NTR) into a 1,000 kilometer parking orbit (following Earth orbit capture of the NTR vehicle back into LEO) results in a modest 2-ton penalty. The mass penalty increases for the more demanding disposal modes into heliocentric and super-geo orbits. The overall impact on IMLEO is modest, however, compared to the chemical/aerobrake baseline system.

The overall mass savings resulting from using NTR technology in the lunar outpost scenario is summarized in Figure 29. Over a 15-year flight schedule, the total computed mass delivered to LEO for the reference aerobraked chemical LTV system was in excess of 5,000 metric tons. Using a conservative NTR growth assumption (I_{sp} of 900 seconds and nozzle expansion ratio of 200-to-1), a "4 burn", all-propulsive NTR LTV system would reduce the delivered mass to LEO to about 4040 tons--a savings of approximately 20 percent.

Since it's probably going to be tough to have the NTR system ready for the proposed first piloted mission in the early 2000's, without a major commitment of resources, the SAIC study also looked at "phasing in" the NTR system into the reference 90-Day Study scenario. This approach would still provide an IMLEO savings and would also provide valuable operational experience in the use of NTR systems in a "nearby" space environment prior to undertaking the more demanding Mars mission. Even with the phased NTR approach, a 15 percent IMLEO savings is indicated with disposal penalties again taken into consideration.

TESTING

In my last few vugraphs I would like to touch briefly on a number of peripheral issues that are very important. The first deals with the ground testing of full scale integrated reactor and flight engine systems. It is obvious that we cannot operate as we did in the past at NRDS with "open air" testing. The Nuclear Propulsion Project will therefore have to address a number of programmatic and development issues associated with NTR ground testing (see Figures 30 and 31). Concepts for "fully contained" test facilities have been proposed based on the earlier Nuclear Furnace experience. A schematic for one such facility, proposed by the Idaho National Engineering Laboratory, is shown in Figure 32. The facility would contain a number of debris traps, water sprays, cooler/scrubbers, filters and charcoal beds for removing particulates, soluble fission products, and noble gases from the engine exhaust prior to the hydrogen being released to the burn stack.

Another option for confining engine exhaust gases might be to use some of the weapons test tunnels at the Nevada Test Site. Tunnel testing could have a number of advantages (Figure 33), and its usefulness for NTR testing will have to be assessed more fully in the future. A number of NASA, DOE and industry people visited the Nevada Test Site about a month ago and toured a weapons test tunnel and portions of the NRDS at Jackass Flats. There are a lot of site assets that still exist at the NRDS (see Figures 34 and 35) that could be put to good use in a future NTR development program.

With regard to NTR safety, the Rover/NERVA programs had an exemplary safety record handling large quantities of liquid hydrogen (on the order of a million gallons or more during some engine tests) and large radioactive systems remotely in its E-MAD facility during the post irradiation disassembly and fuel examination periods. The 1972 NERVA reference engine was also designed to be a "man-rated" system and included redundant turbopumps and valve sets (see Figure 36). Probablistic design and failure mode effects analyses were also done. The NERVA system that resulted from this analysis approach (see Figure 37) had good component redundancy to eliminate a number of identified failure modes that could develop during various phases of a typical lunar mission that was selected by NASA for its Design Reference Mission. A good database and starting point for a "man-rated" NTR system can therefore be found in the NERVA program.

Another issue that has surfaced recently deals with the diffusion of fission product gases from the NTR system during powered operation and the overall dose rates experienced by the crew of an NTR-powered spacecraft during a typical Mars mission. Although work is just being restarted in this area, Figure 38 provides us with some rough numbers. Shown is the temporal variation of dose rate for the "non-optimized" 2016 Mars reference mission that was originally assumed for this workshop. The burn duration for the major maneuvers and the approximate elapsed time between burns is shown at the top of the figure; the variation of dose rate experienced by a crew member standing 100 feet away from the unshielded reactor core center-line (a rather pessimistic assumption) is shown at the bottom. It is quite evident that during the full power TMI burn, the dose rate is lethal. One day after TMI, however, the dose rate has dropped by a factor of 6500, and after the 156-day coast period to Mars it is down to 0.23 Rem/hour. Following the MOC burn, the crew would depart the Mars spacecraft staying within the protected cone area provided by the NTR engine's external disk shield. After a 30-day surface stay, the returning Mars excursion vehicle could fly past the unshielded NTR and receive less than 2 Rem/hour at the 100-foot separation distance. Following the TEI burn-and-coast phase, the dose rate at our reference location is on the order of 75 millirem per hour prior to EOC. Up in the front of the vehicle where the crew will actually be located, the benefits of the external disk shield, core propellant tank, truss structure distance, and solar flare storm shelter will reduce overall accumulated crew dose to the required 5 Rem per year.

Because the NTR system is a high-thrust system, it provides all of its impulse to the spacecraft quickly, unlike the NEP systems that must operate for a major portion of the total mission time--on the order of 10,000 to 20,000 hours. As a result of the NTR system's short burn duration, the radioactive inventory has a significant period of time to decay, thereby reducing the system's overall radiological hazard.

PROJECT PLANNING

We are working and reworking the Project Plan, taking into account inputs from industry sources, NASA sources, and DOE inputs. Our earlier speaker, Gary Bennett, outlined a three-phase program in which the important project elements are system development, nonnuclear component development and nuclear component development.

Obviously, a number of critical tests have to be done right up front. Facilities requirements must be defined in the first couple of years. We need to identify not only the components to be tested on the ground, but also the big ticket items, such as the ground test facility for doing the integrated and full scale engine tests.

Also we will include innovative technology (aimed at 2nd and 3rd generation systems) throughout a good part of the first two phases; we will also be conducting mission studies for a good portion of the early phases, identifying system concepts, and going through preliminary, critical and final design reviews. Potentially there will be a design freeze in which we could be really focusing in on the component and subsystem tests that will be tested in the latter years. Then ultimately, we get into reactor tests.

The NERVA program cost $1.4 billion; escalating that to today's dollars would be almost $10 billion. However, it is important to remember that the NERVA program was a gold-plated program; whole integrated reactors were put together just to test improvements in coating. We think there are better ways to do that with smaller subscale electric furnace, and nuclear furnace tests. Plus, there is now an established database, so while we have to reverify it, I don't know that it's necessary for us to go through the same number of tests. Obviously we must develop a Project Plan in the course of the next couple of months and over the course of the first few years. Also, a number of critical nonnuclear and nuclear component tests have to be done.

DEVELOPMENT COST

My first estimate on the cost of this program is close to $3 billion to take it to technology level readiness 6. Somebody might get up and say they think it's more like 5 billion and I wouldn't argue very strongly. I think the results of this workshop will pull in a lot more information for us to make a more informed judgment on what the program will realistically cost.

Again, I think a critical thing in the program is the facility cost for the full scale engine test. We are certainly going to need a study by an unbiased major contractor who has experience in doing the large scale nuclear facilities.

CONCLUSION

My last vugraph (Figure 39) summarizes my conclusions and observations. The Rover/NERVA programs definitely established an impressive database that demonstrated convincingly the feasibility of the graphite core NTR concept. This database was used in putting together the 1972 NERVA reference engine design. Based on our analysis a "state-of-the-art," graphite core NDR system would have and IMLEO of 725 tons which is 105 tons lighter than the best aerobrake chemical system that NASA can envision today. Even 1972 NERVA can outperform it.

The ground test experience gained during the Rover/NERVA programs was substantial even though most of it was done in the open air. The Nuclear Furnace experiment with its effluent control system provides us with an important database for designing a "contained" test facility meeting today's environmental standards.

With the continued advances in chemical propulsion technology over the last 17 years, higher performance/lighter weight turbopumps, nozzles, and valves should help to improve the engine thrust-to-weight ratio for today's NERVA derivative engine. One should not overlook the impact of a radiation environment on component performance that could present some unforseen problems in a future development effort.

The NTR is an *enabling* technology for future piloted missions to Mars. It can shorten roundtrip mission times substantially allowing one-year missions to be contemplated. We also think that the NTR is *enhancing* for lunar mission applications, providing not only IMLEO savings but valuable operational experience with this impressive new propulsion technology.

A Nuclear Propulsion Program will certainly require a lot of work and a significant infusion of resources to become a reality. For the NTR I think test facilities are the key item with high-temperature fuel development being very important also.

Lastly, I'd like to point out that the projected performance parameters for NTR that we have been using in our analyses thus far are within a factor of 2 or less of those already demonstrated in the Rover/NERVA programs. This provides real confidence that piloted missions to the Moon and Mars will someday be a reality with the NTR system!

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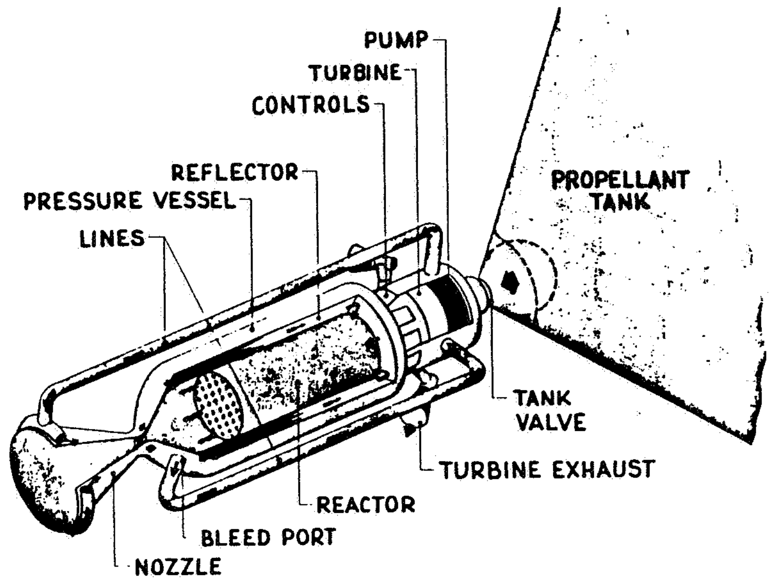
NUCLEAR ROCKET



Nuclear Thermal Rocket - A space propulsion concept in which the heat from a nuclear fission reactor is used to raise the temperature of the propellant, which is then expanded through a nozzle to provide thrust.

Figure 1

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FULL FLOW AND HOT BLEED ENGINES

NUCLEAR-ROCKET ENGINE CYCLES

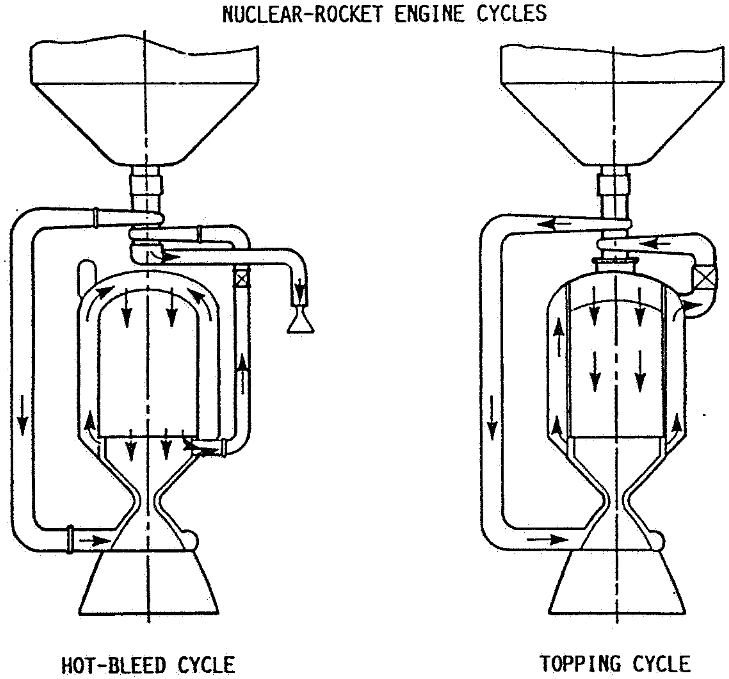

Figure 2

NASA LEWIS RESEARCH CENTER

ROVER/NERVA PROGRAM SUMMARY

- 20 REACTORS DESIGNED, BUILT, AND TESTED BETWEEN 1955 AND 1973 AT A COST OF APPROXIMATELY $1.4 BILLION. (FIRST REACTOR TEST: KIWI-A, JULY 1959)
- DEMONSTRATED PERFORMANCE

POWER (MWt)	~1100 (NRX SERIES) - 4100 (PHOEBUS -2A)
THRUST (klbf)	~55 (NRX SERIES) - 210 (PHOEBUS -2A)
PEAK/EXIT FUEL TEMPS. (K)	~2750/2550 (PEWEE)
EQUIV. SPECIFIC IMPULSE(S)	~850 (PEWEE)
BURN ENDURANCE	1-2 HOURS
- NRX-A6	62 MINUTES AT 1125 MWt (SINGLE BURN)
- NUCLEAR FURNACE	109 MINUTES ACCUMULATED (4 TESTS) AT 44 MWt
START/STOP	28 AUTO START-UPS/SHUTDOWNS WITH XE

- BROAD AND DEEP DATABASE ACHIEVED/USED IN PRELIMINARY NERVA "FLIGHT ENGINE" DESIGN (1972)
- ANTICIPATED PERFORMANCE

BURN ENDURANCE	~10 HOURS (DEMONSTRATED IN ELECTRIC FURNACE TESTS AT WESTINGHOUSE)
SPECIFIC IMPULSE	UP TO 925s (COMPOSITE)/UP TO 1020s (CARBIDE FUELS)

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Figure 3

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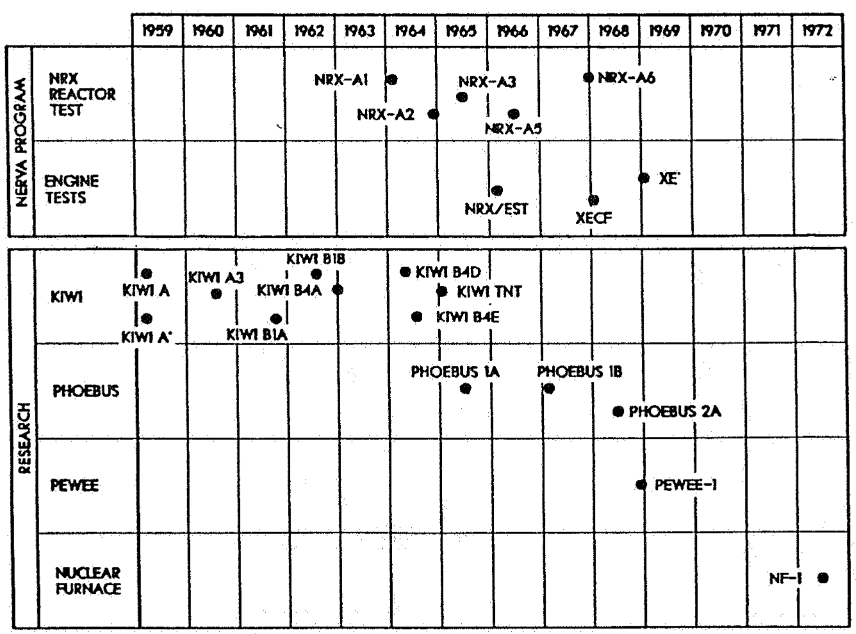
CHRONOLOGY OF MAJOR NUCLEAR ROCKET REACTOR TESTS

ASAO

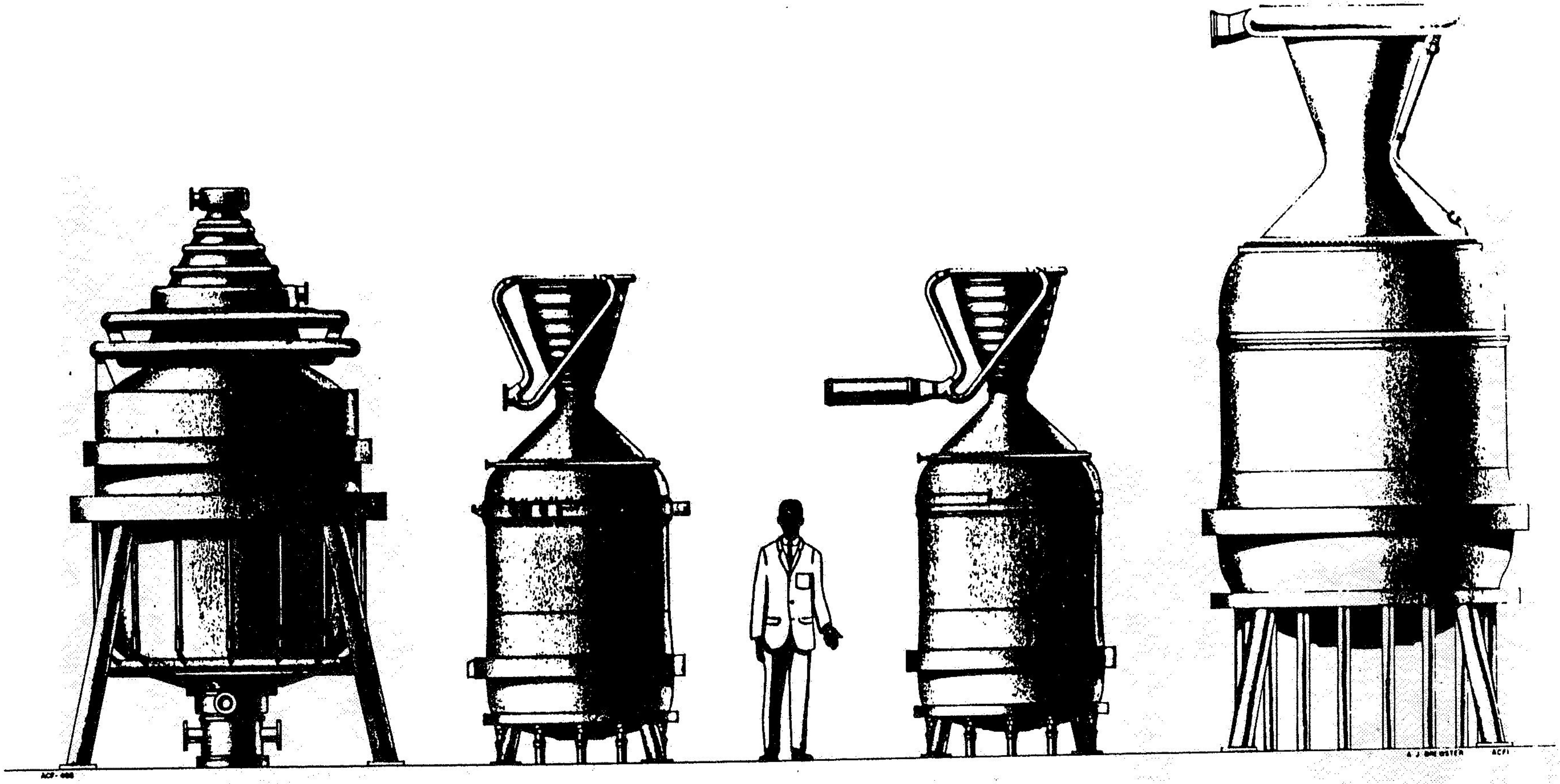


REF: LANL

ADVANCED SPAC 66 NALYSIS OFFICE

Figure 4

REACTORS TESTED IN ROVER PROGRAM (LANL)



KIWI A
1958-60
100 MEGAWATTS
0 LBS. THRUST

KIWI B
1961-64
1000 MEGAWATTS
50,000 LBS. THRUST

PHOEBUS 1
1965-66
1000 & 1500 MEGAWATTS
50,000 LBS. THRUST

PHOEBUS 2
1967
5000 MEGAWATTS
250,000 LBS. THRUST

NRX SERIES BEGINS (6 SYSTEM TESTS) WITH NERVA PROGRAM

Figure 5

Phoebus 2A in Transit to Test Cell C

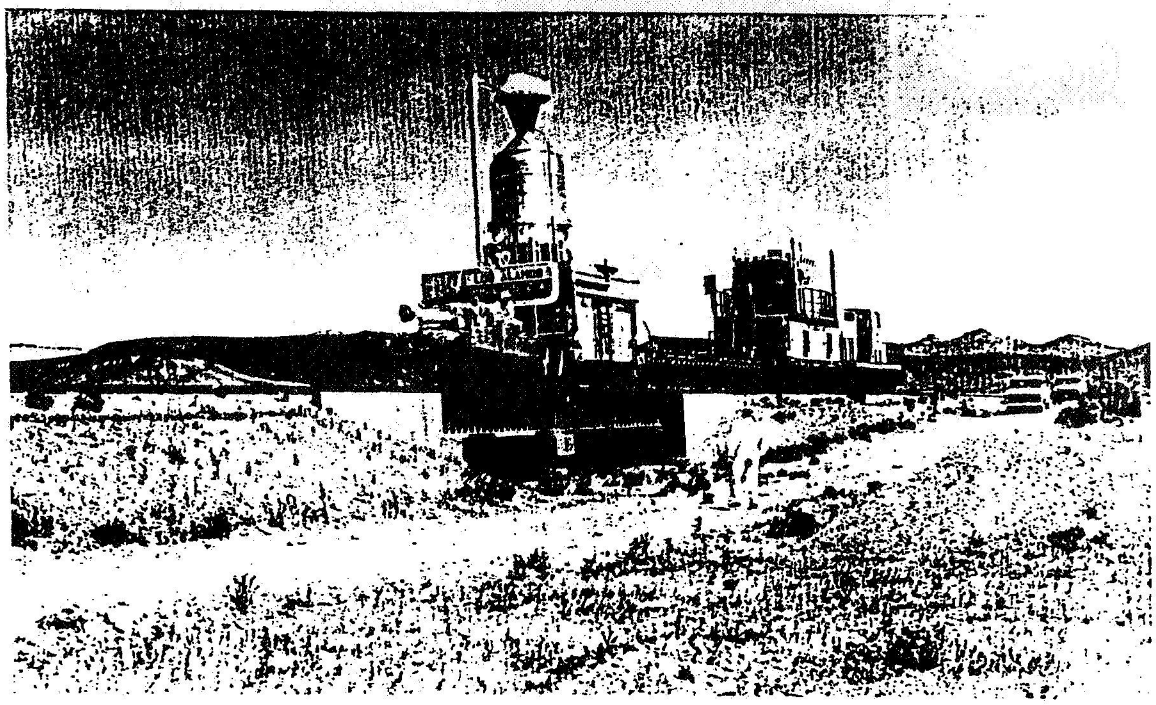


Figure 6

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Test Cell C

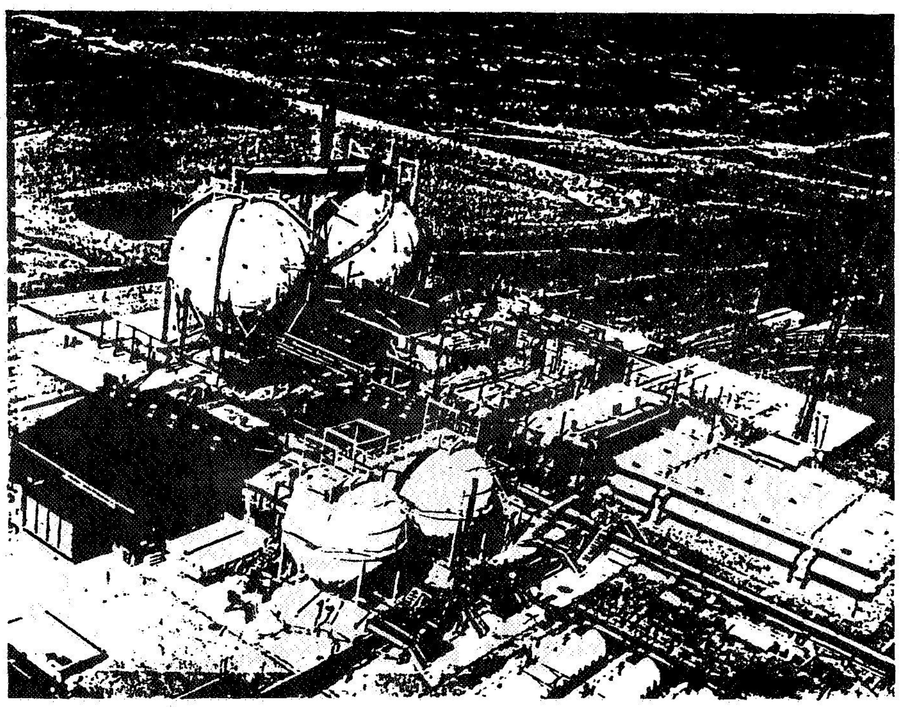

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Figure 7

NRX-A3 BEING PREPARED FOR TEST FIRING AT THE NRDS
JACKASS FLATS, NEVADA

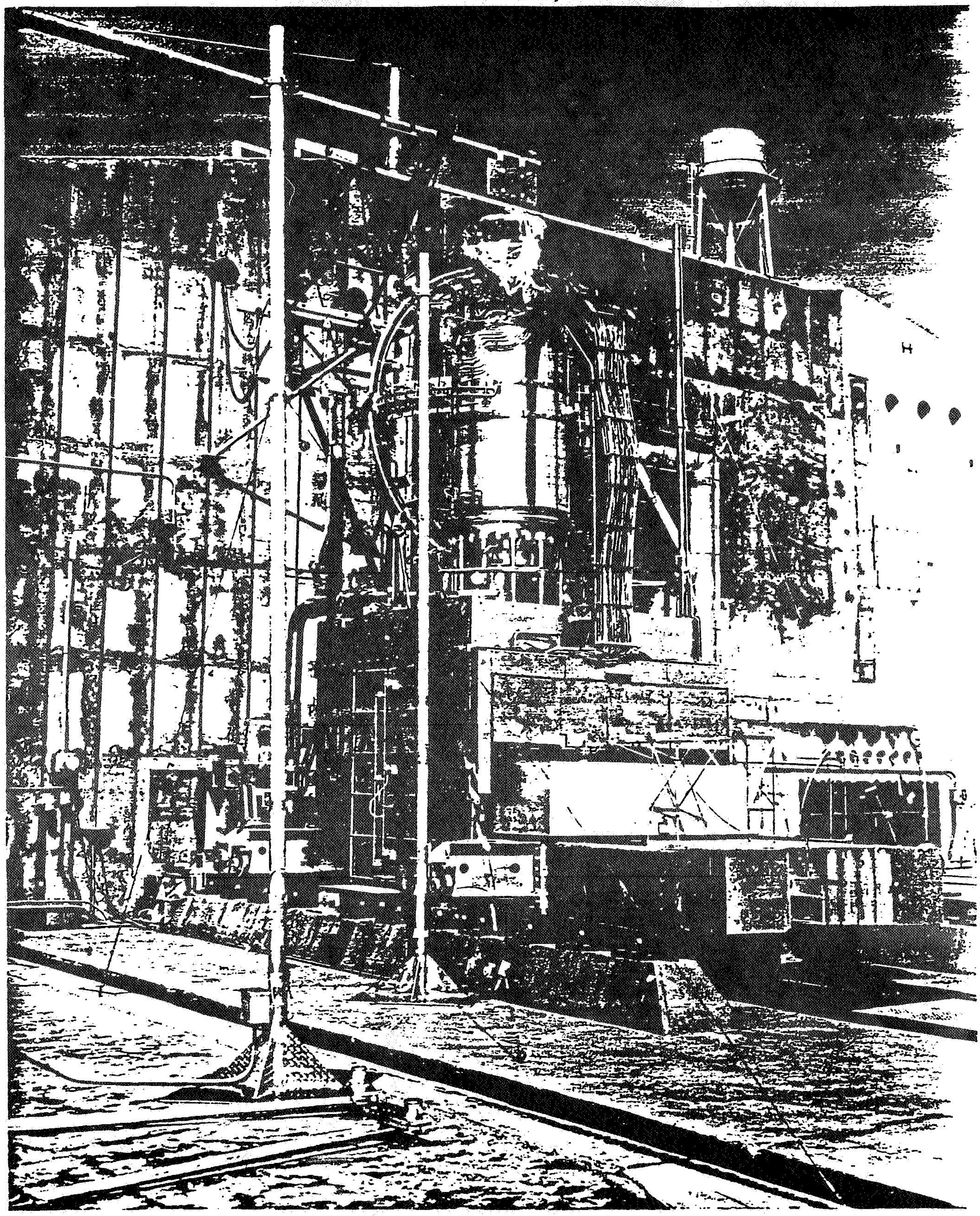

Figure 8

NRX-A6 TEST FIRING (DEC. 13, 1967): APPROXIMATELY 62 MINS. AT 1124MWt

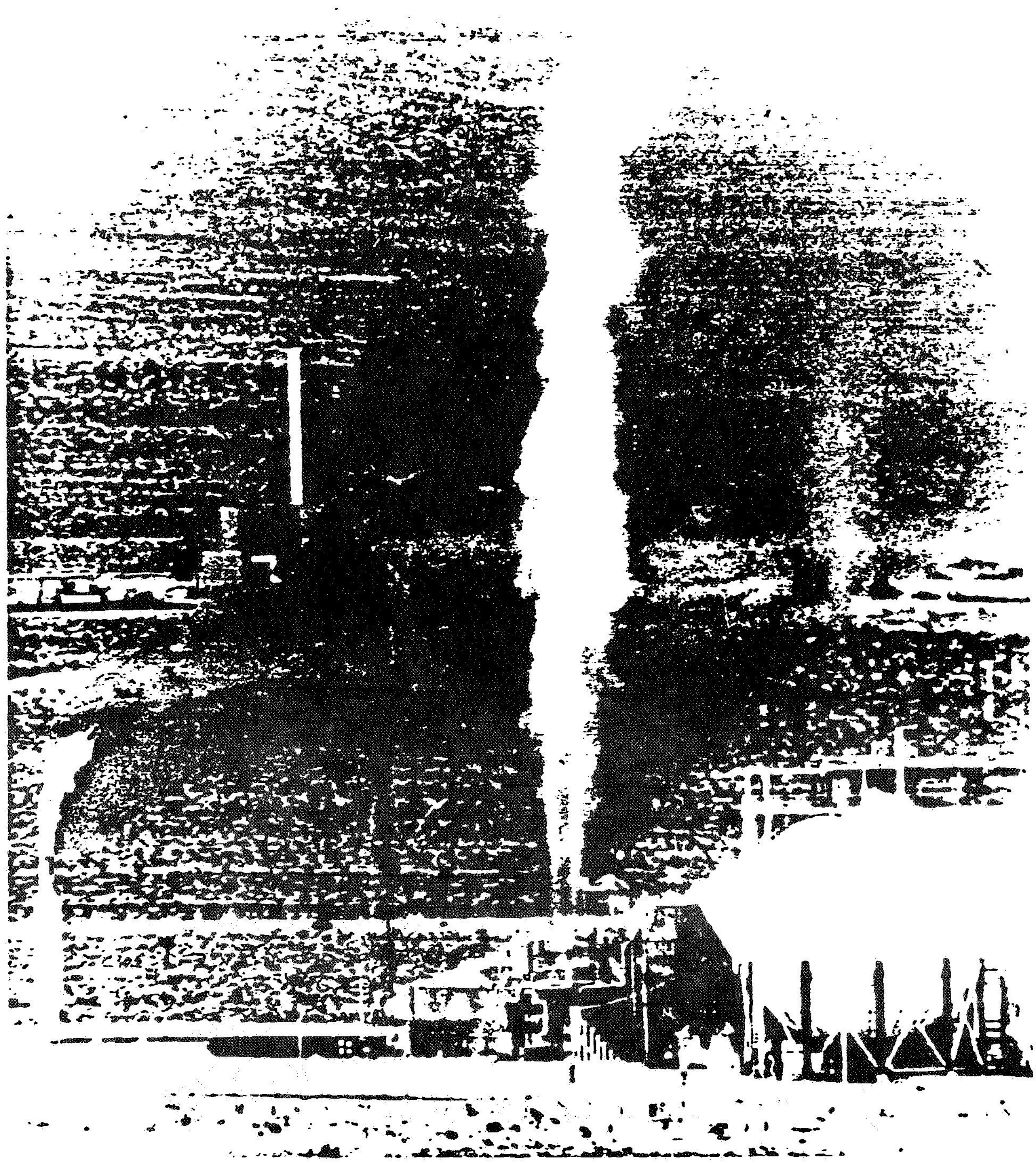

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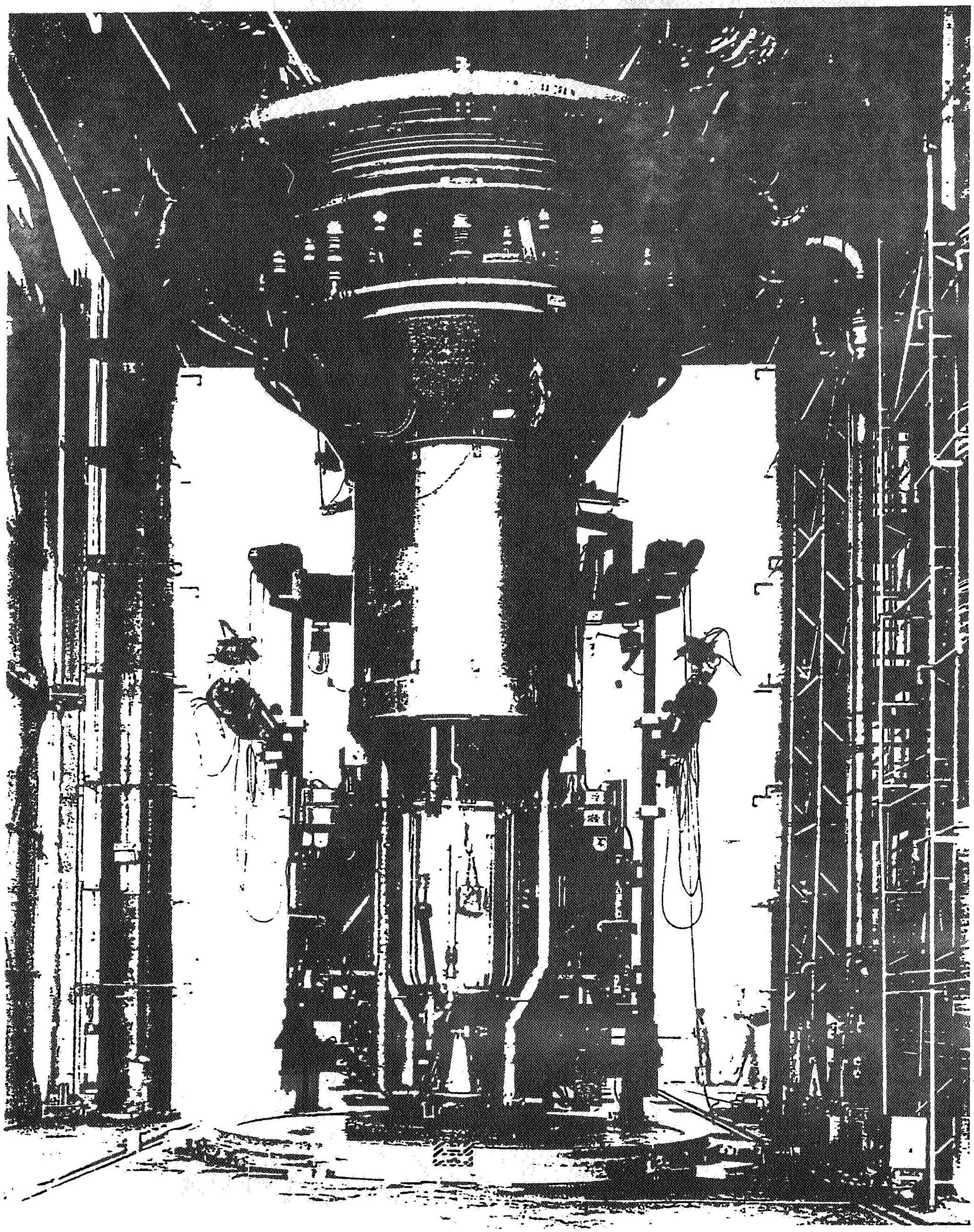
Figure 9

PROTOTYPE NERVA ENGINE - THE NRX/XE -



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OF POOR QUALITY

Figure 10

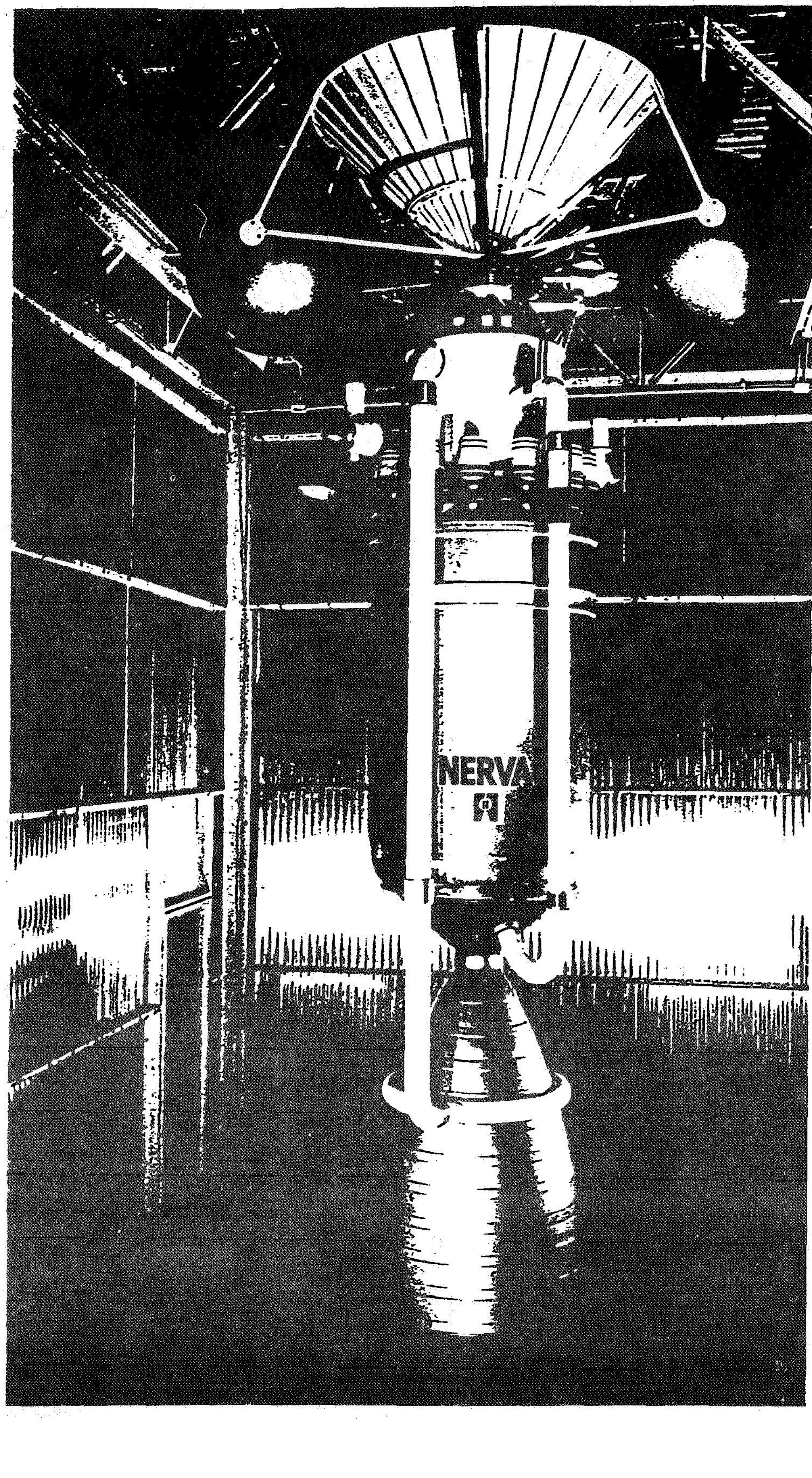

Figure 11

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NUCLEAR SUBSYSTEM COMPONENT MATURITY AND READINESS

	LEVEL OF MATURITY	READINESS
• FUEL		
- MATRIX	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS
- COMPOSITE	5	REQUIRES SOME R&D
- CARBIDE	4	REQUIRES SOME R&D
• FUEL CLUSTER		HOT END SUPPORT REQUIRES
- HARDWARE	5	ADDITIONAL DESIGN AND ANALYSIS
• AXIAL/LATERAL SUPPORT SYSTEMS	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS
• CORE PERIPHERY	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS
• REFLECTOR	5	ADDITIONAL DESIGN AND ANALYSIS REQUIRED
• CONTROL DRUM	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS
• CORE SUPPORT PLATE	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS
• INTERNAL DOME SHIELD	6	MATERIALS AND DESIGN READY FOR FLIGHT TESTS

ASSESSMENT BY WESTINGHOUSE ADVANCED ENERGY SYSTEMS FOR USE IN INEL'S "SAFE COMPACT NUCLEAR PROPULSION DESIGN STUDY FINAL REPORT" PREPARED BY THE AIR FORCE ASTRONAUTICS LABORATORY, SEPTEMBER 1988.

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Figure 12

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NON-NUCLEAR SUBSYSTEM COMPONENT MATURITY AND READINESS

- HYDROGEN TURBOPUMPS: AN EXTENSIVE DATABASE DEVELOPED SINCE NERVA SHOULD ALLOW SIGNIFICANT REDUCTIONS IN WEIGHT, INCREASES IN RELIABILITY AND REDUCED DEVELOPMENT TIME FOR NTR APPLICATIONS
  - SSME: 72.6 KG/S @ 7040 PSI, 350 KG TOTAL MASS
  - NERVA: ~ 40 KG/S @ 1360 PSI, 243 KG TOTAL MASS

- REACTOR PRESSURE VESSEL: AEROSPACE DEVELOPMENT PROGRAMS (BOEING'S SST, SPACE SHUTTLE) HAVE ADVANCED TITANIUM FORMING AND WELDING TECHNOLOGY TO THE POINT THAT FABRICATION OF A HIGH STRENGTH, LOW MASS, HIGH TEMPERATURE TITANIUM PRESSURE VESSEL SHOULD BE POSSIBLE

- NOZZLE DESIGN AND COOLING: TYPICAL NOZZLE DESIGNS NOW CAPABLE OF ~ 98% THEORETICAL EFFICIENCY WITH PERFORMANCE SIGNIFICANTLY GREATER THAN THAT USED ON NERVA

SSME: T_{ex} ~ 3116°K, Pc ~ 3150 PSI, NOZZLE ASSEMBLY MASS ~ 600 kg, HEAT FLUX CAPABILITY ~ 16.4 KW/CM^2 (HYDROGEN REGENERATIVE COOLING)

NERVA: T_{ex} ~ 2500-3000°K, Pc ~ 450 psi, NOZZLE ASSEMBLY MASS ~ 1050 kg, HEAT FLUX CAPABILITY ~ 4.1 KW/CM^2

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Figure 13

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75 klbf NERVA-TYPE ENGINE CHARACTERISTICS*

PARAMETERS	'72 NERVA**	"STATE-OF-THE-ART" NERVA DERIVATIVES**				
ENGINE FLOW CYCLE	HOT BLEED/ TOPPING	TOPPING (EXPANDER)				
FUEL FORM	GRAPHITE	GRAPHITE		COMPOSITE		CARBIDE
CHAMBER TEMP. (K)	2350-2500	2500	2350-2500	2700		3100
CHAMBER PRESS. (psia)	450	500	1000	500	1000	1000
NOZZLE EXP. RATIO	100:1	200:1	500:1	200:1	500:1	500:1
SPECIFIC IMPULSE(s)	825-850/ 845-870	875	850-885	915	925	1,020
ENGINE WEIGHT+(kg)	11,250	7,721	8,000	8,483	8,816	9,313
ENGINE THRUST/WEIGHT (W/INT. SHIELD)++	3.0	4.4	4.3	4.0	3.9	3.7

* INFORMATION PROVIDED BY LERC PROPULSION TOC WITH SAIC AND WESTINGHOUSE
** ENGINE WEIGHTS CONTAIN DUAL TURBOPUMP CAPABILITY FOR REDUNDANCY
\+ W/O EXTERNAL DISK SHIELD
++THRUST-TO-WEIGHT RATIOS FOR NERVA/NDR SYSTEMS ARE ~5-6 AT THE 250 klbf LEVEL

SPACE EXPLORATION INITIATIVE OFFICE

Table 1

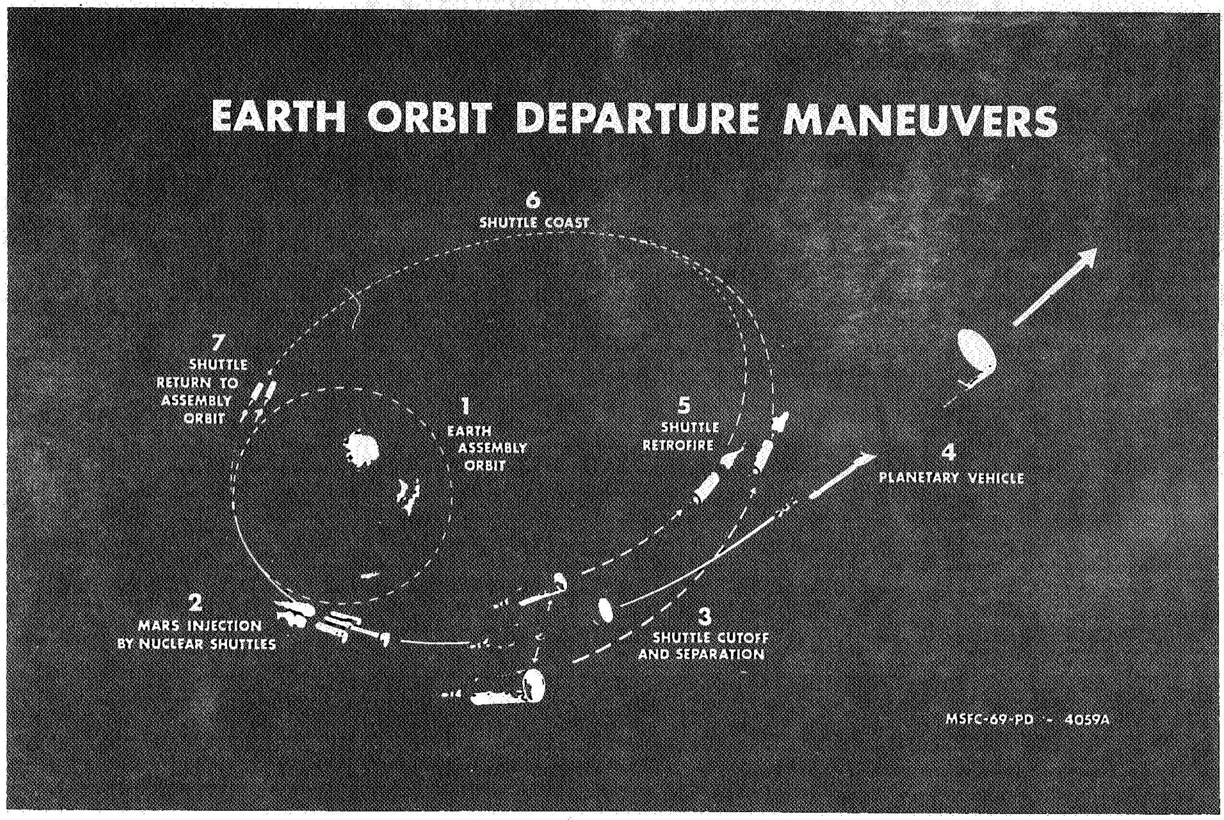

Figure 14

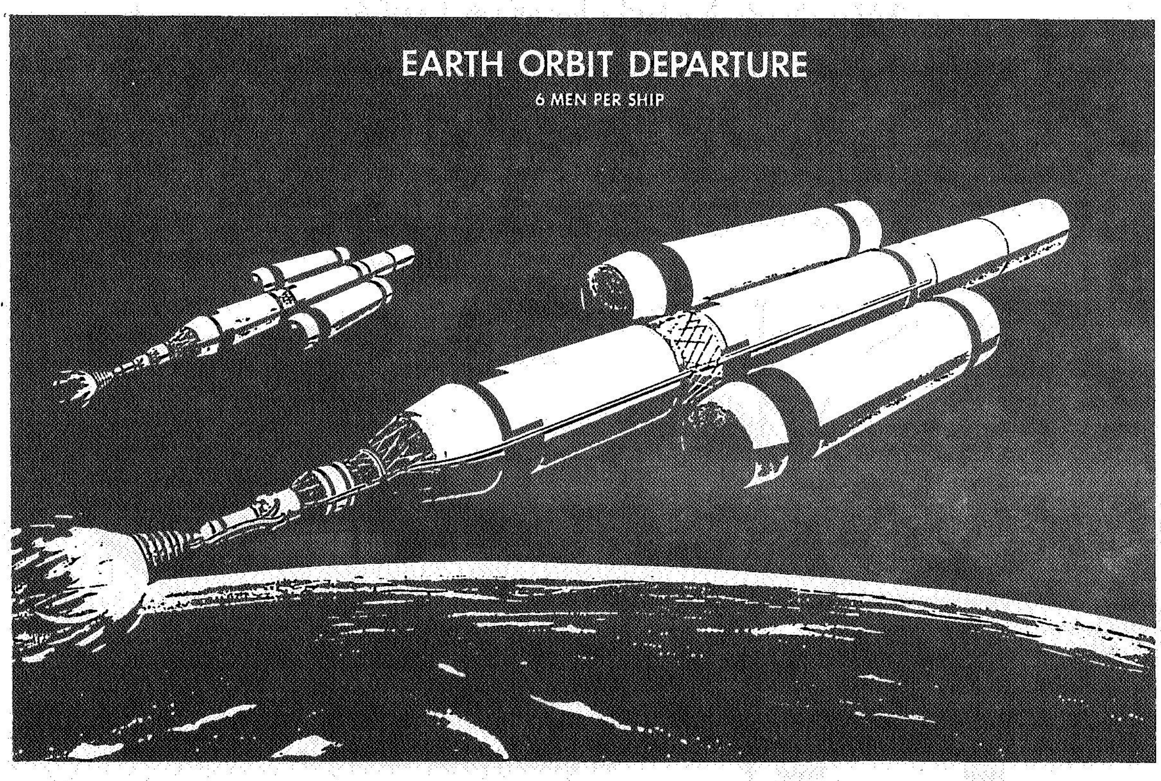

Figure 15

2016 NTR Reference Trajectory

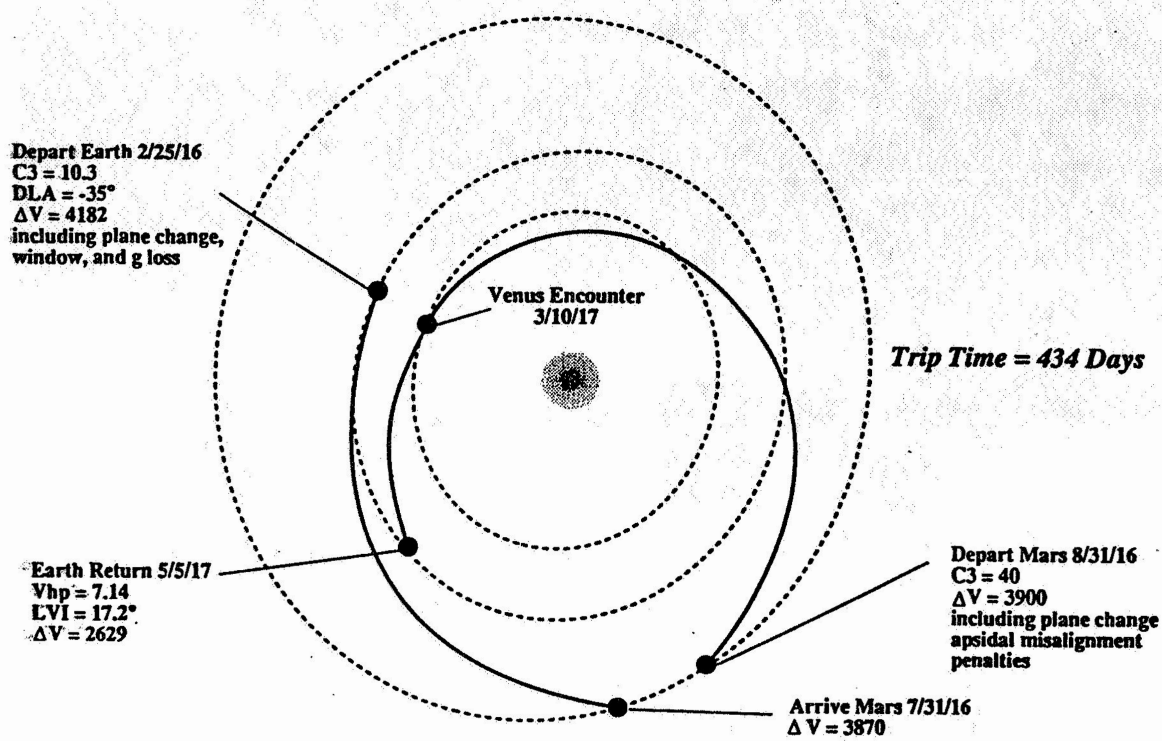

Figure 16

2016 NTR Vehicle Mission Profile

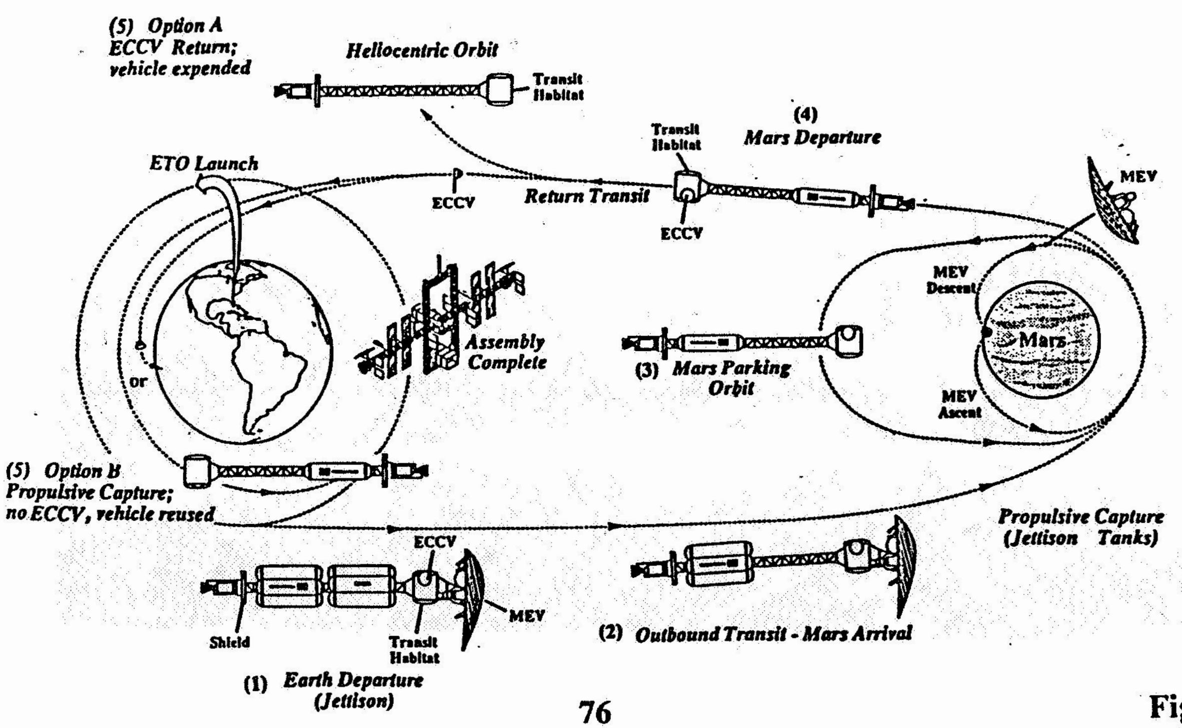

Figure 17

2016 MARS MISSION ASSUMPTIONS/GROUND RULES

GENERAL

- PAYLOAD OUTBOUND: 73.12 t MARS EXCURSION MODULE (MEV); 34.94 t MARS TRANSFER VEHICLE (MTV); 7.00 t EARTH CREW CAPTURE VEHICLE (ECCV)
- PAYLOAD RETURN: 34.94 t MTV; 7.00 t ECCV (USED ONLY W/"EXPENDABLE MODE"); 0.50 t MARS RETURN SAMPLES
- PLANETARY PARKING ORBITS: 407 km CIRCULAR (EARTH DEPARTURE); 250 km x 1 SOL* (MARS ARRIVAL/DEPARTURE); 500 km x 24 hr+ (EARTH ARRIVAL)
- g-LOSSES MODELED FOR EARTH DEPARTURE ONLY
- EARTH DEPARTURE PLANE CHANGE ΔV PENALTIES:
  - 340 m/s (dla > 28.5°)
  - 100 m/s (dla < 28.5°)
- MARS APSIDAL ALIGNMENT ΔV PENALTIES: 560 m/s
- PLANETARY TRAJECTORIES OPTIMIZED FOR "ALL PROPULSIVE" MISSION SCENARIO. FOR 2016 OPPORTUNITY, TRIP TIMES RANGE FROM 120 TO 434 DAYS
- SINGLE BURN AND "3-BURN" PERIGEE DEPARTURES FROM EARTH EXAMINED

* 250 km x 33,852 km = 1 SOL ORBIT = 24.66 HOURS
\+ 500 km x 77,604 km = 24 HOUR ORBIT

Table 2

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PROPULSION SYSTEM/PROPELLANT/TANKAGE ASSUMPTIONS

NTR	PROPELLANT	Isp(s)	USAGE
- PRIMARY	LH_2	850-1020	MAIN IMPULSE
- AUXILIARY	LH_2	500 (NERVA "IDLE MODE")	MID-COURSE CORRECTION
- AUXILIARY	STOR. BIPROP.	320	ATTITUDE/MID-COURSE

ENGINE DESIGN	Isp(s)	THRUST (kN/klbf)	ENGINE+ MASS (t)	EXT. SHIELD (t)* MASS (t)	TOTAL** MASS (t)
'90 GRAPHITE NERVA	850	334/75	8.00	4.5	19.4
'90 COMPOSITE NERVA	925	334/75	8.82	4.5	20.2
'90 CARBIDE NERVA	1020	334/75	9.31	4.5	20.7
'90 COMPOSITE PHOEBUS	925	1112/250	21.76	9.0	37.65

- RESERVE/COOLDOWN PROPELLANT/BOILOFF RATES: 2%/3%/.0.65 kg/m^2/mth
- PROPELLANT TANKS JETTISONED AFTER TMI AND MOC BURNS
- TANKAGE FRACTION (PERCENTAGE OF TOTAL PROPELLANT REQUIRED PER MANUEVER):
  - VARIES WITH TANK SETS: TMI (~ 13%), MOC (~ 15%), COMMON TEI/EOC (~ 16%)

\+ CHAMBER PRESSURE = 1000 psia, ε = 500:1
* ASSUMED VALUE - DETAILED CALCULATIONS REQUIRED TO VERIFY ADEQUACY/INADEQUACY
** INCLUDES MASS FOR RCS ATTITUDE CONTROL WHILE ON STATION, MAIN PROPELLANT FEEDLINE FROM TANK LINES TO ENGINE, RUN TANK, TRUSS, AND INTERSTAGE/THRUST STRUCTURE)

Table 3

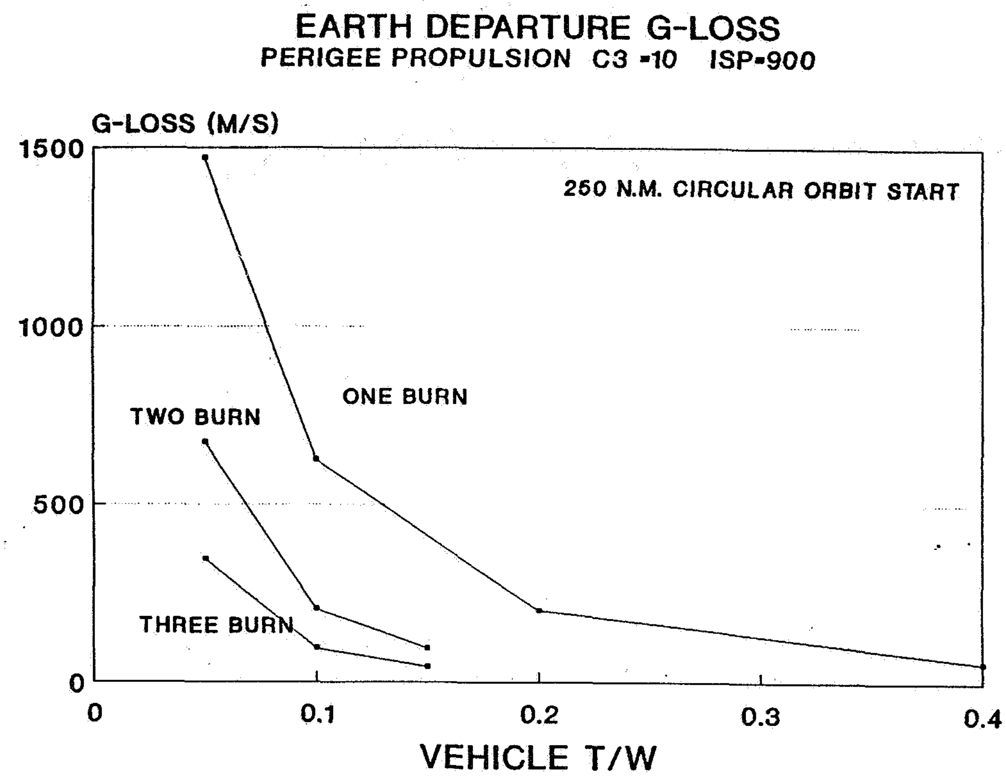

Figure 18

MARS MISSION BASELINE PERFORMANCE - 434 DAYS

	BOEING REF. MISSION	NASA REF. W/MOD.*	ALL-PROPULSIVE OPTIMIZED
DATES			
EARTH DEPARTURE	2/25/2016	2/25/2016	3/15/2016
MARS ARRIVAL	7/31/2016	7/31/2016	8/19/2016
MARS DEPARTURE	8/31/2016	8/31/2016	9/19/2016
VENUS FLYBY	3/10/2017	3/10/2017	3/16/2017
EARTH ARRIVAL	5/04/2017	5/04/2017	5/23/2017
DEPARTURE/ARRIVAL ENERGY			
EARTH DEPARTURE C_3 (KM^2/SEC^2)	10.34	10.34	14.07
MARS ARRIVAL V_H (KM/SEC)	6.82	6.82	5.31
MARS DEPARTURE V_H (KM/SEC)	6.30	6.30	7.11
EARTH ARRIVAL V_H (KM/SEC)	7.30	7.30	5.56
IMLEO (t)	735	766	613

* Al/Li VERSUS SiC/Al METAL MATRIX TANKS ON BOEING REF., G-LOSS AS FUNCTION OF VEHICLE THRUST-TO-WEIGHT (FROM LOOK-UP TABLE) VERSUS ASSUMED CONSTANT VALUE (200 m/s), ETC.

Table 4

2016 NTR MARS VEHICLE SIZE COMPARISON
(OPTIMIZED VS. NON-OPTIMIZED TRAJECTORIES-COMPOSITE FUEL/Isp=925s)

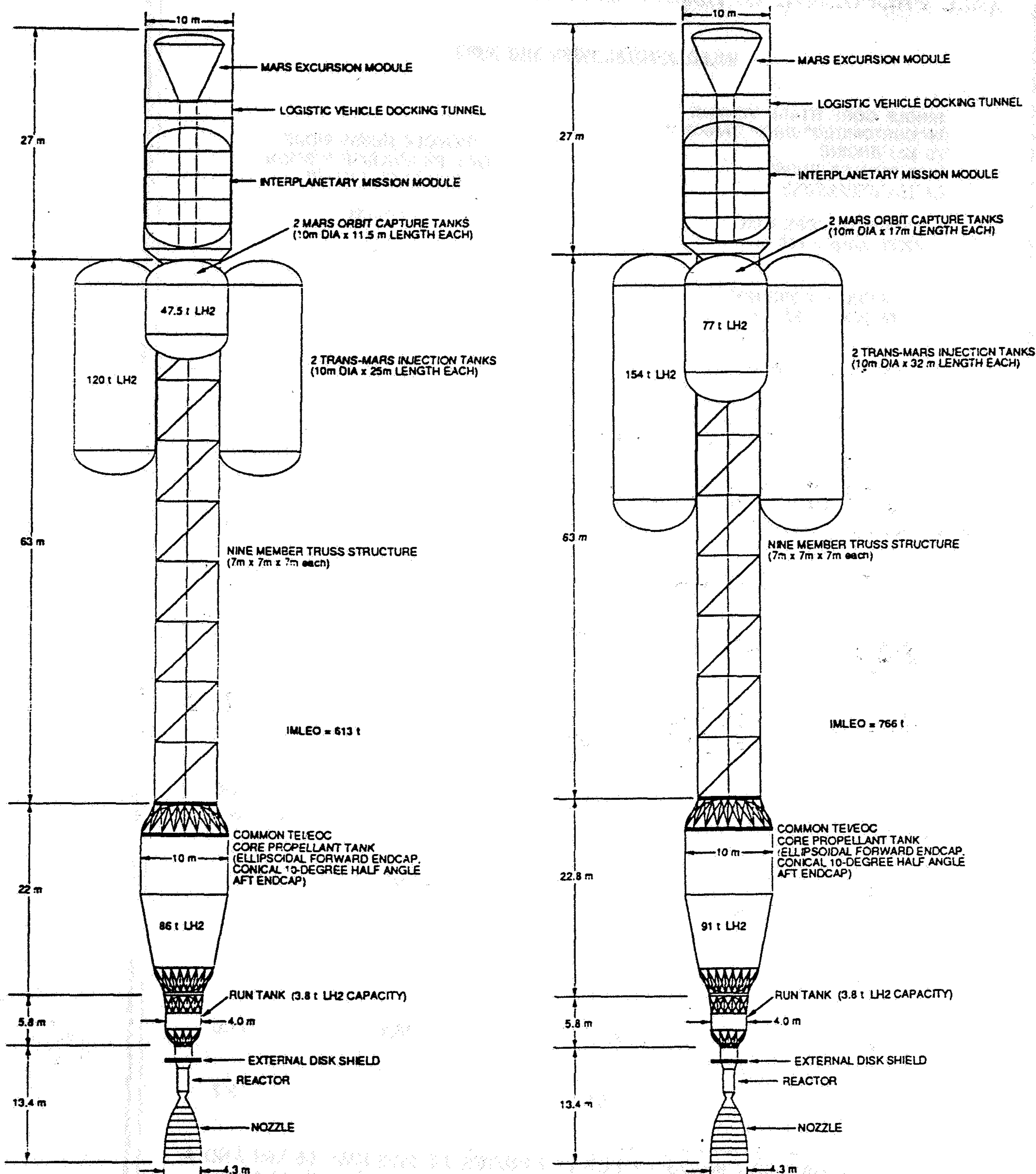

Figure 19

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NERVA-DERIVATIVE ENGINE*/ISP TRADE RESULTS
(ALL PROPULSIVE OPTIMIZED 2016 MARS MISSION - 434 DAYS)+

IMLEO (t)/TOTAL BURN TIME (HRS)

SINGLE CORE STAGE VEHICLE W/"CUSTOMIZED" DROP TANKS ** 75 klbf ENGINE W/"3 PERIGEE BURN" EARTH DEPARTURE	"VEHICLE REUSE MODE" (ALL PROPULSIVE MISSION W/O ECCV RETURN)
1. GRAPHITE CORE NDR (2350 K/Isp = 850 s)	725/3.38
2. COMPOSITE CORE NDR (2700 K/Isp = 925 s)	613/2.99
3. CARBIDE CORE NDR (3100 K/Isp = 1020 s)	518/2.64

+ REFERENCE MTV (90 DAY STUDY): CHEM/AB IMLEO=752t FOR ECCV RETURN/=830t FOR PROPULSIVE EARTH CAPTURE
* (CHAMBER PRESSURE = 1000 psia, ε = 500:1)
** DROP TANKS ASSUMED TO BE CYLINDRICAL W/ROOT2 ELLIPSOIDAL DOMES; DIA.=10M, LENGTH CONSTRAINED TO BE ≤35 M

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Figure 20

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INDIVIDUAL BURN DURATION FOR "ALL PROPULSIVE" OPTIMIZED 2016 MARS MISSION - 434 DAYS

DURATION (mins)	75 klbf GRAPHITE	75 klbf COMPOSITE	75 klbf CARBIDE	250 klbf COMPOSITE
TMI (TOTAL/# PERIGEE BURNS)	~122.1/3	~104/3	~87.8/3	38.2/1
MOC	40.0	36.8	33.8	13.4
TEI	30.0	28.0	26.1	11.0
EOC	7.1	6.9	6.7	2.7

NOTE: NRX-A6 RAN CONTINUOUSLY FOR 62 MINUTES AT 1125 MWt, 55 klbf AND A HYDROGEN FUEL EXIT TEMPERATURE ≥ 2550 K (DECEMBER 1967)
NRX-XE ACCUMULATED APPROXIMATELY 115 MINUTES OF POWERED OPERATION DURING 28 ENGINE RESTART TESTS OCCURRING BETWEEN MARCH AND AUGUST 1969

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Table 5

2016 NTR MARS VEHICLE SIZE COMPARISON
(OPTIMIZED TRAJECTORIES - GRAPHITE, COMPOSITE, CARBIDE FUELS)

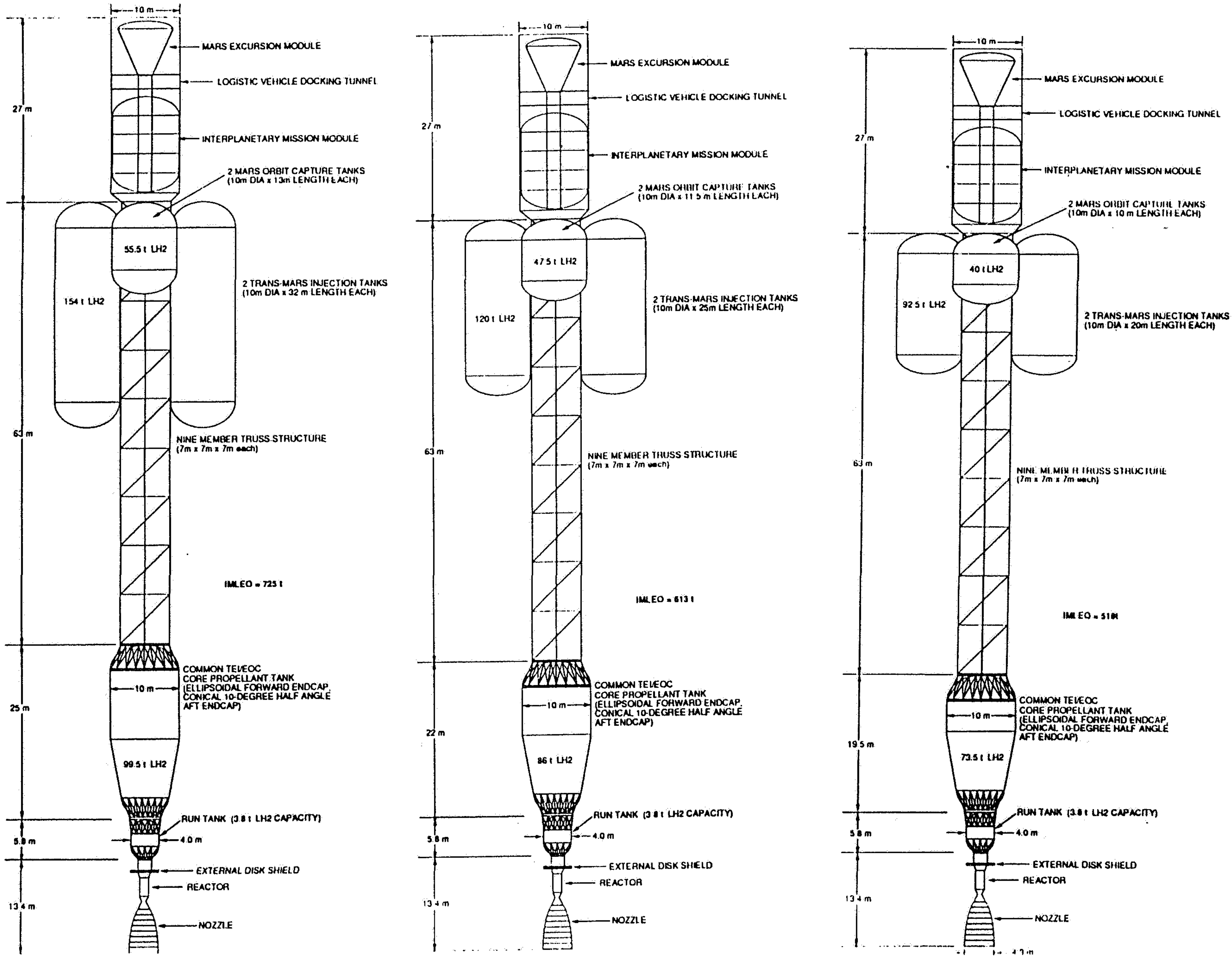

Figure 21

2016 NTR MARS VEHICLE SIZE COMPARISON
(OPTIMIZED TRAJECTORIES - COMPOSITE FUEL/75 klbf & 250 klbf)

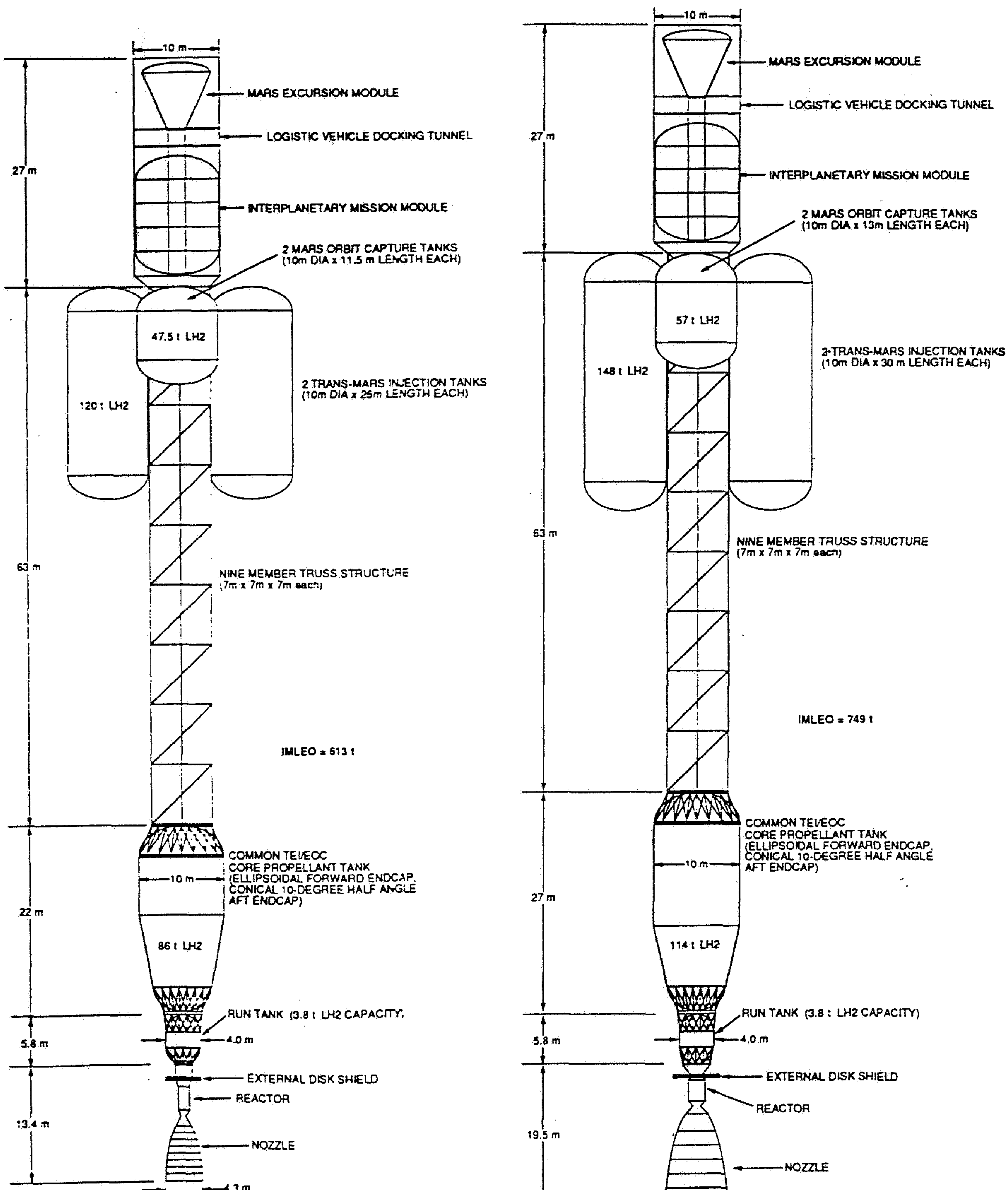

Figure 22

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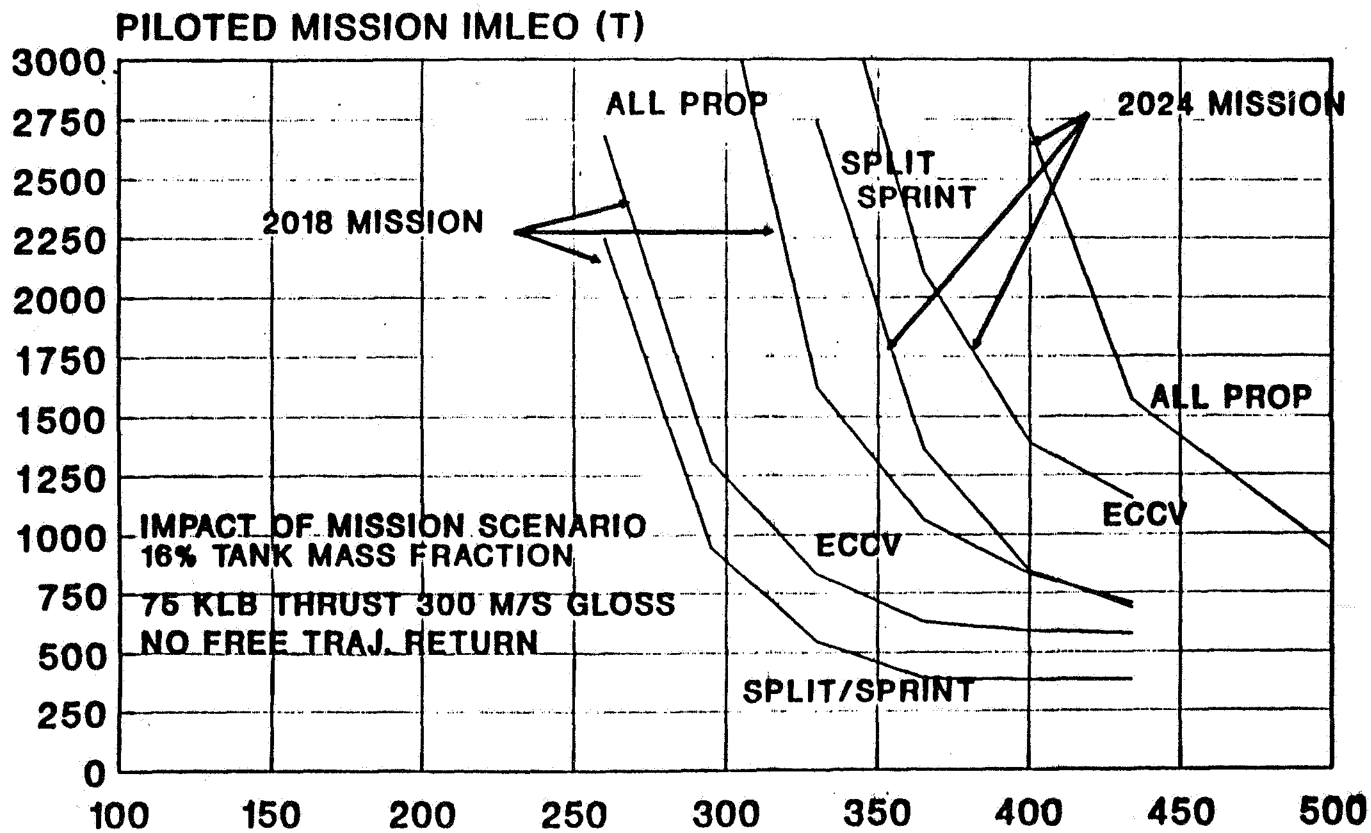
NTP ENGINE PERFORMANCE POTENTIAL
925 SEC ISP -- 4:1 ENGINE T/W

PRELIMINARY



MISSION ROUND TRIP TIME

FOR SPLIT/SPRINT PILOTED VEHICLE HAS RETURN PROPS

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Figure 23

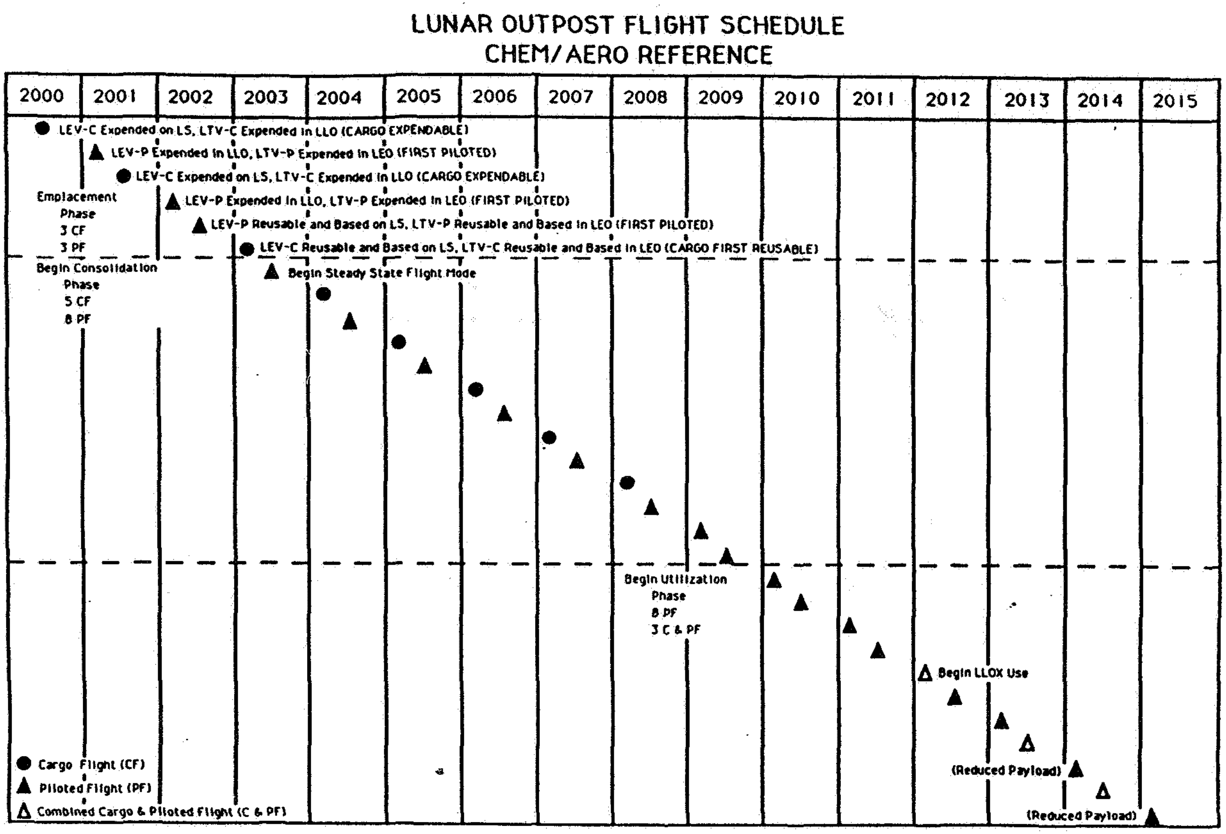

Figure 24

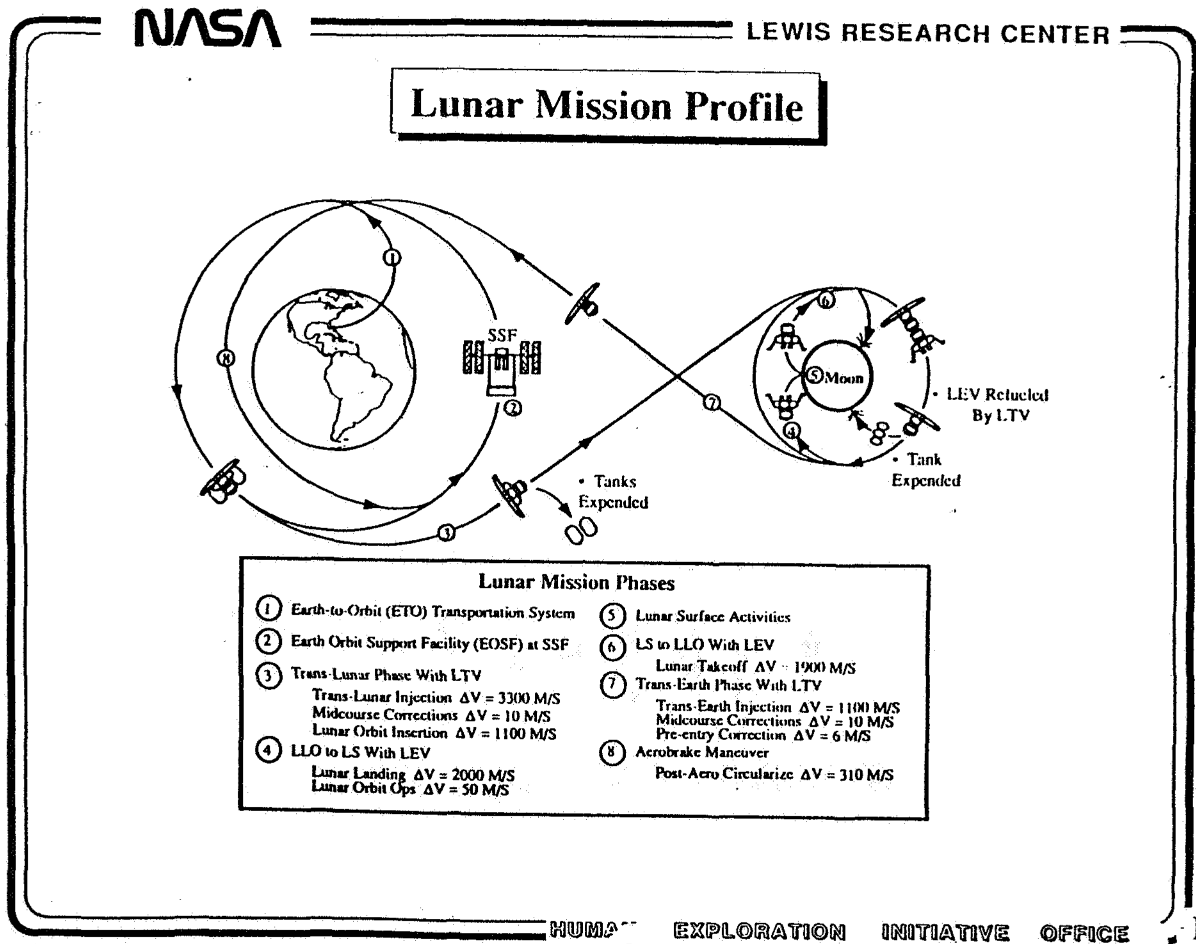

Figure 25

ASSUMPTIONS FOR PERFORMANCE CALCULATIONS

- APPROACH:
  - REFERENCE SCENARIO & ASSUMPTIONS FROM 90 DAY STUDY
  - VARY ONLY AS REQUIRED
- SPECIFIC ASSUMPTIONS
  - LEV AND PAYLOADS PER REFERENCE CHEM/AERO CASE; LEV USES CHEMICAL PROPULSION IN ALL CASES
  - MAJOR IMPULSES AND NAVIGATION BUDGETS PER REFERENCE CASE
  - TOTAL FLIGHT TIME PER LTV TRIP IS 30 DAYS; SIZES TANK INSULATION AND BOILOFF RATES
  - HYDROGEN TANKAGE FACTOR IS 9% (WELDALITE ALUMINUM-LITHIUM); ALSO, ADD INSULATION AND 10% OF TANKS FOR STRUCTURE
  - ALLOWANCE FOR UNUSED PROPELLANT INCLUDES NTR COOLDOWN AT 3.5% (ASSUMES SOME USEFUL THRUST FROM COOLDOWN BURNS)

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Figure 26

NTR SYSTEM DEFINITION

- BASE DESIGN IS 75,000 LBF THRUST NERVA-DERIVATIVE ENGINE WITH
  - (U,Zr)C-COMPOSITE FUEL ELEMENTS (NUCLEAR FURNACE TESTED)
  - 2700 K CHAMBER TEMP; 500 PSI CHAMBER PRESSURE
  - ISP = 900 SECONDS
  - 60 RESTARTS/10 HOUR LIFETIME (TO MAX OF 5 MISSIONS INCL. DISPOSAL)

NTR COMPONENT	MASS (KG)	SOURCE	COMMENTS
REACTOR	5,662	WESTINGHOUSE	NERVA-derivative
INTERNAL SHIELD	1,527	WESTINGHOUSE	*
NOZZLE	867	MMAG*	200:1 expansion 7.4 m length
NON-NUCLEAR HARDWARE	1,194	MMAG*	Incl. pumps, valves, lines, thrust structure, etc., 2% contingency
Subtotal: Engine	9,250		F/W = 3.69
EXTERNAL SHIELD	4,545	NERVA DESIGN*	To be resized based on final design
Total NTR System	13,795		

* = Additional analysis to be performed as part of this study

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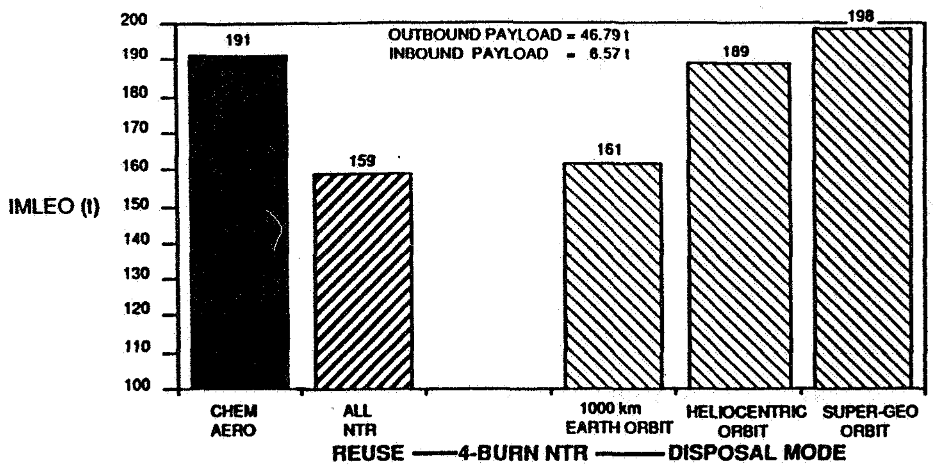
Figure 27

IMLEO REQUIREMENTS FOR FIRST PILOTED MISSION



OTHER NTR OPTIONS:

DESCRIPTION	IMLEO (t)	DISPOSAL MODE
1-BURN NTR	163	HELIOCENTRIC ORBIT VIA LGA
2-BURN NTR	153	LUNAR SURFACE IMPACT
2-BURN NTR	162	LUNAR SURFACE DELIVERY
3-BURN NTR	159	1000 km CIRCULAR EARTH ORBIT
3-BURN NTR	148	SOLAR CIRCULAR ORBIT

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Figure 28

SUMMARY OF MASS SAVINGS

2000 - 2015 FLIGHT SCHEDULE

	MASS DELIVERED TO LEO	SAVINGS
• CHEM/AERO REFERENCE CASE	5030 t	——
• ALL-NTR: 4-BURN LTV USE	4040	20%
• ALL-NTR: 3-BURN LTV USE	3853	23%
• PHASED NTR: 3-BURN LTV USE	4277	15%

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Figure 29

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DEVELOPMENT/PROGRAMMATIC REQUIREMENTS

GROUND TESTING

- ONE OF THE MOST IMPORTANT ASPECTS OF AN NTR OR SPACE NUCLEAR REACTOR DEVELOPMENT PROGRAM IS "PRE-FLIGHT" TESTING.
- THE GROUND TEST PROGRAM WILL COVER ESSENTIALLY ALL COMPONENTS AND SYSTEMS, BEGINNING WITH COMPONENT LEVEL TESTS AND PROCEEDING IN LOGICAL TEST STEPS TO THE FLIGHT SYSTEM DEMONSTRATION IN "HOT, FULL-UP" SYSTEM LEVEL TESTS.
- IN PARALLEL WITH COMPONENT AND SUBSYSTEM DEVELOPMENT IS A CONSTRUCTION AND CHECKOUT PROGRAM FOR THE NUCLEAR TEST FACILITY (NTF) WHERE THE INTEGRATED SYSTEM LEVEL TESTS WILL BE CONDUCTED. CANDIDATE DOE SITES INCLUDE THE NUCLEAR ROCKET DEVELOPMENT STATION (NRDS) AT JACKASS FLATS, NEVADA, OR THE IDAHO NATIONAL ENGINEERING LABORATORY (INEL).

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Figure 30

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REQUIRED FACILITY ACTIVITIES

- THE REACTOR CORE AND COMPLETE ENGINE SYSTEM WILL BE ASSEMBLED AT THE NTF IN A CLEAN ROOM ATMOSPHERE.
- COMPLETED ENGINE SYSTEMS WILL BE MOVED VIA A MOBILE TEST ASSEMBLY (MTA) FROM THE ASSEMBLY AREA TO THE TEST AREA.
- THE TEST SYSTEM WILL BE CONNECTED WITH ALL NECESSARY SUPPORT SYSTEMS AT THE TEST CELL (E.G., CRYOGENIC TANK FARM, DECAY HEAT REMOVAL SYSTEM, ETC.).
- TESTS TO BE CONDUCTED INCLUDE COLD FLOW TESTS, STARTUP TRANSIENTS, RAMPS TO INTERMEDIATE HOLD POINTS, FULL POWER OPERATION, SHUTDOWN, AND COOLDOWN.
- ENGINE EXHAUST IS CONTAINED AND PROCESSED WITHIN AN EFFLUENT TREATMENT SYSTEM WHICH DIRECTS HYDROGEN AWAY FROM THE ENGINE SYSTEM, REMOVES FISSION PRODUCTS AND DISPOSES OF THE HYDROGEN IN A SAFE MANNER.
- THE TESTED RADIOACTIVE ENGINE IS MOVED TO A HOT CELL FACILITY FOR POST-TEST EXAMINATION OF THE FUEL AND COMPONENTS.

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Figure 31

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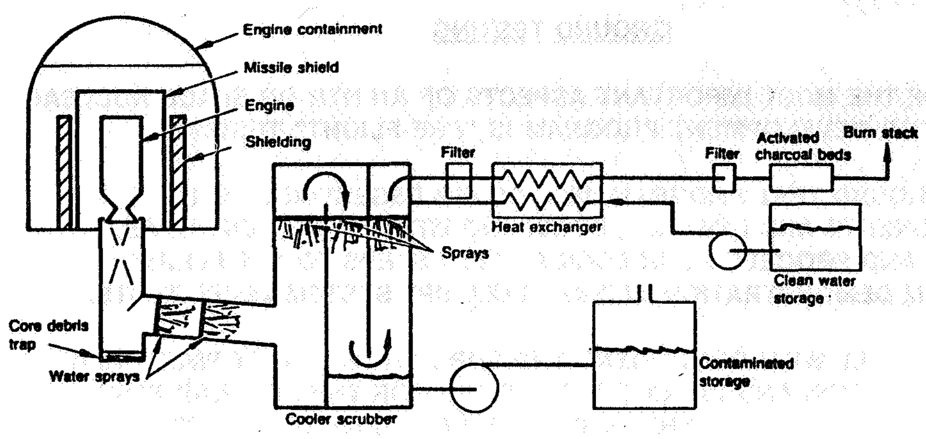
EFFLUENT TREATMENT SYSTEMS

ASAO



SCHEMATIC OF TEST CELL SHOWING SYSTEMS FOR REMOVING SOLUBLE FISSION PRODUCTS, PARTICULATES, AND NOBLE GAS FROM THE ENGINE EXHAUST

Figure 32

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TESTING IN TUNNELS

1. A CONTAINMENT OPTION FOR CONSIDERATION IS TO EXHAUST THE ENGINE INTO A LARGE UNDERGROUND TUNNEL

2. SUCH TUNNELS ARE ROUTINELY CONSTRUCTED AT THE NEVADA TEST SITE FOR CONTAINMENT OF NUCLEAR WEAPONS TESTS (SEVERAL TUNNELS ALREADY EXIST WITHIN A MILE OR TWO FROM NRDS)

3. TUNNELS CAN BE EVACUATED AND USED TO COLLECT THE ENGINE EFFLUENT

4. FLEXIBLE EFFLUENT SCRUBBING TIME (CLEANUP OF EXHAUST GASES CAN PROCEED AT SLOWER RATES (LOWER MASS FLOWS) THAN THE ENGINE EXHAUST MASS FLOW RATE)

5. NO ENVIRONMENTAL CONTAMINATION IN THE EVENT OF OPERATIONAL ACCIDENT

6. TEST APPROVAL NOT FUNCTION OF WEATHER CONDITIONS

RICHARD J. BOHL
LOS ALAMOS NATIONAL LABORATORY

ADVANCED SPACE ANALYSIS OFFICE

Figure 33

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Nuclear Rocket Development Station
Site 400, Nevada Test Site

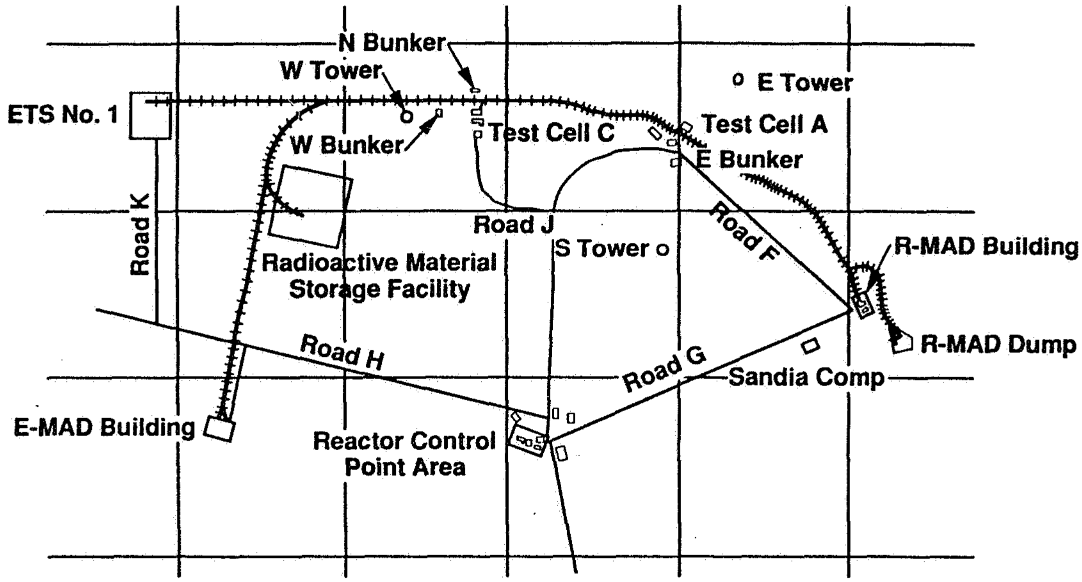

Figure 34

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TESTING AT THE NEVADA TEST SITE (NTS)

- VISIT TO DOE NEVADA OPERATIONS OFFICE ON JUNE 7, 8, 1990, WITH TOURS OF WEAPONS TESTS TUNNELS AND NUCLEAR ROCKET DEVELOPMENT STATION (NRDS) AT NTS BY NASA, DOE, AND INDUSTRY PERSONNEL
- SIGNIFICANT SITE ASSETS EXIST AT JACKASS FLATS
  - TEST CELL"C" AND ETS #1 IN GOOD AND FAIR CONDITION, RESPECTIVELY, (ESTIMATE COST TO REFURBISH ~ 10 TO 25 M$)
  - SEVERAL LARGE LH_2 DEWARS AVAILABLE (2 AT $5X10^5$ GAL. CAPACITY)
  - ENGINE MAINTENANCE ASSEMBLY AND DISASSEMBLY (EMAD) BUILDING IN EXCELLENT CONDITION FOR REMOTE HANDLING OF RADIOACTIVE COMPONENTS
  - INFRASTRUCTURE IN PLACE FOR HANDLING LARGE, COMPLEX, HAZARDOUS TEST OPERATIONS IS IN PLACE
  - FULLY FUNCTIONAL RAILROAD (JACKASS AND WESTERN R.R.)
  - 60,000 $FT.^2$ OFFICE BUILDING BEING RENOVATED/AVAILABILITY?
  - TWO TUNNELS ALREADY EXIST WITHIN FEW MILES OF EMAD

Figure 35

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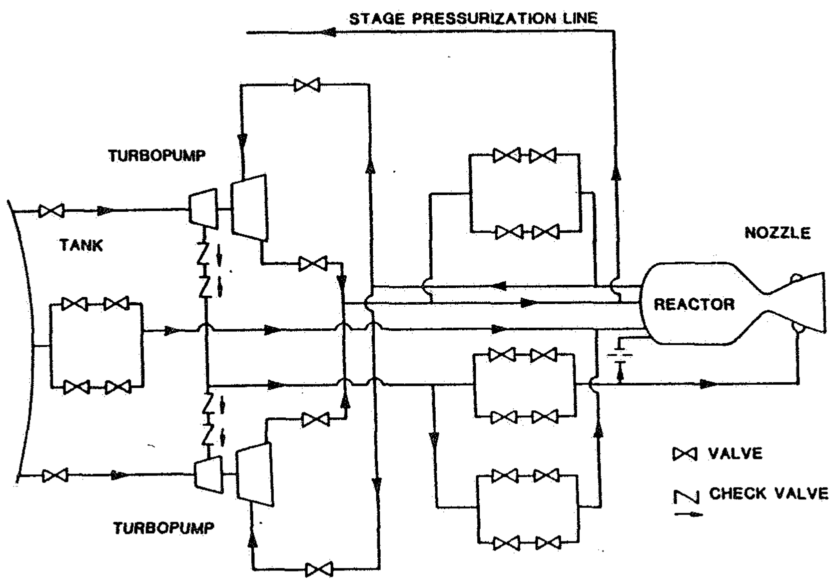
NERVA FLIGHT ENGINE COOLANT FLOW DIAGRAM



TO AVOID SINGLE-POINT FAILURES IN THE NERVA COOLANT CIRCUIT, REDUNDANT VALVES (26) AND TURBOPUMPS (2) WERE ADDED TO THE ENGINE DESIGN

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Figure 36

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NERVA SAFETY CONSIDERATIONS

- RELIABILITY AND SAFETY OF THE ENGINE DESIGN WERE OF PARAMOUNT IMPORTANCE DURING ALL PHASES OF THE NERVA PROGRAM.
- A MAJOR, HIGH PRIORITY EFFORT WAS DIRECTED TOWARD ELIMINATING FROM THE ENGINE DESIGN THOSE SINGLE FAILURES OR COMBINATIONS OF FAILURES WHICH COULD ENDANGER MISSION COMPLETION, THE FLIGHT CREW, THE LAUNCH CREW, OR THE GENERAL PUBLIC.
- PROBABILISTIC DESIGN AND FAILURE MODE AND EFFECTS (FM&E) ANALYSIS WERE INCLUDED IN THIS EFFORT.
  - EXAMPLES FROM THESE ANALYSES LED TO INCORPORATION OF DUAL TURBOPUMPS AND THE USE OF FOUR VALVES IN PLACE OF EACH SINGLE VALVE IN THE NERVA ENGINE DESIGN.
- WHERE NO PRACTICAL ENGINE DESIGN SOLUTIONS WERE FOUND FOR CREDIBLE SINGLE OR MULTIPLE FAILURES THAT COULD JEOPARDIZE CREW OR POPULATION SAFETY, APPROPRIATE COUNTERMEASURES, LARGE SAFETY MARGINS, AND ALTERNATIVE OPERATING MODES WERE USED.
  - OPTION FOR "EMERGENCY MODE" OPERATION DEVELOPED.

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Figure 37

TEMPORAL VARIATION OF DOSE RATE FOR MSFC-BOEING "NON-OPTIMIZED" REFERENCE 2016 NTR MISSION

Maneuver	1575 MW_t Engine Operating Time (minutes)	Mission Elapsed Time (days)
Trans Mars Injection	123.5	0
Mars Orbital Capture	62.3	156
Trans Earth Injection	24.1	187
Earth Orbital Capture	10.7	435

Event	Dose Rate* (Rem/hr)
Full Power Operation	7.2×10^{5}
Trans Mars Injection Plus 1 Day	1.1×10^{2}
Prior to Mars Orbital Capture	2.3×10^{-1}
Prior to Trans Earth Injection	1.9×10^{0}
Prior to Earth Orbital Capture	7.5×10^{-2}

*Dose point on axial midplane 100 feet from core centerline

REF. B. SCHNITZLER (INEL)

Figure 38

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OBSERVATIONS/CONCLUDING REMARKS

- ROVER/NERVA PROGRAMS ESTABLISHED A SIGNIFICANT DATA BASE ON WHICH THE '72 REFERENCE NERVA ENGINE WAS BASED
- EXPERIENCE ALSO OBTAINED IN OPERATING "FULL-SCALE" ENGINE FACILITIES (ALBEIT IN "OPEN CYCLE" MODE), HANDLING LARGE QUANTITIES OF LH_2 AND RADIOACTIVE SYSTEMS (E-MAD FACILITY), SAFETY, AND IN THE BEGINNINGS OF "EFFLUENT CLEAN-UP" (WITH THE NUCLEAR FURNACE)
- CONTINUED DEVELOPMENT OF CHEMICAL PROPULSION SYSTEMS HAVE ADVANCED SUBSTANTIALLY THE STATE-OF-THE-ART OF NON-NUCLEAR ENGINE COMPONENT (E.G., NOZZLES, TURBOPUMPS, ETC.)
- NTR PROPULSION IS <u>ENABLING</u> FOR MARS MISSIONS AND CAN BE <u>ENHANCING</u> FOR LUNAR MISSIONS PROVIDING BOTH IMLEO BENEFITS AND OPERATIONAL EXPERIENCE IN A RELATIVELY "NEARBY" SPACE ENVIRONMENT
- AN NTR PROGRAM WILL REQUIRE A LOT OF WORK - FACILITY REQUIREMENTS KEY FOLLOWED BY HIGH TEMPERATURE FUEL DEVELOPMENT
- PERFORMANCE PARAMETERS ACHIEVED IN ROVER/NERVA PROGRAM ARE WITHIN A FACTOR OF TWO OR LESS OF THOSE CURRENTLY BEING EXAMINED FOR SEI'S LUNAR AND MARS MISSIONS

SPACE EXPLORATION INITIATIVE OFFICE

Figure 39

N92-11092

NERVA UPGRADE: NON-NUCLEAR COMPONENTS

Stanley Gunn
Rockwell International, Rocketdyne Division

As Stan Borowski pointed out, the technology that did exist back in the 1960's and at the start of the 1970's under the ROVER/NERVA program was rather substantial, but there have been advances that have occurred since the initial design of the NERVA. Some of those advances were accomplished under the Phoebus program, the technology program that Los Alamos and Rocketdyne were involved with in Nevada.

Other advances have occurred in the development of the shuttle engine and related chemical rocket engines. What I would like to talk about is what would be realized if we were designing an engine today based upon the original accomplishments of the ROVER/NERVA program, but feeding in these advanced technologies; what would its characteristics be, what would it be able to accomplish?

Now to start off, I have set down some hypothetical requirements for a typical manned Mars mission. I'll try to highlight the areas that would influence the selection of the design details of the engine. As shown in Figure 1, I am assuming; 100,000 pound thrust engine with performance requirements in excess of 900 seconds; a maximum weight of 14,000 pounds without the shield, which is going to be a bit of a challenge to achieve; a full performance operating range of 50 percent thrust at full Isp up to 110 percent. Then, reflecting the concern to have a very reliable system, we had dual turbopumps with a pump-out capability that would give us a capability of operating at 70 percent of rated thrust at full Isp. (Incidentally, we ran Phoebus 2A with dual turbopumps).

Further I am going to assume that we are going to be able to engineer the pumping system so it will be able to take hydrogen as a saturated liquid from the tank, accelerate it to one velocity head (which means that we are going to be ingesting vapor), and pump it to the full requirements of the reactor in terms of pump outlet pressure and flow rate.

As far as the maximum operating time of two hours, that comes, in part, from a belief that by getting the engine thrust up to about 100,000 pounds for the typical Delta V's that we have been talking about here (fairly fast trip times), we will be able to limit the burn time, of the engine that runs the longest, to two hours. I have done that because I wanted to tie it back to what was accomplished in Nevada and the nuclear furnace, where fuel elements were run for approximately two hours.

I have assumed 6 restarts, and that takes into account using one engine for a number of maneuvers, and a transition from flow initiation to full thrust of 30 seconds. That goes with a ramp rate of something like 100-150 degrees per second coming up in temperature and thrust, with transition from 50% thrust to cut-off of about 30 seconds, and a maximum core temperature that we have to remove afterheat of 1800 degrees R.

There were a number of reactors that were examined back in the 1960's and a number that are now currently under examination. Back in the 1960's, we had the solid core reactor, which is typified by the ROVER/NERVA program, and there were some fast metallic systems that were looked at.

We did some engineering design studies of engine systems incorporating the GE 710 reactor, and also the Argonne National Lab had a similar fast metallic concept. We also did a design study with Frank Rom here on an engine based upon the utilization of tungsten 184, and the use of water moderation to provide very attractive engine cycles.

The presentation that I am limiting myself to today is the solid core reactor - ROVER/NERVA. I would like to say a couple things about the expected performance as shown in Figure 2.

We are talking now about temperatures in the range of 4500 degrees R to about 5580. The epsilons (nozzle expansion ratio) show what we can expect for the performance of a high pressure system. This includes nozzle losses from the gas kinetics and nozzle boundary layer. Divergence effects are also included. For a condition of an epsilon of 500 and a gas temperature of 4860 R or 2700 degrees Kelvin, the Isp is on the order of 920-925 seconds. You can see from this chart that there is not a lot to be gained by going to expansion ratios higher than 500. We are collapsing down to 800 to 1000 at almost the same value.

Now, if you want to look a little bit closer at how those numbers came about, there is a series of comparisons that might be of interest. A 250 K engine at a chamber pressure of 1000 psi, has a theoretical performance of 1029 seconds. If you take into account the kinetics of what's occurring in the expansion process, it drops it down to 1028 seconds. The boundary layer losses drop you down to 1014, and the divergence effects drop you down to 1011 seconds.

It's important to look at the boundary layer effects, because if we go next to a 75 K engine, where we have less flow and therefore, more boundary layer effect, we've dropped it down to 1010, with divergence of about 1007. This is all for the case of 3100 degrees Kelvin that we have examined here.

Now if we go clear on down to a very low pressure to take advantage of the increase due to dissociation and reassociation, here is what your numbers come down to. The theoretical performance is very high. But as you examine what happens in the kinetics (the recombination, relaxation), you find that you drop down to about 1372 seconds -- this one is for 7,000 degrees R, -- and you suffer losses down to about 1300 for the 7,000 case. When you take a look at that same effect based upon 3100 K, you find that the performance is somewhat higher than the 1020, but not a lot.

Now, I would like to talk a little bit about the selection of cycle. In this particular presentation, we have made Isp our "God." We are trying to find out what we can do to get the maximum Isp from the temperature, and so you will see this emphasized in the charts that follow.

One of the first things you would like to do, given a certain Isp or, rather, temperature, is to use an expander cycle. The designer has several approaches available to be able to accomplish the circuitry of the flow to get the temperature of the hydrogen, up high enough to be able to drive the turbine. The obvious reason for this is you don't want to pay a penalty in terms of the Isp by having less than full temperature in all the gas.

In Configuration A in Figure 3, we have taken a portion of flow down through the tie tubes on up to the point where it is going to join some flow that has come down through the nozzle. It joins the flow that has been split off and that goes up through the reflector.

The goal is to get the temperature of all the gas coming into the core as high as possible, to maximize the amount of heat that the fuel elements can give to the hydrogen, which will allow you to go to as high a thrust as possible. That is a key point in being able to raise your thrust-to-weight ratio: get the temperature up so that the full power of the reactor can heat more working fluid.

Now in Configuration B in Figure 3, we have done something a little differently. What we have done here is assume that we can get all the heat we need to drive the turbine through the tie tubes. This allows us to minimize the heat pickup up through the nozzle and then up through the reflector. However, remember I said I want to make Isp my "God" here. Any heat that is transferred to the nozzle up to this point is a loss. It's taking enthalpy out of the expanding gas, and it drops your Isp a little bit. So what you would like to do is make the nozzle all adiabatic, but we can't do that because of the materials.

This study is based upon a ROVER/NERVA core that makes use of a number of clusters, as shown in Figure 4. In this case, there are a total of 6 of these 19-hole fuel elements residing around a center element, resting on a core support block. The fuel element is approximately 52 inches long. The tie tube assembly is used to get the enthalpy to drive the turbine. This particular design has pneumatically driven actuators.

Now, let's take a look at what happens to a high expansion ratio nozzle if we try to design it to make use of the maximum amount of enthalpy. In Figure 5 we have assumed carbon/carbon composite as the material, and we have plotted the maximum wall temperatures, both inside and outside, as a function of area ratio .

For the design you will see in a minute, we have chosen to limit the expansion ratio at which we attached this adiabatic cooled nozzle to about 150; the wall temperature goes to about 2600 degrees R on the composite. We can be tempted perhaps to go to a smaller cooled expansion ratio, which would mean that we would be extracting less heat. We could help ourselves with Isp, but in doing so we are going to run into a problem in what we think the maximum temperature is that the uncooled noozle can handle.

Figure 6 shows the plot of some of the calculations that have been made on the heat load on this kind of a nozzle. In this particular case, we have assumed that the hydrogen comes in and flows two ways at an epsilon of about 6. A portion of the flow goes down to that 150 to 1 expansion ratio point, then back up, which gives rise to these two values of the wall temperatures shown here. Notice that the heat flux hits the maximum around the throat, and in this particular case we are talking about 40 BTU's per square inch per second. It then drops drastically down.

In the shuttle engine, we are able to withstand heat fluxes at the throat area of about 75 BTU's per square inch per second, so we could go a bit further on up in chamber pressure.

One way to get the thrust-to-weight ratio of your system up is to reduce the size of the entire assembly by going on up in chamber pressure. One of the big drivers in terms of size is the nozzle, so that if we do succeed in operating at a higher chamber pressure, that will shove up this heat flux at the throat. However, we still have some margin to deal with there.

Figure 7 shows what this thing looks like when you make the assumptions that I just talked about. Here is a nozzle assembly involving the reactor, the throat area, the point at which the hydrogen comes in and makes a pass, and a half-portion of it down through the throat. The other part goes up through the converging section where your big heat load is. Notice how big this whole assembly has gotten. I have shown it here as if it were an extendable nozzle (this is the uncooled portion), and it's translated up around the engine.

This particular size is dictated in part not only by what I was just talking about, but by the size of this interface. If you try to get a very high Isp with a conventional nozzle system (and the reason this thing looks this way is because we have tried to avoid mach lines; shock losses in that expansion process to minimize Isp again) you are talking about a very large assembly.

There may be ways to get this down to a more manageable size. One can think about the idea of the collapsible drinking cup you take on camping trips, and put several interfaces there to be able to pull down this size. There are also some other nozzle concepts that lend themselves to better packaging, but if you go conventional and you go for maximum Isp, this is what you are faced with.

I mentioned earlier that our interest was in having a reliable system so that if we should have a failure, or sense an incipient failure of one of the turbopumps, we could continue to operate and get the mission accomplished or at least retreat gracefully to some sort of a recovery plan. If you try to provide for dual turbopumps to do this, you may have a very complex system involving a number of valves that can isolate the pumping, or that can also get the pumps to share the load equally.

In Figure 8, which we have patterned after our experiences in Nevada, we had actual flow pumps that had negative HQ curves. That kind of a system is inherently self-balancing. The pumps share the load equally.

We have done it here by picking a design point that is close to the design specific speed line, but is far enough over from the predicted stall region so we can actually throttle at full temperature down to 50 percent. This is the value that I assumed in my example. It also represents the kind of limit that the reactor people are comfortable with in terms of having full temperature, but reduced coolant flow going through. If you suffered a failure of a turbopump, you would move to a new operating point out to the right, where the developed head and the flow rate intersect with the reactor load line.

This example shows an ability to meet that requirement, but notice that we are getting close to what was called a negative flow incidence. That's the case on the inducer at the front end of the pump where the flow incidence angles on the impeller, the front end of the impeller, goes negative, and then your NPSH requirements come up. So if you want to try to operate this pump at 70 percent thrust (for reasons of retrieval on your mission) and you want to operate with negative NPSH, it may be necessary to add a boost pump that would be hydraulically driven so you could match the speed to give you the proper incidence angle.

Figure 9 is a cartoon of one version of the pump that could do this. In this particular case we have patterned it after what we did on the Mark 25. The design incorporates hybrid hydrostatic bearings, at the outboard end of the turbine, which were proven in Nevada to be able to operate in a very satisfactory manner.

We wanted to go for the hydrostatic bearing to get rid of any materials that were in the bearings or anywhere else in this turbopump that would suffer any kind of damage from the intense nuclear environment that we anticipated. Those bearings actually provided the means of doing that, and they worked.

We also tested an advanced inducer which actually went up in flow capacity by 50 percent area. It was made out of titanium so that we could keep the weight overhang off the stub of the shaft reasonable, to maintain a critical speed where we wanted it. We also reduced the incidence angle, a design Q over N, to about 1.5 degrees, and we tested it. We found that we were able to ingest not only a saturated liquid going into the pump, but we kept going and we found that we could ingest up to 30 percent vapor and still the pump put out full pressure.

If that can be realized in a flight system, it enables us to pump a saturated fluid from a tank. The other thing we have shown is a single stage turbine over at the drive end. We could play the games of going to two stages there, take the pressure ratio and adjust it across each stage and make more effective utilization of the turbine drive fluid, depending upon the temperature that we are able to put into that drive fluid by the tie tube circuitry. So, we have some flexibility there.

We have also looked at integrated, pneumatic fluidics control systems to come up with a control system that would enable us to operate the entire engine in an intense radiation environment (see figure 10). Based upon the development work that was accomplished, it looks like we could do it. In other words, it would not be necessary to shield this engine from anything it does to itself radiation-wise. If you want to put a shield in this engine, it would be to protect the crew, but not because the engine requires it.

Figure 11 shows the final version of what an expander cycle engine system would look like in terms of its operating conditions. This particular setup allows us to meet the requirements I talked about, except that we are talking here about a weight of about 18,000 pounds and not 14,000 pounds.

How do we get that thrust-to-weight ratio up? Obviously most of the weight is in the reactor, but there may be a limit to what can be done there to make the weight as low as you would like to. There is another way to get that thrust-to-weight ratio up, and that is to get more thrust out of this configuration. And the thing that determines what you can get out of this engine is not the design of the pumps, not the design of the nozzle, but the power density in the fuel elements.

This particular design at 100 K has a power density of 1 megawatt per fuel element, which was actually demonstrated in the Phoebus program. There are some indications that you can get as high as 1 and a quarter megawatts per fuel element. That would raise the same engine to 125,000 pounds of thrust. That's the route that you need to examine: how hard you can push the fuel elements in power density for a given outlet temperature and a given total operating time?

TYPICAL MANNED MARS MISSION NTR PROPULSION REQUIREMENTS

• THRUST, NOMINAL	100,000 LBF
• PERFORMANCE	≥ 900 SEC
• MAXIMUM WEIGHT	≤ 14,000 LBS (WITHOUT SHIELD)
• FULL PERFORMANCE OPERATING RANGE	110% → 50%
• EMERGENCY THRUST (W ONE PUMP OUT)	70%
• NPSH (MIN)	ONE VELOCITY HEAD FROM SATURATED LIQUID IN TANK
• MAXIMUM OPERATING TIME	2 HOURS
• NUMBER OF RESTARTS	≥ 6
• TRANSITION, FLOW INITIATION TO FULL THRUST	30 SEC
• TRANSITION, 50% THRUST TO CUT-OFF	30 SEC
• MAXIMUM CORE TEMPERATURE, AFTER HEAT	1800°R

Figure 1

Performance* of NTR Operating with High Pressure Hydrogen

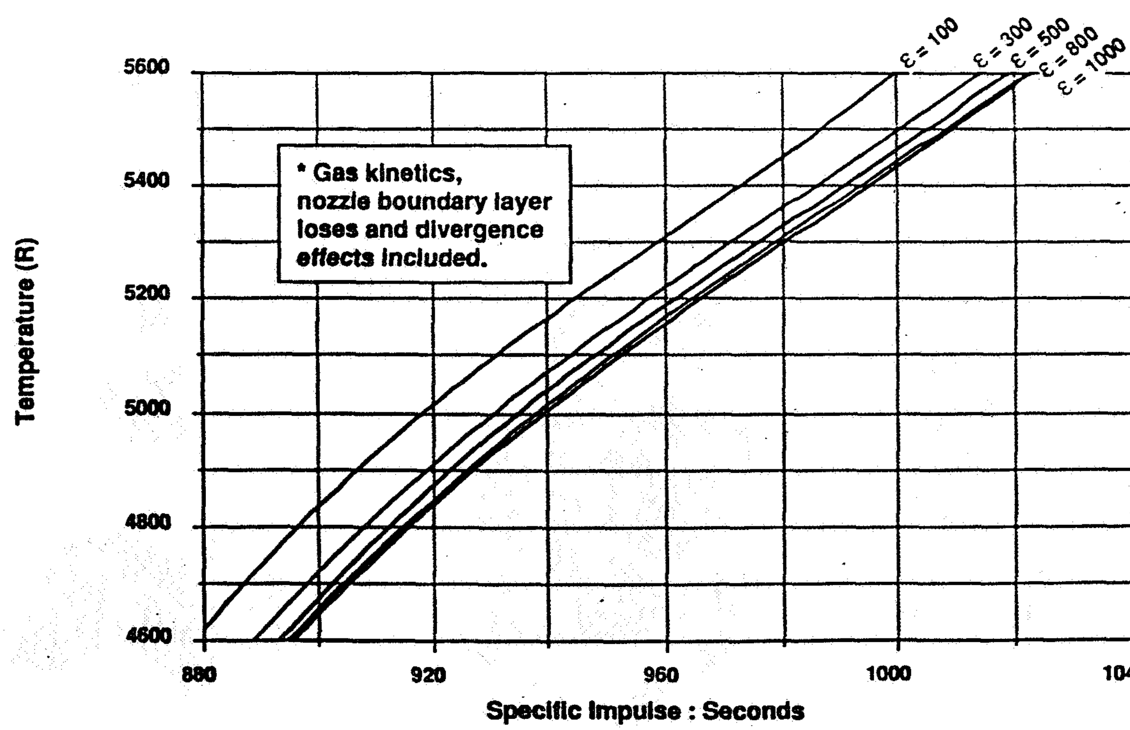


Figure 2

NTR Expander Cycles (Typical)

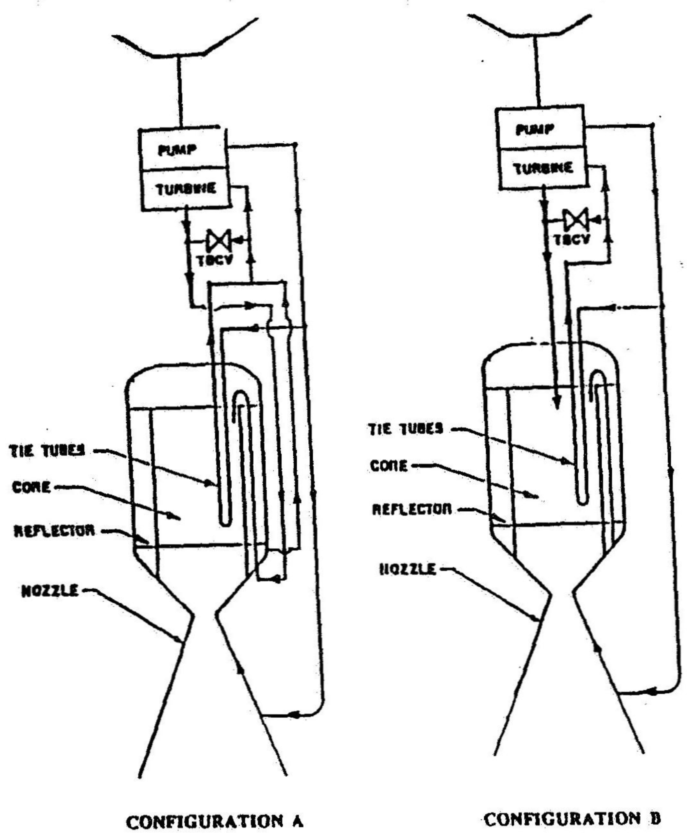


Figure 3

Rover/Nerva Core-Reactor

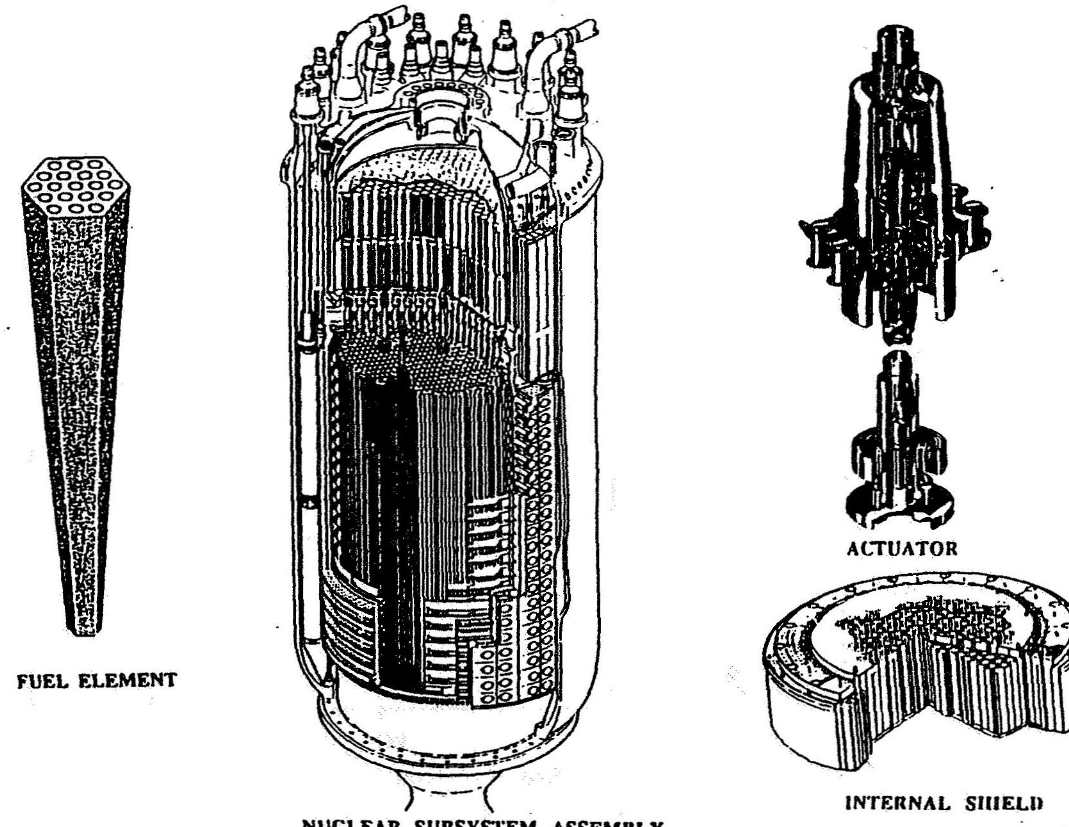


Figure 4

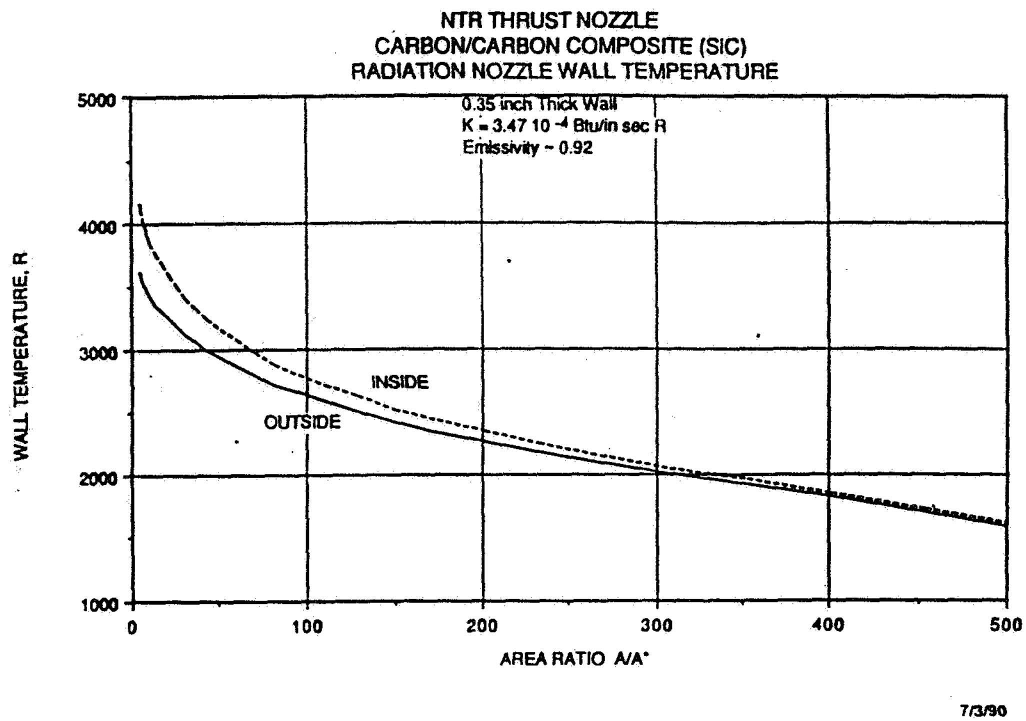


Figure 5

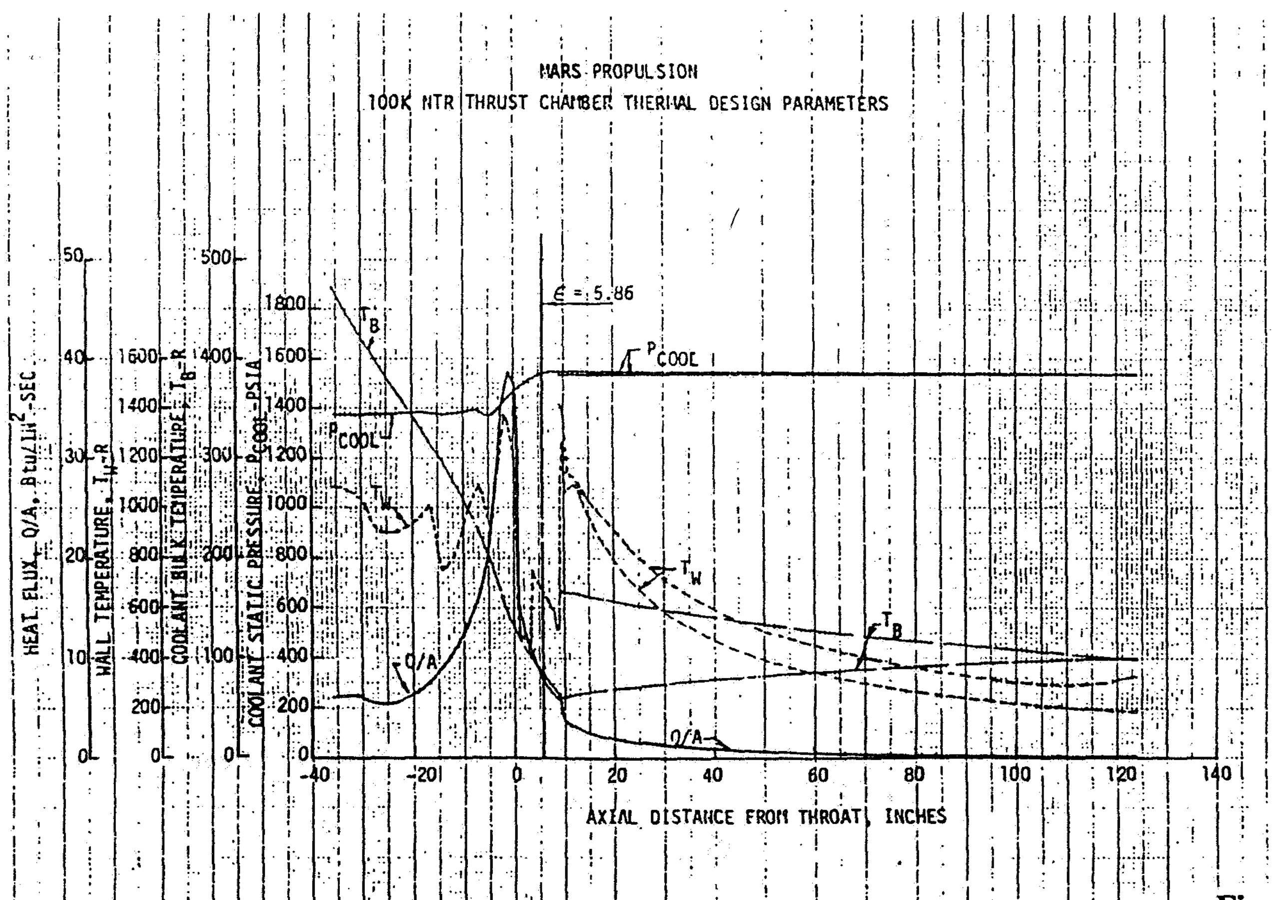
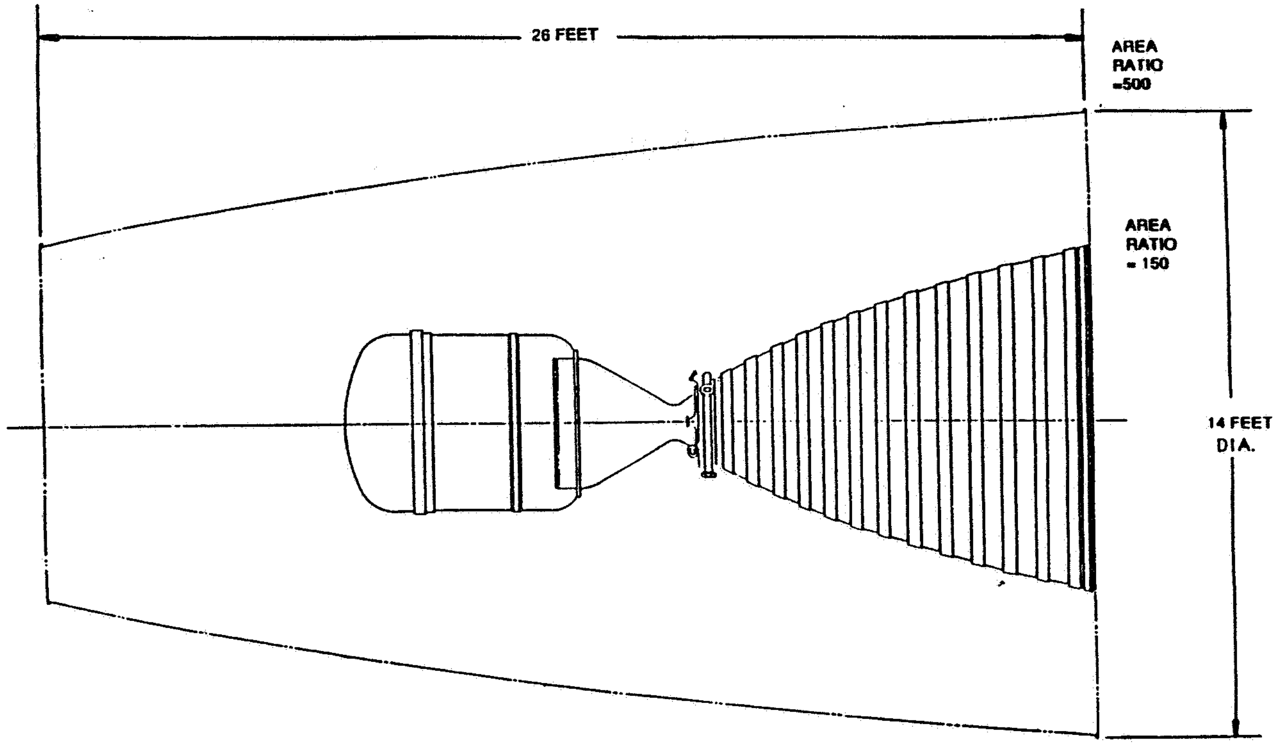


Figure 6



100K NTR NOZZLE ASSEMBLY

Figure 7

CANDIDATE DUAL PUMP OPERATING MAP

100K NTR ENGINE

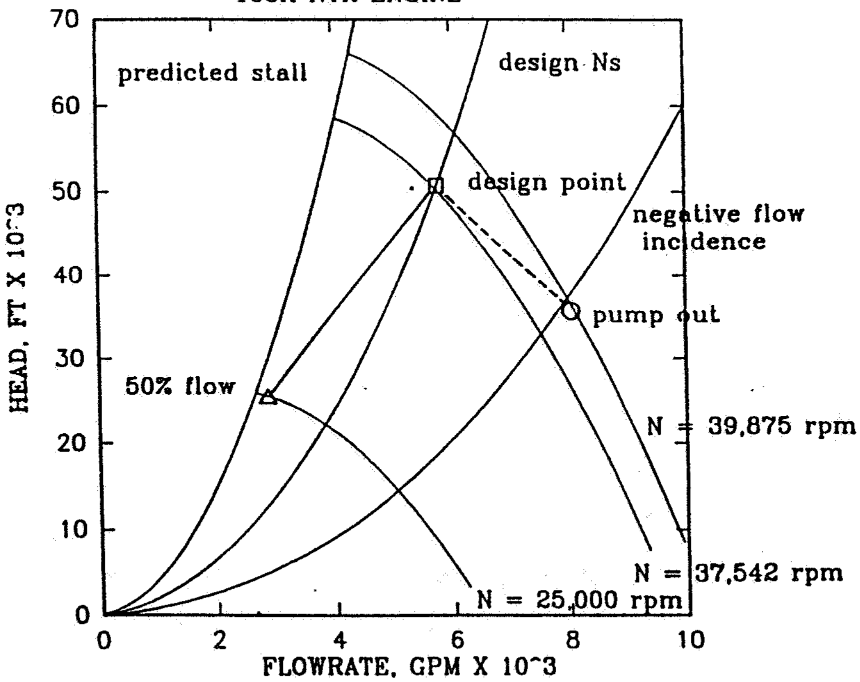
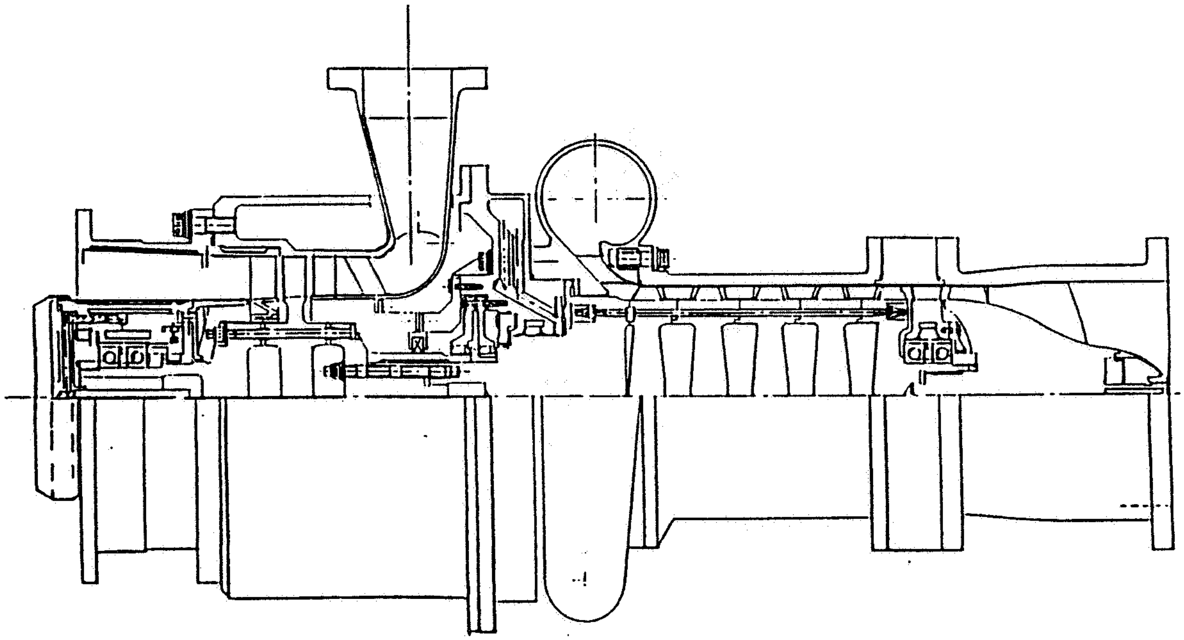


Figure 8



SCALED 5 STAGE MK25 PUMP
SINGLE STAGE EXPANDER TURBINE
PUMP FLOWRATE 54 LB/SEC
RPM 37,500
PUMP INLET DIAMETER 5.71 IN.

Figure 9

NTR Integrated Pneumatic-Fluidics Control System

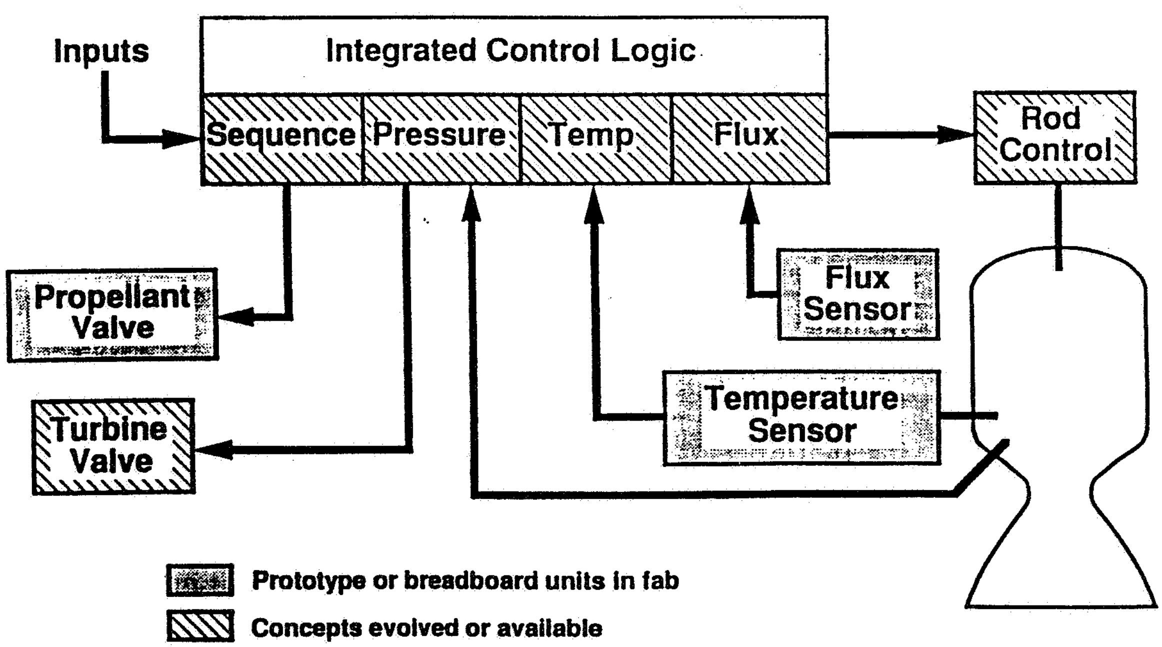
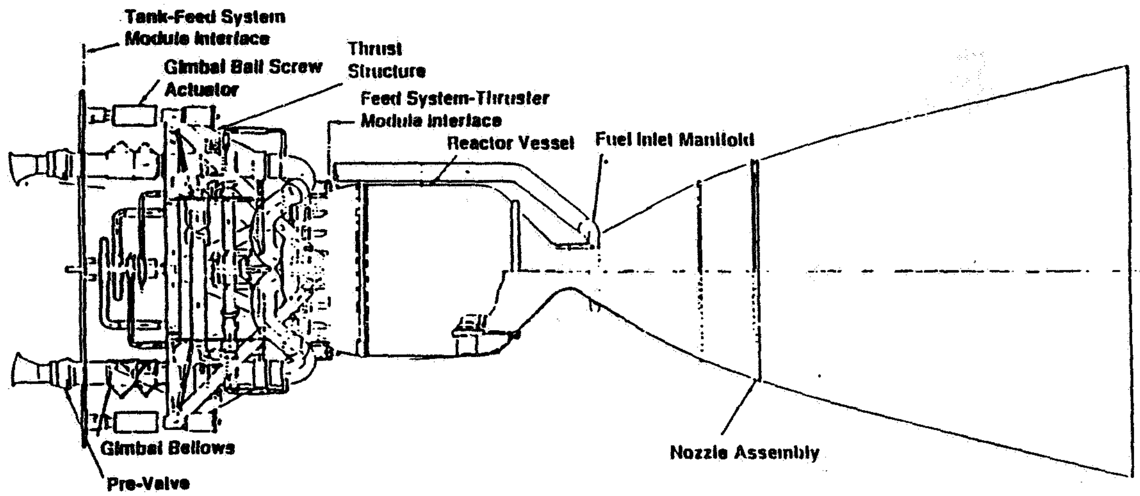


Figure 10

100K FLIGHT THRUST MODULE/FEED SYSTEM MODULE ASSEMBLY



Is (ϵ =500:1, UNCOOLED SKIRT)	922 SEC	ϵ (COOLED SKIRT)	150:1
WEIGHT	14,000 LB	POWER DENSITY	85 MW/FT^3
FLOW RATE	108 LB/SEC	ENGINE CYCLE	EXPANDER

Figure 11

N92-11093

UPGRADED NERVA SYSTEMS: ENABLER NUCLEAR SYSTEM

Gerry Farbman
Westinghouse

What is the "Enabler?" (see Figure 1). That's a term from people like myself who are incurable marketeers. We say the "Enabler." When Westinghouse talks about it, it's NERVA/ROVER, when Los Alamos talks about it, it's ROVER/NERVA.

The NERVA/ROVER "Enabler" technology enables things to be done. It enables you to go on a low risk, short-term program to meet the requirements of the Mars mission and maybe even some lunar missions.

To put things in perspective a little bit, Figure 2 shows a full-page ad back in early 1966 published in the New York Times, the Washington Post, the Los Angeles Times, the Wall Street Journal and the Pittsburgh Post Gazette. This was after we had tested the first nuclear engine at Jackass Flats, and the words say "Today Mars is closer." And I sure wish we were able to say we continued that effort and we are that much closer right now, but at least that's where we were at that time. Hopefully we can get to that kind of point again.

Our contention is that the NERVA technology, the Enabler, is a foundation for tomorrow's space missions (Figure 3).

The pictures we have here are fuel elements, the NRX/EST. Again, NRX/EST was the reactor that was tested and was the system that made Mars that much closer in February 1966, and it was real.

Figure 4 lists all the tests that were made during the program.

Figure 5 approaches the NERVA program from a little different perspective. It shows the overall program objectives and milestones, the progress made, and where we were when the program ended. The program started with a demo flight engine objective (which got changed partway through the program). It was changed to a technology program to demonstrate rated thrust for 20 minutes, and then 60 minutes, and then demonstrate operation of engines, restart, cool down, and mapping. Then, we were to develop an engineered flight system.

Westinghouse bid on the program back in early 1961. We were under contract in late 1961. The first test that we put together on an engine, (a complete engine as opposed to just reactors with nozzles or orifices at the aft end to give us a pressure drop), was the EST engine. That was in early 1966.

The reactor technology goals were met by 1968. The engine technology goals were met by 1969 when we tested the XE prime engine. The preliminary design report on the flight engine was completed and presented and approved, and then the program was terminated.

We had gone through all of these activities and had things left to do, such as the final reactor design, and the final engine design. What it turned out was just storage of technology data.

Although the program ended in that time period, we at Westinghouse continued working on the program as best we could. We kept the technology alive (at least we sure tried to), with a whole bunch of miscellaneous contracts, all of them small compared to the NERVA contract (see Figure 6). We tried to keep a cadre of people knowledgeable of the NERVA technology, using the NERVA technology, so that today the technology is available and ready to be used. It can provide a meaningful start to the revisit of a Mars mission.

We have talked about a $1.4 billion ROVER/NERVA program several times. I thought it might be interesting just to show how that was broken down (see Figure 7). The Los Alamos part, as best we can reconstruct the numbers was about $177 million. The Westinghouse Aerojet NERVA program was $660 million; technology, $328 million; operating costs at the test site, $90 million; and facilities at the test site, $153 million. So that's your $1.4 billion in "then year" dollars. I just added on what we have spent at Westinghouse, both other people's money, and our own money since then keeping the thing alive.

In Figure 8 we are talking about direct thermal propulsion. A couple of weeks ago at the NEP Workshop, we talked about a steady state electric power system using NERVA technology. Tomorrow, at the mission analysis panel, we will also talk about a dual power system where we can get direct thrust and electric power for whatever purpose you want, either propulsion or housekeeping. The same technology is available to be used in all of these kind of systems.

With the NDR engine, direct propulsion, we are trying to provide an optimum amount of energy to the turbo pump and the optimum temperature out of the reactor to get the optimum Isp. All these things are based on a 75,000 pound thrust engine because that was the requirement that had been established for this application.

We have looked at flow here in Figure 9. Figure 10 is a color picture of the NERVA nuclear subsystem, but I don't want to spend the time going through it. We have all seen this, and the model (NERVA model at the workshop) alludes to the kind of design we're talking about anyway.

Figure 11 shows the arrangement of fuel within the reactor. We are talking of fuel elements that are extruded composite matrix elements containing fuel within the structure. There are 19 coolant holes within each of the fuel elements, coated both on the external and internal surfaces with zirconium carbide to provide resistance to hot hydrogen attack.

These are then assembled around central elements which are support tubes, the tie tubes. The tie tubes have associated with them some zirconium hydride moderator to thermalize the neutron spectrum in the reactor and reduce the amount of uranium that's needed for criticality. We show here the materials and how the whole thing is put together. The tie tubes are reentry type tubes where the coolant flows down and then back up and out. That was shown on the schematic. This is an approach to show you how these things look.

For our reference case, we are talking about composite fuel, as shown in Figure 12. The vintage 1972 NERVA was a beaded fuel within a graphite substrate. We are talking about UC-zirconium dispersion within a graphite substrate, and this is what has been termed the "composite" fuel.

The shaded NDR column in all cases is what we have set as the reference case for the Enabler reactor system. Column one on the left describes the reactor that was run as XE prime. I put that in here because that had the technology readiness level of 6 by everybody's assessment. Composite fuel was developed late in the ROVER/NERVA program, but never fully tested to technology level equivalent to the fuel in the XE prime.

The NERVA '72 update incorporates today's requirements and could include some general improvements, like improved beads, and is the next step in a NERVA-type system. The composite is the Enabler target for now. From there, we can go to a different fuel material, a binary carbide, and get a temperature of 3100 K as chamber temperature and increase the Isp to 1020. Perhaps we can even go to ternary carbides, although the technology level on the ternary carbides is pretty low. But if we can get there, we can further increase our chamber temperature to 3300 K and get an Isp perhaps of about 1080 seconds.

So, there is room for improvement in the technology. We are not pushing things excessively. We are working on a system that had a reasonable amount of demonstration and testing in respect to fuel during the NERVA program. Composite fuel was run in the nuclear furnace, and it was run in electrical tests, and so we had a reasonable database.

What is the technology level? (Refer to Figure 13) Again, for argument sake, I assigned a 6 to everything on XE prime because that is conventional wisdom. Things kind of back off as you start adding new requirements and changes, but the things that are most significant are really in the fuel area, where we are now talking about composite fuel probably at a technology level today of somewhere in the order of 4 to 5. There has been testing done on it, but not enough to give you the good, comfortable feeling that you know all about the fuel.

If we go to a binary carbide, you have to back off a little bit more on technology level as assessed today. If we go to the ternary carbide, it's kind of like a semi-dream, not a full dream, because we know something about it but not enough to really assess what its capability is. The other things are generally all 5's and 6's.

We are adapting the SP-100 approach for putting additional control rods within the reactor core to meet some of the new safety requirements on multiple capability for shut down, positive shut down, and positive protection against launch accidents, immersion, and things of this sort. Therefore, we backed off on the technology level a little bit because there is more work to be done on that to be able to assess the adequacy of that design and the applicability to a propulsion system.

The key design parameters for all the systems are listed on Figure 14. For the composite NDR column, thrust is 75,000 pounds (not 75 pounds), engine availability at 2006, reactor power 1600 megawatts, and you can read the rest of the numbers. The engine thrust-to-weight without a shield is 4. And that's where we pegged it because that's where the baseline requirements said. I will present some curves to show where it can go if you change some parameters.

Adding in a nonoptimized shield, far from being optimum, the thrust to weight drops down to 2.3. We are talking about a specific impulse of 925 seconds. This is a thousand pound chamber pressure, 500 to 1 expansion ratio nozzle, and so forth.

Stan Borowski talked about core power density having an effect on thrust-to-weight ratio. Figure 15 shows that if we increase core power density we can go from a 4 to perhaps a 6 and a half. This results from shrinking the reactor as you get more and more power per fuel element. Of course there is some additional risk as you do that, but it's within the realm of possibility. For the purpose of this workshop we did not try to push the reactor, we tried to be reasonably conservative in the approach we used.

We also took a look at what the thrust-to-weight ratio would be as we changed the thrust level of the engine and reactor. On Figure 16 you can see that going from about 25 pounds of thrust up to 250,000 pounds of thrust, this is the kind of range you get for thrust-to-weight ratio. Again, these are representative numbers.

The reactor is not growing on a linear basis with increased power. Recall we are thermalizing the reactor quite a bit, so it's a basically thermal machine. You are just putting in some more flow area for the higher power requirements, but it's not growing linearly.

One is always concerned as to what kind of life you can get out of these reactors. Of course, that's a function of the kind of fuel you have and the temperatures of the fuel. The lower the temperature, the more life you will get out of it, the higher temperature obviously the shorter life. The curves on Figure 17 are really bands and not single lines. They ought to be thought of in terms of bands to give an indication of what you can do in fuel life as a function of temperature.

The lower curve represents the vintage NERVA type of design. The middle line represents the composite design, recognizing we are going to operate at about 2700 K nozzle chamber temperature, which says we ought to be able to get, without any strain at all, two hours of operation based on the data that was assembled during the NERVA program. More data ought to be assembled to see where the true limits are.

With carbide fuel, where we were hoping to operate at about 3100 K chamber temperature, we ought to get several hours worth of operation. Again, more data is required to pinpoint what the limits are and what the capabilities ultimately ought to be.

What are some of the key technical issues? (Refer to Figure 18) Fuel has to be one of them. We need more data on fuel. There was limited testing in the nuclear furnace. We have to do more testing. We have to demonstrate once more the effectiveness of the zirc-carbide coating, the so-called "super-coat" that in electrical test did last ten hours through some 64 cycles of temperature swings. We have to do it again, show that we can do it, and demonstrate the lifetime.

Safety. Somebody earlier today said safety has to be the byword, and that surely has to be addressed in anything we do. It is a key issue, not only a technical issue but it's a programmatic issue and an emotional issue and a public perception issue. Therefore, we call safety inherent, engineered-in. Public perception, and all of these things, have to be addressed, some from a technical viewpoint.

The issue of intact reentry, permanent shutdown and fuel integrity are some of the technical issues. The public perception issue is one that has to be addressed in a different fashion and doesn't get addressed really in a research and development program or demonstration program.

Critical tests and activities are listed in Figures 19-22. We have gone through what we think might be a first-year type of program in Figure 19. One of the key issues in the first year is to initiate design of the ground test facility. Whatever this ground test facility is going to be, it is on the critical path. And the sooner we can get started on that, the sooner we are ready for anything that comes along later on.

We have also looked at near term activities, including fuel elements tests and showing that we can meet the fuel reactor safety issues. (See Figure 20).

Far term tests include nuclear subsystem tests of all sorts (See Figure 21). And then further into the far term, there are engine tests to be done where you put the whole system together and run it through its mapping and performance characterization (Figure 22).

We then get to something that is very controversial, and that's how long does it take to do this? (Refer to Figure 23). Any number that I put up (any number that anybody puts up) for the schedule is obviously not the right answer, because we don't know what the right answer is. We don't know what the parameters of the problem are or the funding availability. So what we have done is said, okay, if we had to get to technology readiness level 6 and we were not constrained by funding but constrained by the time that it takes to do things --where a critical piece of the whole thing is the test facility -- how long would it take to get to technology level 6? And we think we can be there in eight years.

Will it take eight years? Undoubtedly it will take longer because the money is it not going to flow this way. What will it cost? Well, this one I guarantee is the wrong number (see Figure 24). But it is a number, and again it's based on saying, we are going to be success-oriented. We are going to do things quick, we are not going to stretch the program out. If you want to round that off to around $1 billion, I am willing to go from $755 million to $1 billion and say it's the same number.

But it's an order of magnitude for a program that is an eight-year program and not a program that, as I fear will happen with the way government funding tends to go, be a lot longer program as costs obviously go up when programs stretch out.

There are two sets of facilities that one needs (see Figure 25). One is the major facility for full-scale, ground testing of the engine. The other facility that is needed is for fuel testing, and here there are several options available to us: the ATR (Advanced Test Reactor at INEL), and also some Soviet test reactors where they are very anxious to test fuel, U.S. space reactors within their currently available and operating reactor systems. It's an option that might be considered. Figure 26 is a different version of the same sketch that Stan showed. I won't go into that.

And again, as the unrepentant marketeer, I have my final vugraph. Figure 27 lists all the goodies that come with this kind of system: it's technology-based, demonstrated under demanding ground test conditions. We went through a whole series of ground tests in the 1960's and early 1970's, and it worked; a wide range of thrust capabilities; no need for technical breakthroughs (we are talking about evolutionary changes, evolutionary changes to get to the composite fuel, evolutionary changes to get us beyond that); there is a technology synergism between the direct thermal thrust and other uses in space of the same kind of technology.

We think we have identified solutions to all the safety concerns, the technical safety concerns. The public perception concerns I back off on. There are modest development needs. Modest is in the view of the beholder. Your idea of modest may be different than my idea of modest. And as I said at the beginning, it's an Enabler for near term, low risk, low cost power systems. At least that's our position.

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The Enabler
(Based On Proven NERVA Technology)

G. H. Farbman/B. L. Pierce
Westinghouse Electric Corporation

Presented at the
NASA/DOE/DOD
Nuclear Thermal Propulsion Workshop
July 10 – 12, 1990

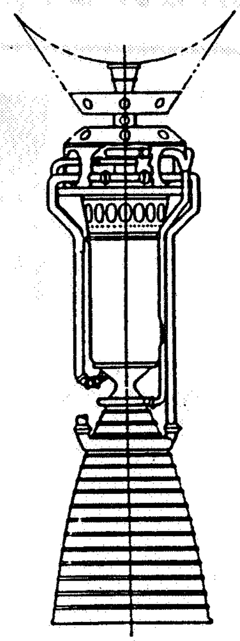


Figure 1

As Appeared In:

- *New York Times*
- *Washington Post*
- *Los Angeles Times*
- *Wall Street Journal*
- *Pittsburgh Post Gazette*

Figure 2

NERVA Technology – The Foundation for Tommorrow's Space Missions

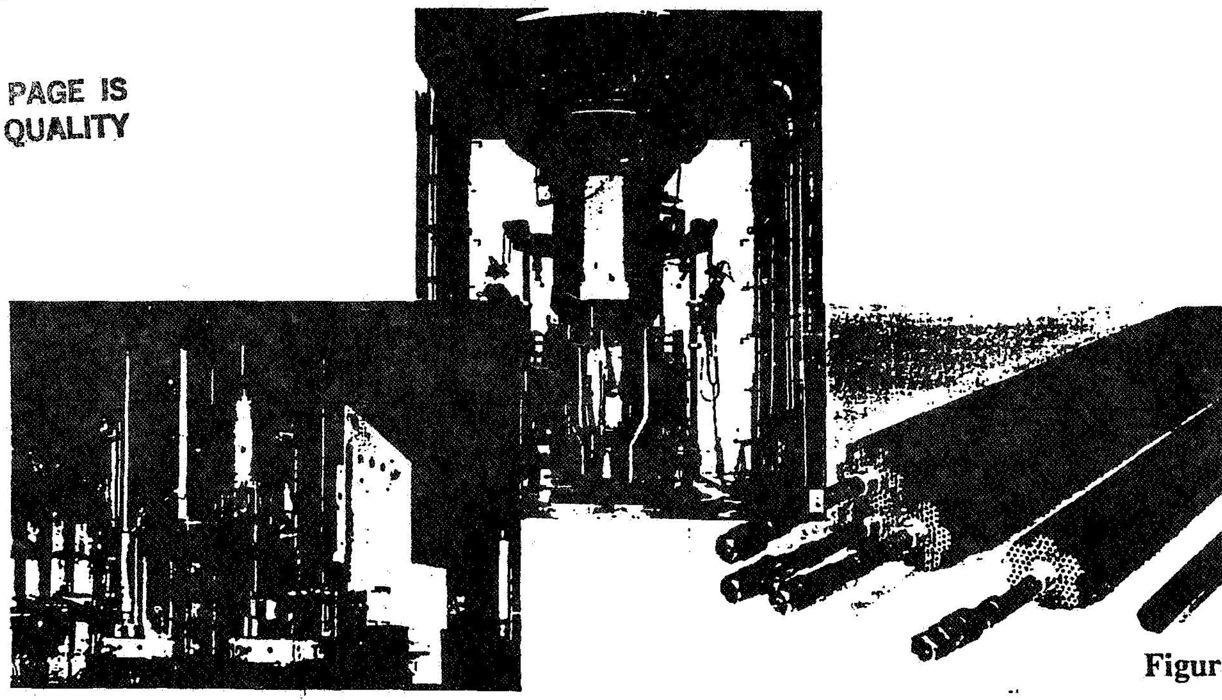


Figure 3

NERVA/Rover Reactor System Test Sequence

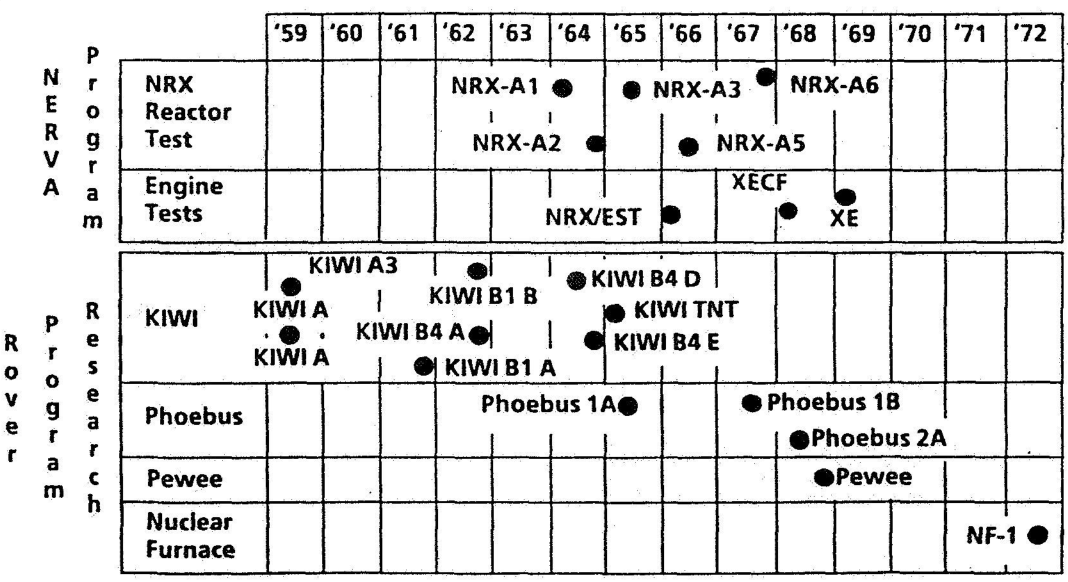


Figure 4

NERVA Engine Development Program

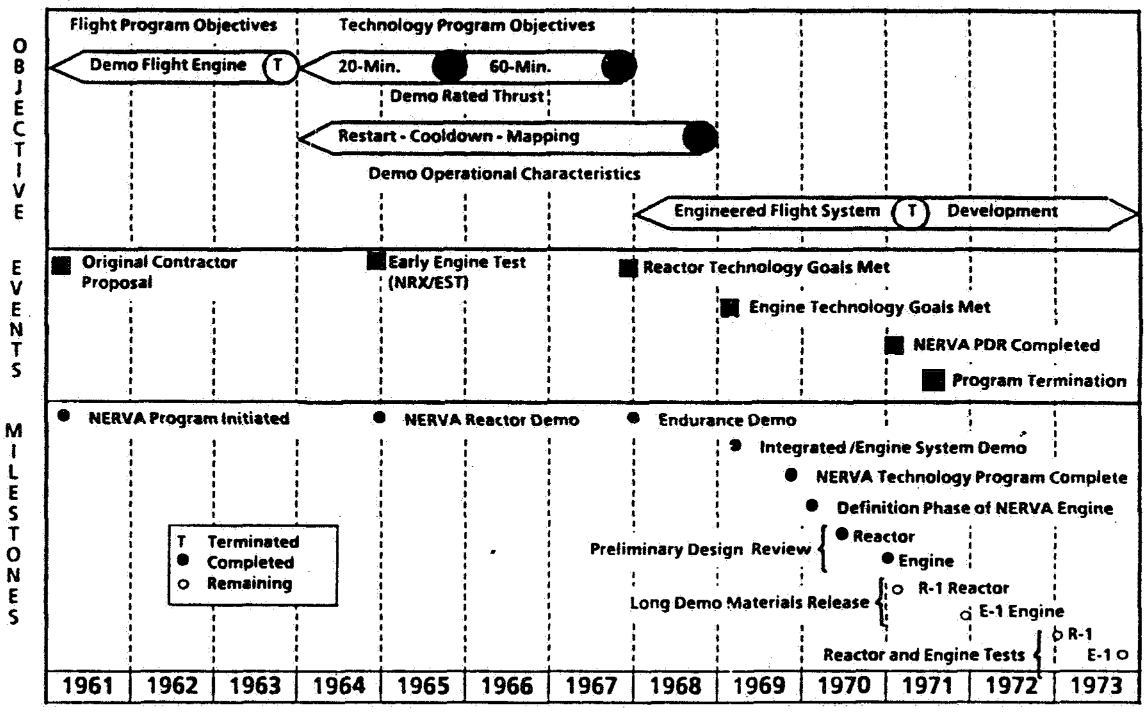


Figure 5

Nuclear Thermal Reactor Capability Based On Many Related Westinghouse Technology Programs

Program	Customer	Time Period
NERVA	NASA/DOE	1961–72
Nuclear Rocket Shielding Methods	NASA/LeRC	1969–70
Afterheat Distribution	NASA/LeRC	1970–71
Reactor Study for Nuclear Powered Aircraft	USAF	1970–72
In-Space Maintenance Concepts	NASA/MSFC	1971–72
Terrestrial Applications of NERVA Technology (Marine, transportable, etc.)	IR&D	1972–82
Nuclear Waste Disposal in Space	NASA/LeRC	1972–73
The Very High Temperature Reactor for Process Heat	DOE	1974–76
Closed Cycle Brayton Gas–Cooled System Feasibility Study	ONR	1976–79
Nuclear Bi–Brayton System for Aircraft Propulsion	USAF	1978
Space Reactor Evaluation	DOE	1983
Space Applications of NERVA Technology	IR&D	1983–Present
Integrated Spacecraft Total Energy System Analysis	AFWAL/TRW	1983–85
Gas Cooled Reactors for Advanced Terrestrial Applications	IR&D	1985
SDI Architecture Study	SDIO/SAIC	1985–86
NDR/MHD (Linear Channel) Study	INEL	1986
NDR/Brayton MMWe Space Power System	DOE	1986–87
Space Power Architecture Study	AFSTC/TRW	1986–87
NDR Nuclear Space Propulsion	AFAL/INEL	1986–88
NDR/MHD (Disk Generator) MMWe Study	DOE	1987–88
MSNPS	DOE	1988–Present
Manned Lunar/Mars Mission Propulsion System Studies	SAIC/NASA-LeRC	1989–Present

Figure 6

2379-21

Rover/NERVA Technology Represents A Significant Investment

		Total
• Rover/NERVA program (1955–1972)		$1,400 million
– KIWI	– $177 M (LANL)	
– NERVA	– $662 M (W/Aerojet)	
– Technology	– 328 M (technology)	
– NRDS	– 90 M (operating)	
– Facilities	– 153 M (capital/test facilities)	
• W Post–NERVA technology programs (1972–present)		$15 million

Figure 7

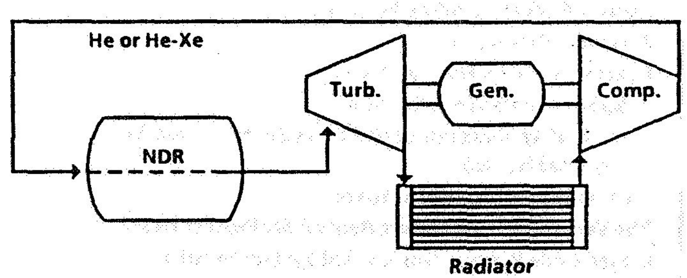
NERVA Technology Has Synergistic Applications

Steady-State Power
- 10's of MWe for electric propulsion

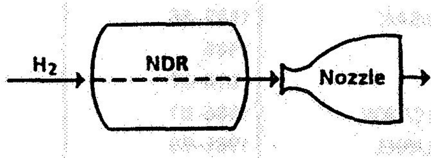
Direct thermal propulsion
- 15,000 to 250,000 pounds of thrust

Dual Power Systems
- High direct thrust (e.g., 75,000 pounds) plus low electric propulsion (e.g., 1MWe)

Steady-State Power



Direct Thermal Propulsion



Dual Power System

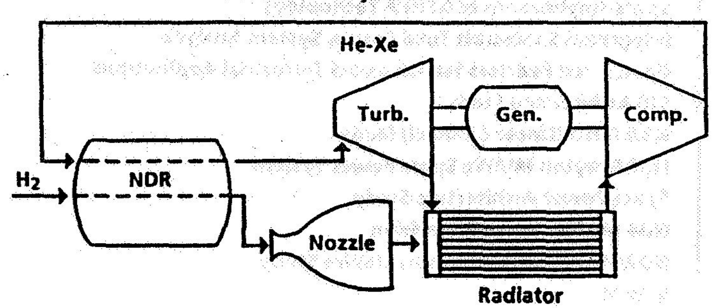


Figure 8

Flow Schematic of the NDR Engine

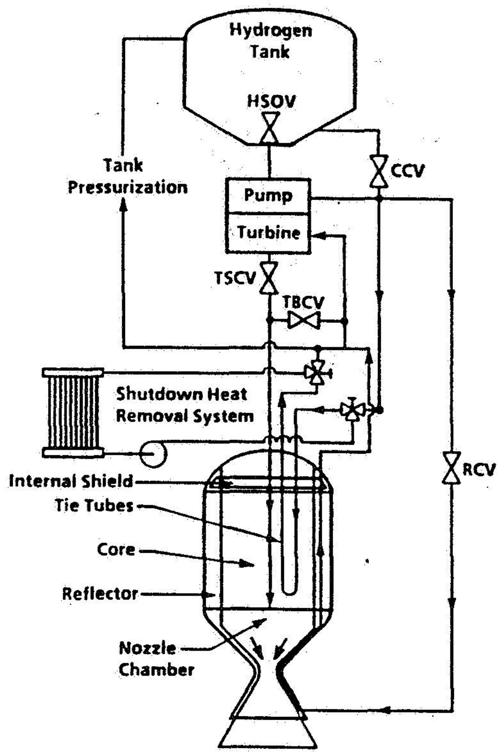


Figure 9

The NERVA Nuclear Subsystem

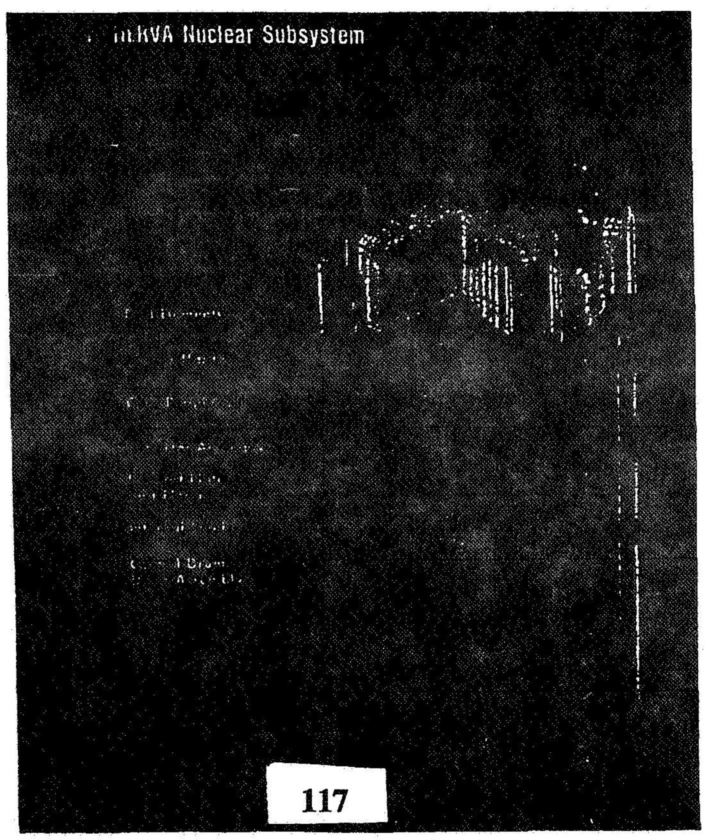


Figure 10

Proven NERVA/Rover Reactors

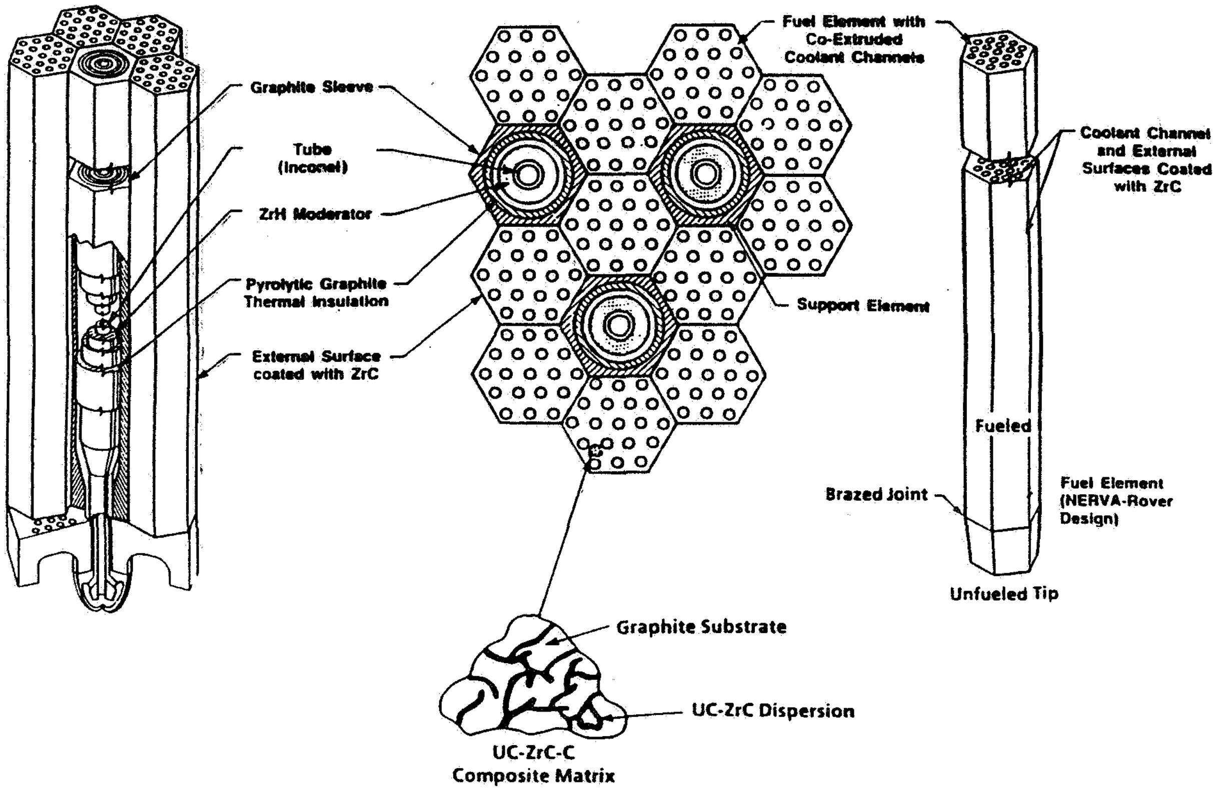


Figure 11

TECHNOLOGY EVOLUTION

	NRX XE'	NERVA 72 UPDATE	COMPOSITE	BINARY CARBIDE	TERNARY CARBIDE
• Fuel	Duplex Bead	Duplex Bead/ZrC Coated	UC-ZrC-C	ZrC-UC	UC-ZrC-NbC
• Moderator	Graphite	$Zr\ H_x/C$	$Zr\ H_x/C$	$Zr\ H_x/C$	$Zr\ H_x/C$
• Reactivity Control	Drums Plus Poision Wires	Drums Plus Safety Rods	Drums Plus Safety Rods	Drums Plus Safety Rods	Drums Plus Safety Rods
• Axial Support	Tie Rod	Tie Tubes	Tie Tubes	Tie Tubes	Tie Tubes
• Control Drives	Pneumatic	Electric	Electric	Electric	Electric
• Reactor Vessel	Aluminum	High Strength Steel	High Strength Steel	High Strength Steel	High Strength Steel
• Chamber Temp (K)	2270	2500	2700	3100	3300
• Pressure (psia)	450	1000	1000	1000	1000
• Nozzle Exp. Ratio	100:1	500:1	500:1	500:1	500:1
• Isp (sec)	710	890	925	1020	1080

NDR

Figure 12

STATUS OF TECHNOLOGY LEVEL

Nuclear Subsystem:	NRX XE'	NERVA 72 UPDATE	COMPOSITE	BINARY CARBIDE	TERNARY CARBIDE
• Fuel	6	4-5	4-5	3-4	2
• Moderator	6	5	5	5	5
• Fuel Element	6	5	5	3-4	2
• Axial Supports	6	5	5	5	5
• Lateral Supports	6	6	6	6	6
• Core Periphery	6	6	5	5	5
• Reflector	6	5-6	5-6	5-6	5-6
• Control Drums/ Drives	6	5-6	5-6	5-6	6
• Internal Dome Shield	6	6	6	6	6
• Core Support Plate	6	5-6	5-6	5-6	5-6
• Safety Rods	–	4	4	4	4

NDR

Figure 13

Key Design Parameters

	NRX XE'	NERVA 72 UPDATE	COMPOSITE	BINARY CARBIDE	TERNARY CARBIDE
• Thrust (lb)	55	75	75	75	75
• Engine Availability (yr)	–	–	2006	–	–
• Reactor Power (MWt)	1120	1520	1613	1787	1877
• Engine Thrust/ Weight (w/o Shield)	3.9	4.2	4.0	3.7	
• Engine Thrust/ Weight (w/Shield) (lbf/lbm)	–	2.4	2.3	2.2	–
• Specific Impulse (s)	710	890	925	1020	1080
• Mass (kg)					
Reactor	3159	5476	5853	6579	–
Non-nuclear	3225	2559	2559	2624	–
Internal Shield	1316	1524	1517	1517	–
External Shield	–	4537	4674	4967	–

NDR

Figure 14

Thrust To Weight Ratio Dependant On Core Power Density

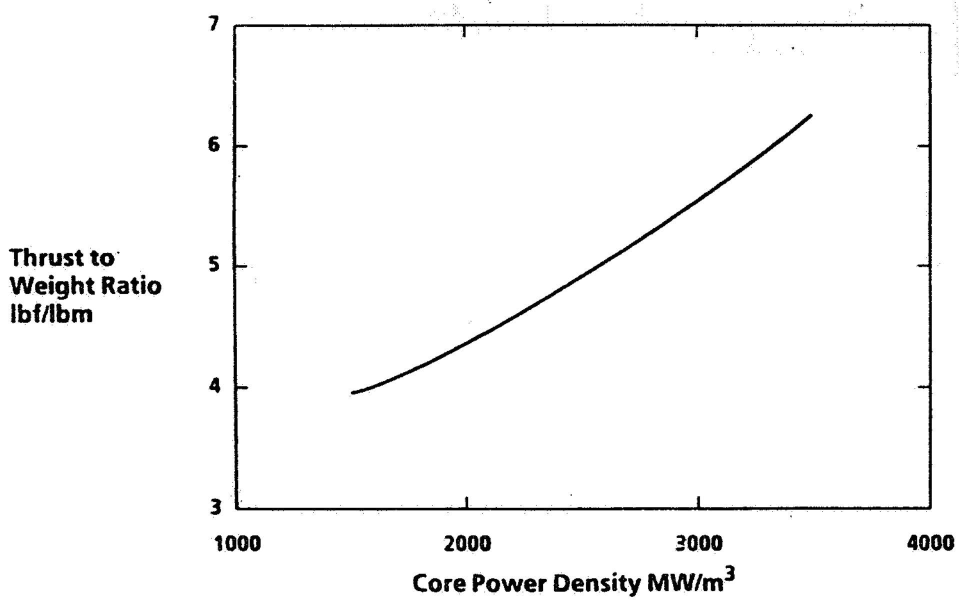


Figure 15

Growth Capability Of The NDR Engine

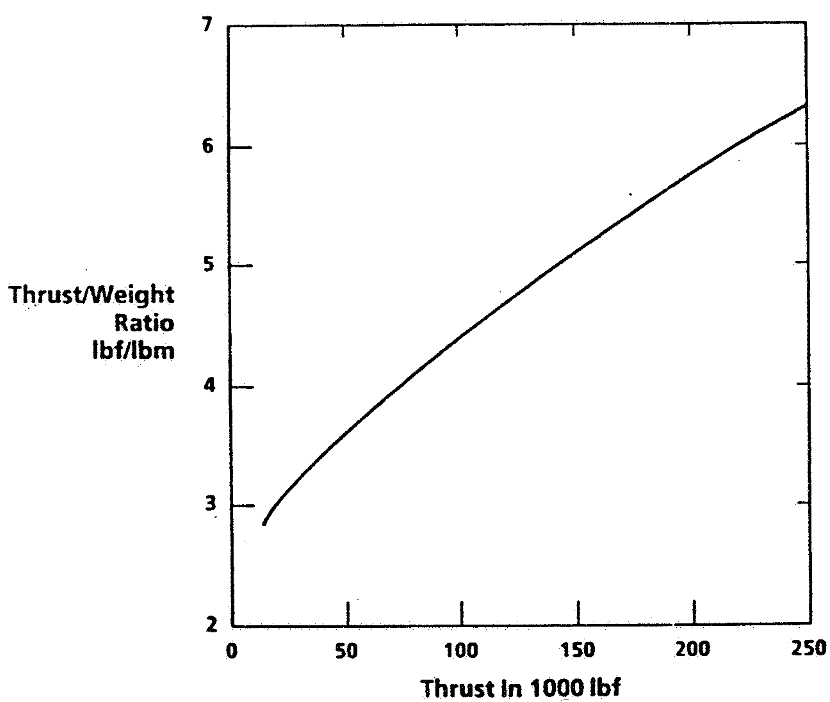


Figure 16

Effect Of Nozzle Chamber Temperature And Fuel Form

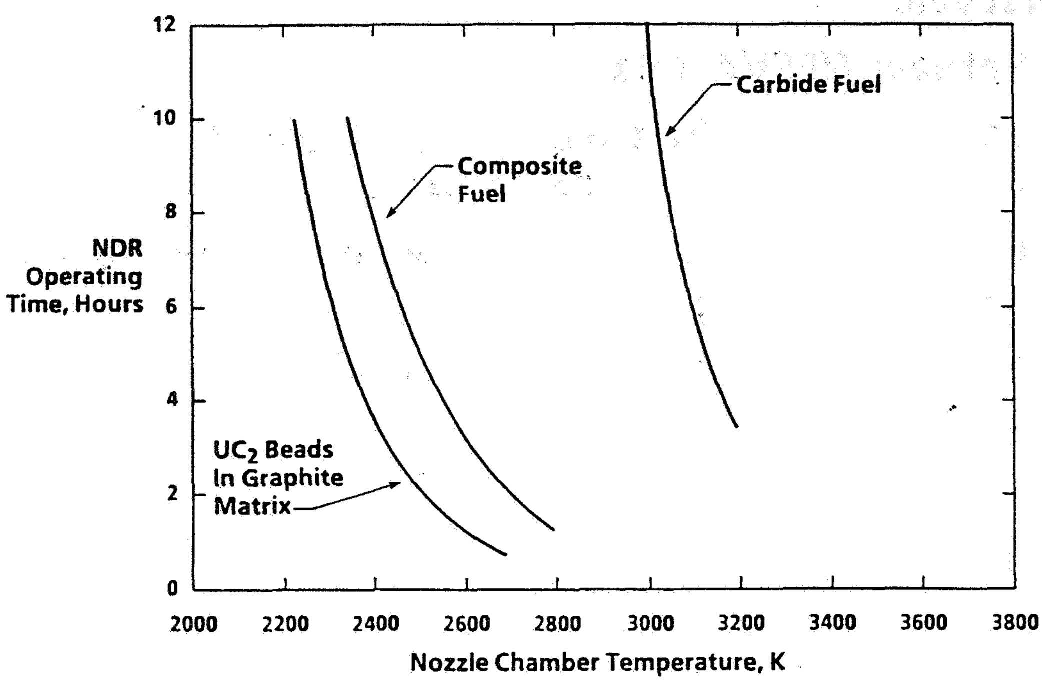


Figure 17

Key Technical Issues

- UC-ZrC-C composite fuel
  - Limited testing in nuclear furnace near end of Rover/NERVA program
  - Demonstrate effectiveness of ZrC coating ("Super Coat")
  - Demonstrate lifetime
- Safety: Inherent, engineered and public perception
  - Intact reentry
  - Permanent shutdown - applicable experience with comparable system operations (SP-100)
  - Fuel integrity

Figure 18

Critical Tests/Activities

- First year
  - Retrieve NERVA data
  - Review for required and/or desirable updates of drawings, specifications and procedures
  - Identify required analytical models and update or revise for current computer use
  - Initiate design of ground test facility

Figure 19

Critical Tests/Activities

- Near term (Phase I)
  - Fuel element test: demonstrate the capability of the composite fuel elements to meet current performance requirements
    - Demonstrate effectiveness of ZrC coating
    - Demonstrate fuel integrity/lifetime
  - Demonstration of complete fabrication of fuel element
    - Extrude fuel elements
    - Tests to assure quality of extrusion
    - Conduct in-pile tests
  - Reactor safety issues:
    - Subcriticality issues under full core immersion and core compaction
    - Approach to ensure intact reentry depends on design and materials for reactor vessel and internals
    - Demonstrate reactor shutdown and final shutdown capability in critical tests (drums and safety rods)

Figure 20

Critical Tests/Activities

- Far term (Phase II)
  - Nuclear subsystem tests
    - Control train tests
    - Test system controls and prototypic flight system control tests
    - Shielding tests
    - Feature tests - support structures, etc.
    - Safety tests
    - Etc.

Figure 21

Critical Tests/Activities

- Far term (Phase III)
  - Engine tests
    - Demonstrate operating envelopes
    - Perform cold flow experiments
    - Demonstrate startups/shutdowns/cooldowns/ emergency responses
    - Verify endurance/cyclic performance capability/ component interactions
    - Verify post test component conditions

Figure 22

Schedule For Ground Test Of Nuclear Thermal Rocket

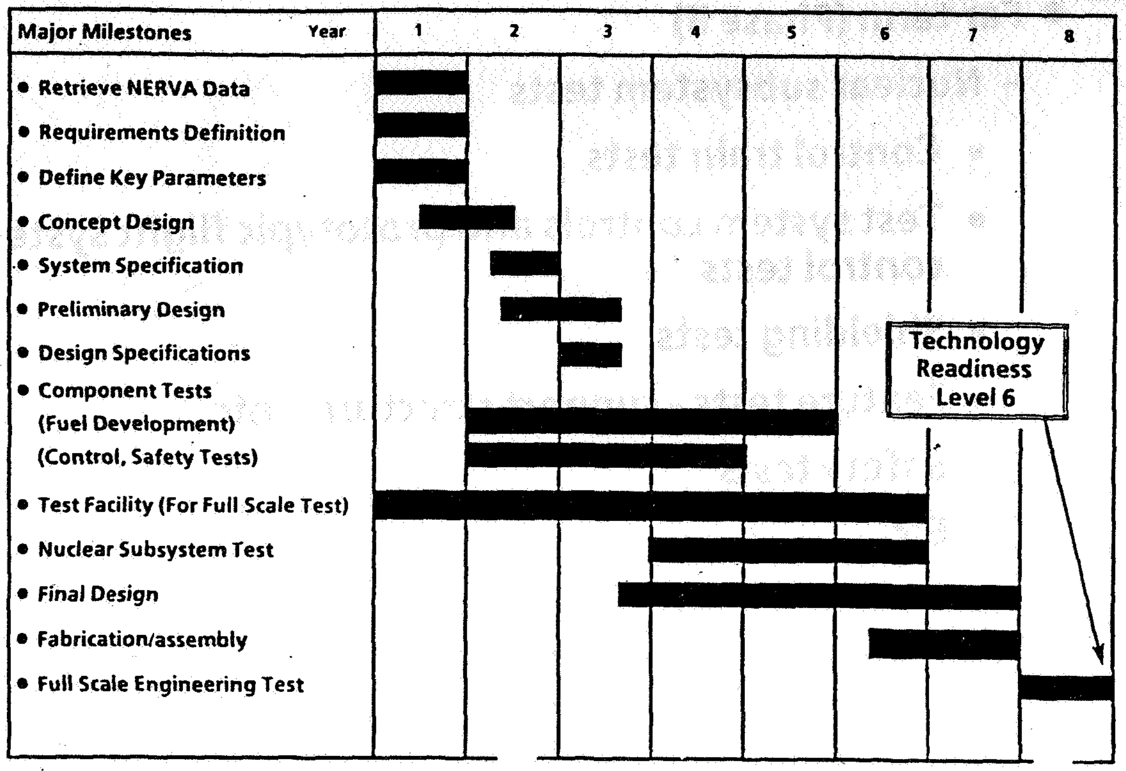


Figure 23

Development Costs For NTP/NDR

	($ M)
Reactor development and design	350
Engine development and design	150
Procure and assembly for full scale test	100
Facility preparation	125
Test costs	30
Total	755

Figure 24

Facility Requirements

- Options for integrated tests:
  - Exhaust hydrogen to a cleanup/scrubber system
  - Exhaust hydrogen into an underground tunnel
  - Test in space
- Can use existing containment facilities with modification for the hydrogen cleanup system

Figure 25

Nuclear Test Facility Option – Hydrogen Cleanup System Concept

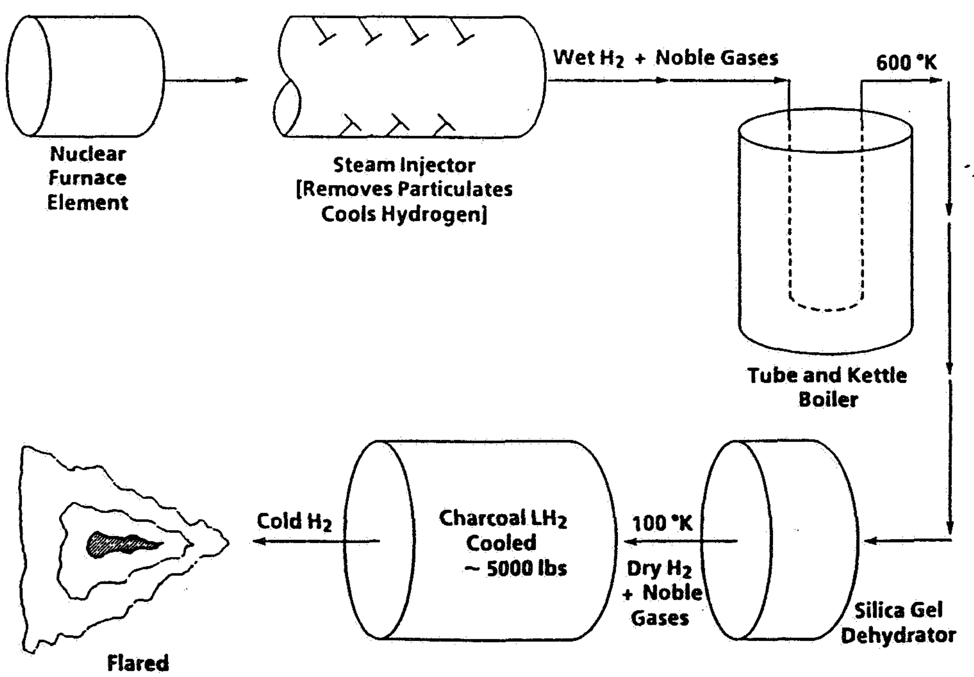


Figure source: LANL

Figure 26

Rover/NERVA Technology Provides Closed Cycle Power Systems With:

- A technology base demonstrated under demanding ground test conditions
- A wide range of thrust capabilities
- No need for technical "breakthroughs"
- Technology synergism between electric and direct thermal thrust propulsion systems for overall program economies
- Identified solutions to safety concerns
- Modest development needs
- An ENABLER for near-term, low-risk, low-cost power systems for nuclear thermal rocket applications

Figure 27

N92-11094

LOW PRESSURE NUCLEAR THERMAL ROCKET CONCEPT (LPNTR)

J. H. Ramsthaler
Idaho National Engineering Laboratory

I am going to talk about the low pressure nuclear thermal concept. The concept initiator is Carl Leyse from INEL.

First, I will give you a little background and a description of the system. Then, I will discuss performance, mission analysis, development, critical issues, and some conclusions.

The low pressure nuclear thermal rocket has a number of inherent advantages in critical NTR requirements (see Figure 1). First of all, performance-wise, it looks as though we can get into the order of 1050 to 1350 seconds for specific impulse, and we think we can get up to six to one thrust-to-weight. Reliability is a difficult thing to project. If you had enough money you could test everything and make it reliable, but when you are starting, if you can eliminate some of your troublesome components, you have a better chance of getting there. And that's what we have done in our design concept. With safety, you also have the same issue. We took a look at some of the safety critical failures and saw how we stand relative to them. Have we gotten rid of the initiators for these? I think you will find the answer is yes. For versatility, we have gone to a multiple engine concept. We believe that one of the major requirements is a "two-engine-out" capability. We have met that with the concept we are going to propose. We are at a NASA technology readiness level of two. I think that "concept verification" is required.

We have done some trade studies at INEL on what a nuclear thermal rocket concept should be. The reason is that I am an old "Nervite." I have believed in it since the 1960s. In 1986, the Air Force gave me the opportunity to go back and study it again. Since I knew NERVA, we picked it out as our concept and I got results very similar to what Stan said. We showed about a 20 percent cost advantage in everything we did. However, the reaction throughout the contry was "20 percent isn't enough." So we started looking at how could we build a better mouse trap. We went through a series of trade studies. About the only ground rule we had was that we believed the solid core reactor was going to be the first one we developed. So we limited ourselves to the solid core reactors.

We set safety as our primary requirement. This meant eliminating inherently unsafe design features if possible (see Figure 2).

For performance, temperature is the name of the game. We want to be able to operate at as high a temperature as we can. We want favorable neutronics for the highest temperature fuel.

There are some that go beyond the zirconium such as Tantalum or Hafnium. They are lousy neutronically, but if we can get the neutronics correct, we can operate at higher temperatures. We tried to do that. If you look into where you lose a lot of your Isp in these things, it is the balance between flow and temperature. You are going to be limited by your maximum fuel temperature. If you come up with a concept where you can balance this nicely, you are going to gain a lot of Isp.

We looked into low pressure because, when we get up to 3,000 Kelvin, you get significant dissociation of hydrogen. There may be a real performance advantage when you get into that area.

In weight, you have heard a number of people say you should get at least six-to-one thrust-to-weight. So, we set that as a requirement. Reliability, at this stage of the game, boils down to simplicity. I will show you that we have a fairly simple concept.

With that we came up with our reference low pressure thermal reactor (see Figure 3). The concept was designed to maximize flow at low pressure and high temperature. In order to do that we came up with a radial outflow core.

If you look at NERVA and other concepts at low pressure, you reach critical flow at the exit of the core. In order to get a lot of core exit flow area, we went to the radial outflow. We have almost 50 percent flow area at the exit of the core. We can use virtually any kind of fuel that comes out of the fuel development program. We can use particles, plates, or whatever proves best.

An important feature would be that we can operate on tank pressure. We do not need a turbopump. We think we can operate with reactivity power control and eliminate the control drums.

The reference engine is an 11,000-pound thrust engine that weighs 1,840 pounds, (about a six-to-one thrust-to-weight). We are estimating a minimum specific impulse of 1050 seconds, with up to 1210 at full thrust. Then, low Isp is with a minimum of recombination, and the high Isp is with a maximum of recombination.

One of the unique features of the low pressure engine is that as you continue to drop pressure, you continue to get more dissociation, which increases Isp. We decided that if we took it as a good demanding objective, maybe we could get down to 20 percent full thrust. If you can do that, you get to a theoretical 1,350 seconds specific impulse.

The thrust level is too low for Earth escape, but it is useful for other manuevers. So we propose a dual function capability with one engine.

If you take a look at this particular concept, the main structural part is a large central area that is probably steel or some neutronically favorable material. It is surrounded by two berylium structures with a series of holes in them. The flow enters the nozzle and cools the nozzle and pressure vessel. It enters the center cavity of the reactor and blows radially outward, through the fuel modules.

Now, you notice we have a very short nozzle exit cone. One of the advantages of low pressure is that the heat flux is greatly reduced. As a matter of fact, it is about a factor of 50-to-one less than the high pressure NERVA engine. Thus, you can have a very short exit cone and lose very little heat going out the nozzle.

We flow around this way: we come in to the center of the core and then exit through our fuel elements (see Figure 4). Reactivity control comes by running hydrogen down into the center. We have a large center cavity, and fill it with hydrogen for reactivity control.

I might mention that NERVA demonstrated that you could operate with reactivity control fixed. The drums were fixed and could run a complete startup, full power hold, and complete shutdown on reactivity feedback (no control drum movement).

NERVA also demonstrated that with your control drums full-in, you can get enough reactivity in to go critical, despite the fact that you had the control drums in. Therefore, we think it is a very desirable option to eliminate them. If you look at the safety analysis report, almost all of it was addressed to what you do about control drum roll out and all the associated problems.

Our fuel bed assembly is very similar to the particle bed that Brookhaven has been proposing (see Figure 5). Cold hydrogen comes in, flows through the core structure, and flows through a fuel bed. In this concept you have particle fuel, a hot frit, and a cold frit. You also have a reflector area beyond the fuel bed. You can substitute fuel plates for the particles. We don't operate at a high power density. We plan to operate at 3-4 MW/L and the plates would have sufficient heat transfer surface.

The fuels that people are considering, carbides in particular, are ceramics. At the time of the NERVA program, there were many problems fabricating fuel forms. If there is one thing we have learned a lot about since the days of the NERVA program, it's how to fabricate ceramics. So, I think there is a good possibility that we can come up with some rather novel fuel forms with new fabrication technologies. I would even propose that we have carbide-carbide composites. I would propose a carbide-carbide composite might be a very viable way to make plates. The concept can use plates or particles or whatever type of a fuel form you come up with.

At the end of the NERVA program, we are projecting the capability to operate at 3,200 Kelvin. They were planning on doing that with zirconium carbide or uranium carbide composites. I suggest that you look at tantalum carbides that have approximately a 600 to 700 degree advantage over zirconium. There are also ternaries that may be able to operate at higher temperatures (see Figures 6 & 7).

In other words, if you pull out one of the old data points, there are some hafnium tantalum carbides that are higher than the tantalum carbide by itself. If you use melting point as a figure of merit and assume the structural properties will go with it, you have the potential to operate greater than 3,600 Kelvin, if you can design it to handle the unfavorable neutronic properties of the tantalum carbides and the hafnium carbides.

Figure 8 shows that once you get up to the higher temperatures, there is performance advantage for operating at low pressure. The capability to operate at 3000 K did not exist when NERVA was being developed and there was no reason to consider operating at low pressure.

But with this capability to operate at higher temperature, you begin to show the possibility for substantial improvements in performance if you can operate at low pressure.

First, we have done a preliminary neutronic study (see Figure 9). This particular one was done on a reactor OD of 1.2 meters. It's a little bit larger than our reference, with a core OD of one meter and 50 percent exhaust flow area. The basic flow is through the fuel element as shown on the right.

We have a zirc hydride sleeve on the outside; a very small one (one millimeter) to improve our moderation. We had a cold section (but actually it's not that cold) of uranium zirconium carbide particles, then we went up through the hot section of the uranium hafnium carbide. We used hafnium 180. The reason we used hafnium 180 is that the code was set up with hafnium 180 properties, so it was an easy way to make our first run using this isotope.

The significant point is that we did get a K effective greater than one. We had a fuel loading of a half gram of uranium 235 per cc. It indicates that we could operate at higher temperatures if the structural properties of the fuels were adequate. There is no data on these materials at present.

Now, what does this mean in specific impulse? Go back to the 1960s data and get the King report where they talked about the equilibrium data (see Figure 8). What does hydrogen look like at equilibrium as it comes out? You find that around 10 psi chamber pressure operating at 3,500 K you are over 1,400 seconds in specific impulse.

When we started on this work, we had a data base in the old NERVA code. In other words, we did have a thrust cell when we ran the XE tests. We ran nozzle tests out in the old Aerojet test area. We had some specific impulse data.

With a computer code, you have a table of temperature and pressure and you can go to areas where you haven't tested; namely, you can go to high temperature and low pressure.

When we first did that, we got some very favorable results and we said that this looked like it was worth considering. When you pull out Bussard's old data, (he wrote the "Bible" of nuclear propulsion in the old days) and look at his data plots, you will find you are well over 1,200 specific impulse. Corliss had a similar plot, indicating up around 1,200 or so.

The present state-of-the-art kinetics codes that Rocketdyne ran (the ODK code -- we ran the TDK code) are chemical kinetic codes designed for burning LOX hydrogen. They do have a hydrogen recombination routine in them, but it was a very small part of what was in the code. If you strip out all the LOX hydrogen and just use what is left, you will obtain the results shown on Figure 9. We and Rocketdyne got similar results. But if you check the data base for these, you will find that in the area that we are talking, there really is no data. Therefore, you don't know what kind of performance you are going to get.

The second point I would make is that if you start to play around with these codes and change the shape of your nozzle, you will find your performance improves (see Figure 10). In other words, you need resonance time for the recombination of hydrogen to occur. If you can get the recombination, you can begin to get the large performance improvements. You may call them losses in a conventional nozzle, whereas they may be a gain to you in this case.

How do you design a thrust chamber and a nozzle to maximize the performance you can get out of a dissociated and recombined hydrogen system? This is the type of thing that I am referring to (see Figure 11). This is again taken out of Bussard's data. What it shows is in a core, when you get to high temperature and low pressure, you get up to a factor of 10 apparent augmentation in your heat transfer. What it really amounts to is that, on the wall you are dissociating the hydrogen; it takes a lot of energy to dissociate the hydrogen. It dissociates on the wall, goes back into the mainstream and then recombines and increases in temperature. The net effect is an increase in heat transfer.

Based on this type of data, and talking with most of the people we can find, it appears that when you come out of the core, you will be in equilibrium dissociation. The problem is, as you get into the nozzle and begin the supersonic expansion, do you get the recombination that goes with the lower pressure? This can amount to as much as 1,500 degrees Kelvin difference in your exit temperature at the maximum expansion point of the nozzle. So there is a real issue of how do you expand that nozzle? We have looked at a lot of novel concepts and I will just show you one here in Figure 12.

Some of the things that have been rejected in the chemical engines, such as expansion deflection nozzle, spike nozzles, and plug nozzles, all become candidates for reexamination to see what would be the optimum way to design a thrust chamber/nozzle for hydrogen recombination.

We have not considered any of those advanced nozzles for our baseline studies. We stuck with a rather conventional thrust chamber bell nozzle approach.

MASE says you may have a requirement of two engines out. So, to have two engines out and do this mission, we thought you had to start building small engines. We picked as our reference an 11,000 pound thrust engine. We limited ourselves to a launch envelope (diameter) of 10 meters. We went through some trade-offs between the pressure and the expansion ratio.

We assume you could control thrust alignment with engine thrust (see Figure 13). In other words, with a nuclear engine, you can run the thrust up and down to get thrust alignment with your seven engines. You would abort the mission with any failures during the perigee pulse phase. After you left Earth with your perigee pulses, you can have the partial thrust with any two engines' failure after you left. The advantage of this is you have no gimbals. And you can completely assemble this thing on the ground.

We believe the small engines are going to be easier to develop and ground test. This clustering arrangement can be used for both lunar and planetary missions. We think we have a very versatile engine with this concept.

Figure 14 is a cartoon of a tank arrangement. We have our seven engines, each with a shield above it and then an elongated tanks above that. We took a penalty and put in part of our shielding into the bottom of these tanks. In other words, we have extra propellant on board in order to cut down on the weight of the disk shield.

The advantage of this is that when you are at high power, this propellant is available to you for shielding. When you shut down, you no longer require all the shielding, so you can use that propellant up as a way of doing your cool down. This looks like a way to save shielding weight.

This particular configuration also fits into what our ground rule says is the launch envelope. We have 10 meters in diameter and 30 meters in length. You can completely assemble it on the ground, and you can launch it as a unit. If you have the ten-hour life capability, you could even take this stage and use it for a lunar mission as part of your check out, then bring it back. After a lunar mission, you are sure you have a stage that works and you can then mount all the stuff up for a Mars mission. It is a pretty versatile stage.

For our mission analysis we picked three cases: low, medium, and high performance (see Figure 15). The low performance is the 3,200 K, the medium performance is 3,600 K, and the high performance is 3,600 K--dual mode--where we operate at 15,000 pounds thrust for everything except Earth departure.

We ran at 15 psi pressure for our main thrust, and 3 psi for our low thrust. The specific impulse for low performance would be 1,190 seconds, if we were to find a way to get hydrogen to equilibrium. If it were 1,012, it would completely be frozen, with no recombination. We picked 1,050.

We are very conservative in what we assume (see Figure 16). If we can get to 3,600 K, these jump to 1,400. We picked 1,210. Again, this is very conservative.

If we look at our dual mode performance, we picked 1,350 seconds. This is a little more optimistic, but it is based upon the gain that was predicted by Bussard and Corliss in the old days. It looked like they have done a lot of thinking about it because as they got to the point where the hydrogen densities became too low, they showed a loss of performance. So we use that as our basis and projected the 1350.

If you look at the mass in orbit, we looked at two missions (Figure 16). The reference mission left the engine in a huge ecliptic orbit around Earth, where it was going to take a lot of energy to make it reusable. We took advantage of the specific impulse we had by circularizing. It is one of the ways that you can take advantage of the increase in capability. You cut your initial mass in orbit in half, if you are going to leave it in the highly ecliptic orbit. If you are going to circularize, you gain almost a factor of three in your performance advantage. It looks like if you are willing to put that much mass in orbit, you can do the mission in a hundred days out and get a substantial gain in time.

If you look at reliability potential of this concept, you see the elimination of troublesome components (see Figure 17). We have eliminated the turbo pumps, the control drums, the engine gimbal and the valves, and the number of reactor parts have been reduced.

We have a complete "two-engine-out" capability, with a seven engine configuration. The low pressure does a lot for you on thermal problems. You get improved core heat transfer. Because of the dissociation/recombination, you have much reduction in your nozzle heat flux. Aerojet even proposed that we not cool it at all. You have the potential to not cool your pressure vessel because the heat flux is down, but we didn't take advantage of that. We assumed you had to cool it.

We picked three major safety areas (Figures 18 & 19): If you look at explosive rupture, you have no pumps. You operate below the tank pressure, so you are pretty sure there is no way to get a high pressure. In other words, it can't go over the tank pressures.

For reactivity insertion, we have eliminated the mechanical drums. There is a whole gamut of potential accidents we got rid of.

On loss of flow, which is the other major safety issue, you can manifold this to get your emergency flow from any one of the tanks, so that if any engine goes out, you can keep the flow into them.

The development program (Figure 20) is a fuel development program. I really believe that any concept that can get high temperature fuels will be able to get a good specific impulse. In order to prove your fuel, you are going to have to have reactors. In other words, you can run all the electrical tests you want, but if you read the final report on NERVA, they were arguing how good the electrical tests were. You have to get into a reactor. If you only consider the U.S. reactors, I think the fastest one you can get into that comes close to doing what you want is to go into the ATR. We projected you could get into there by the middle of 1994.

The best way to test is what we call the "nuclear furnace." What it really amounts to is a driver core with a hole in the center where you can test all kind of fuel elements. It's very versatile, gets the power densities you want and provides a real configuration.

We feel that you must have your environmental impact statement before you start on the facility. Therefore you really have a problem in getting into a reactor in fast order. As a solution to the problem, we went ahead and showed both types of contexts (see Figure 20). Ultimately, you have to get into your engine testing.

We have some cost data (Figure 21). We have two big costs; lab fuel development and environmental impact statement. The design work on the engine is very small. Generally, we talk of a few million dollars to do an environmental impact statement. When you get into this environmental impact statement, you are going to have to do a study that says <u>where</u> you are going to test. You are also going to have to do a study that says <u>how</u> do you want to test. It is more than a typical environmental impact statement, so I put in $7 million to do the whole job.

In order to get these things available to you by the end of year four, you must spend most of your money on getting the facilities ready. By the end of year four you would have resolved the issues of temperature, fuel form, dissociation/recombination, and engine design. You would have made the decision of what performance you are going to get and how you package this thing and put it together.

I came up with $4 billion for the whole program. But I have a lot in there (Figure 22). I have defined all the tests in Figures 23-28.

I had 11 complete engine tests to get qualified. I built three flight engines. I tested for three years in the test reactor. In the nuclear furnace, I tested the whole time. In the cases when I completed my development program, I kept those facilities operating on my quality control. In other words, I continue to use the nuclear furnace to check out the what is being built at that point.

In order to get through this, I will just summarize the major technical issues (Figure 29).

First of all, you have to look at the nozzle pressure vessel design to optimize performance. You are talking hundreds of points of specific impulse, if you can find a way to to approach equilibrium recombination. It is really worth looking carefully at that because it is one of the biggest payoffs you are going to get.

The second point is you have got to be able to have a good flow/power match within the fuel element and core. There was a lot of money spent in NERVA getting that match. They were talking about running at 3,200 K core outlet gas temperature, with a material that melts at 3,600 K; there were 400 degrees (which is a lot of specific impulse) that went with the mismatch in order to put a real engine together. We have got to be that good, or better, to get any of the performance claims we have made, so you have to look into that detailed design.

It's going to cost more to test this on the ground. Because we are at low pressure, you have got to put some pumping systems in to run your exhaust clean up system.

You must decide what fuel form you are going to use to operate at these maximum temperatures.

We have assumed that you don't need pumps. We have come to some preliminary pressure drop calculations that looks as though you can do it. But within this core you have got to have a lot of little cooling channels that keep everything cool. In order to get your flow to distribute through these cooling channels, you have to have pressure drop. We haven't done all the detailed design work to see if you can really keep everything cooled properly. We also need to investigate the viability of the feedback power control.

To summarize this (see Figure 30), we have not identified any problems that require technical breakthroughs. There are many engineering problems that could reduce the performance. Typically these things go against you. But we have been on the very conservative side as far as the dissociation/recombination issue goes.

Everything ought to be a plus in that area if we can find a way to design it. So we have plus pluses and minuses. We think the performance, reliability and safety makes a promising candidate for early development and we think you ought to start on it next year.

LPNTR Has Inherent Advantages In Critical NTR Requirement

Performance	1SP 1050-1350, up to 6/1 T/W
Reliability	Potential to eliminate troublesome components
Safety	Reduced susceptability to safety critical failures
Versatility	Two engine out capability Multimode operation for maximum performance

Currently at NASA level 2

0-7790

Concept verification required

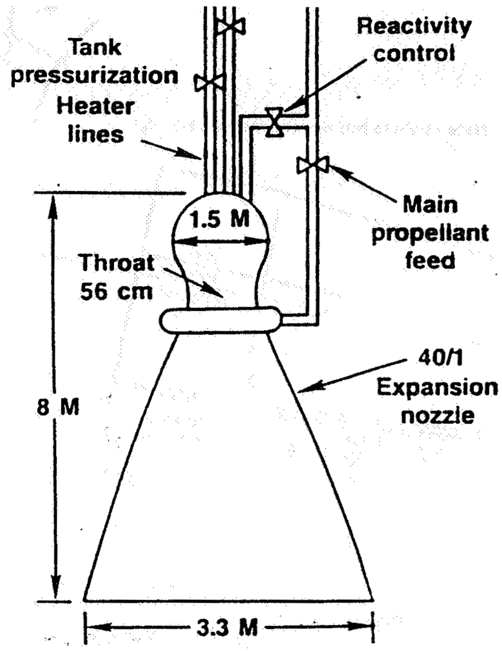
Figure 1

Preliminary Considerations Reactor Trade Studies

• Safety	Eliminate inherently unsafe design features
• Performance	
High temp	— Favorable neutronics for highest temp fuels
Minimum temp losses	— Good power/flow matching no leakage
Low pressure	— Optimum nozzle/reactor engine configuration
• Operational utiiity	
Weight	— $>$ 6/1 T/W
Reliability	— Simplicity, minimum of troublesome components
Other	— Look for obvious problem areas

0-7487

Figure 2



Reference LPNTR

- Concept designed to maximize flow at low pressure & high temp

Features

Radial outflow core
Particle or plate fuel
Operates on tank pressure
Reactivity power control

Performance

Thrust 11,000#
Weight 1840#
T/W ~ 6/1
1sp ~ 1050-1210 @ full thrust
~ 1350 @ low thrust

0-7785

Figure 3

Preliminary LPNTR

Internal Configuration and Flow

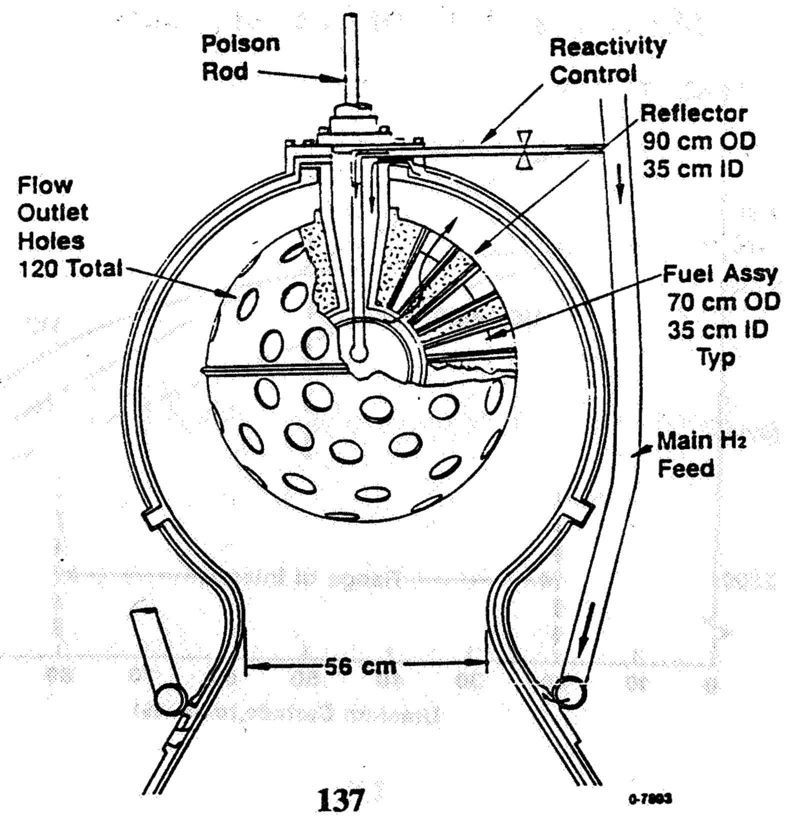


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Figure 4

LPNTR - Particle Bed Fuel Assembly Schematic

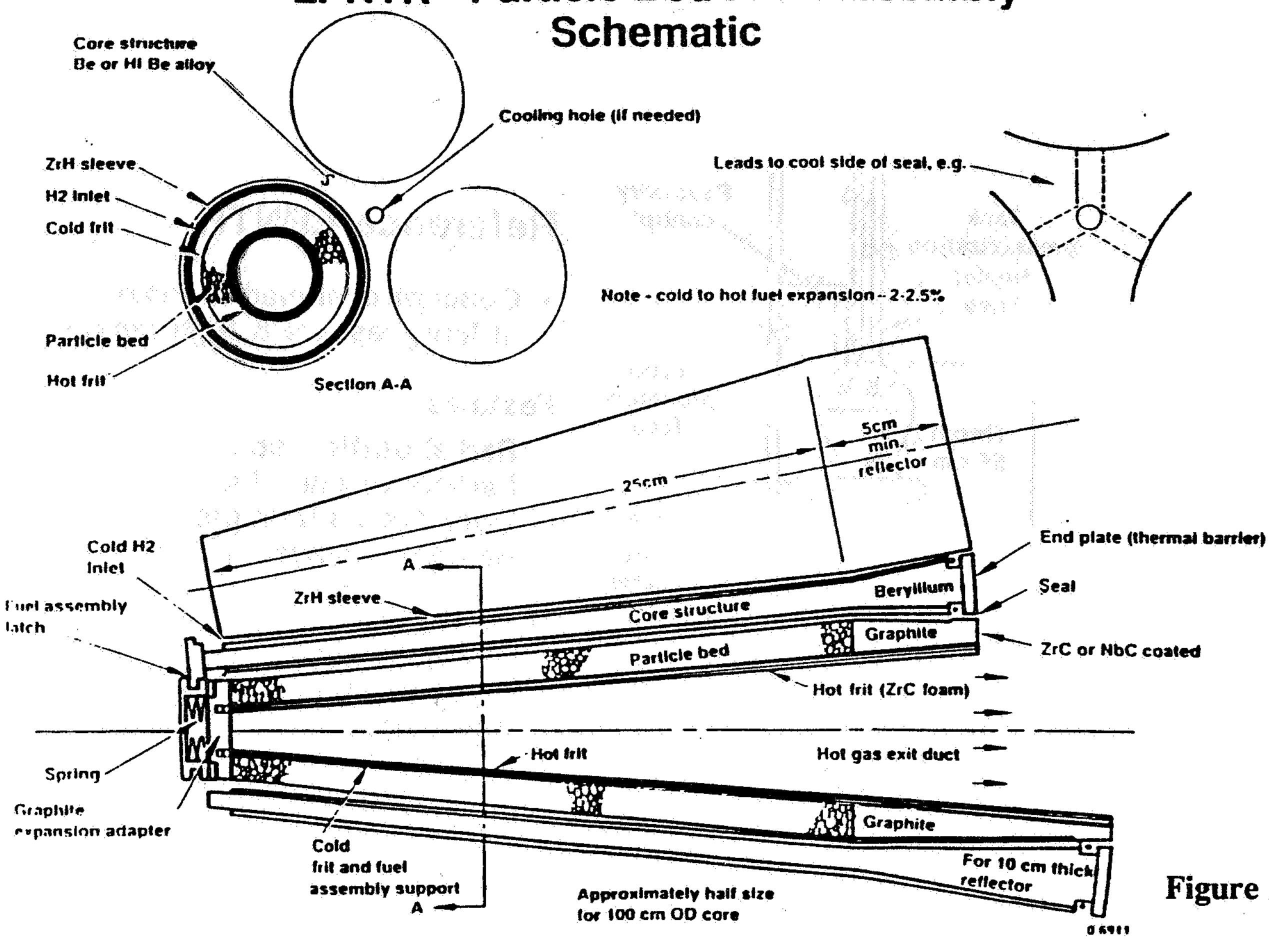


Figure 5

Melting Points of Ternary Carbide Fuels

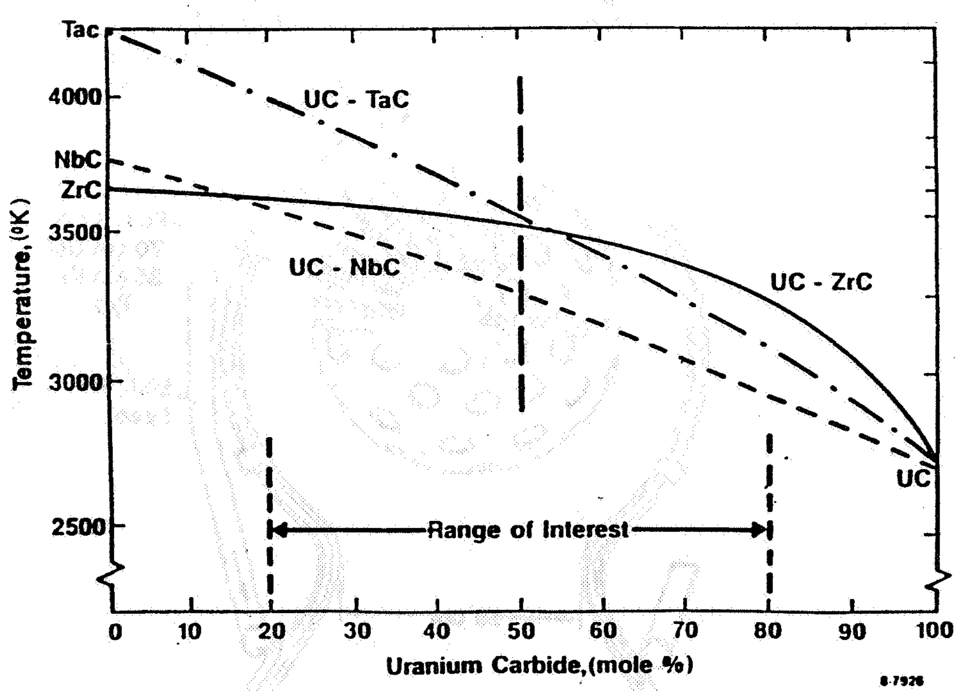


Figure 6

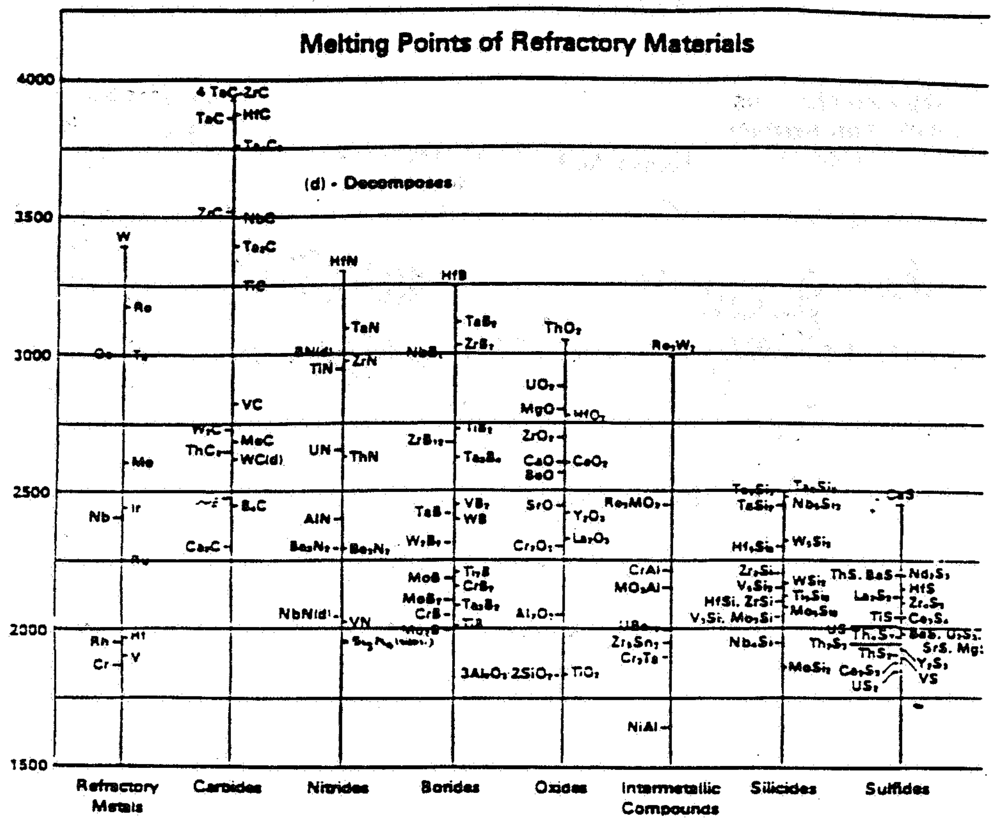


Figure 7

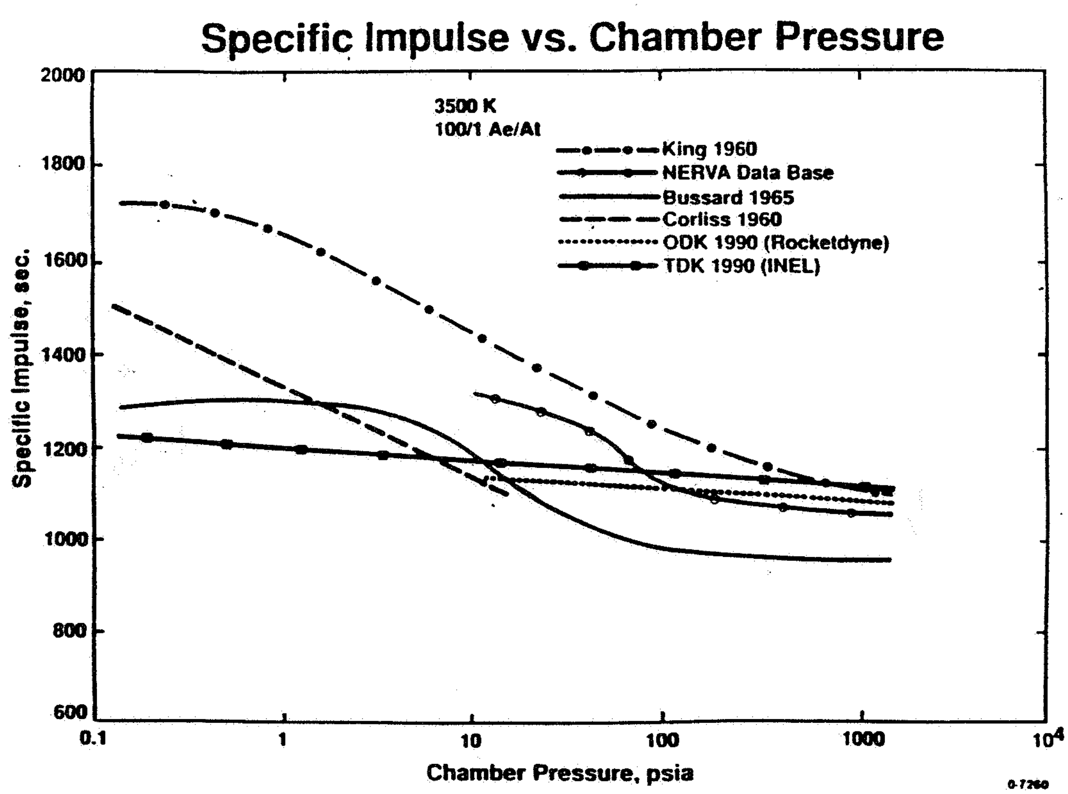


Figure 8

Preliminary LPNTR Neutronic Study Results

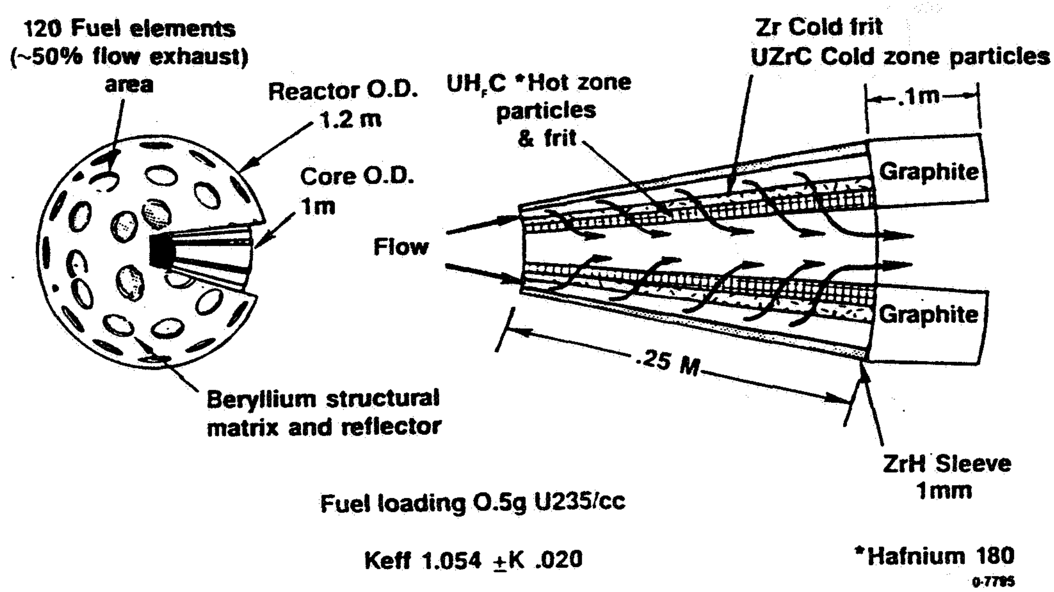


Figure 9

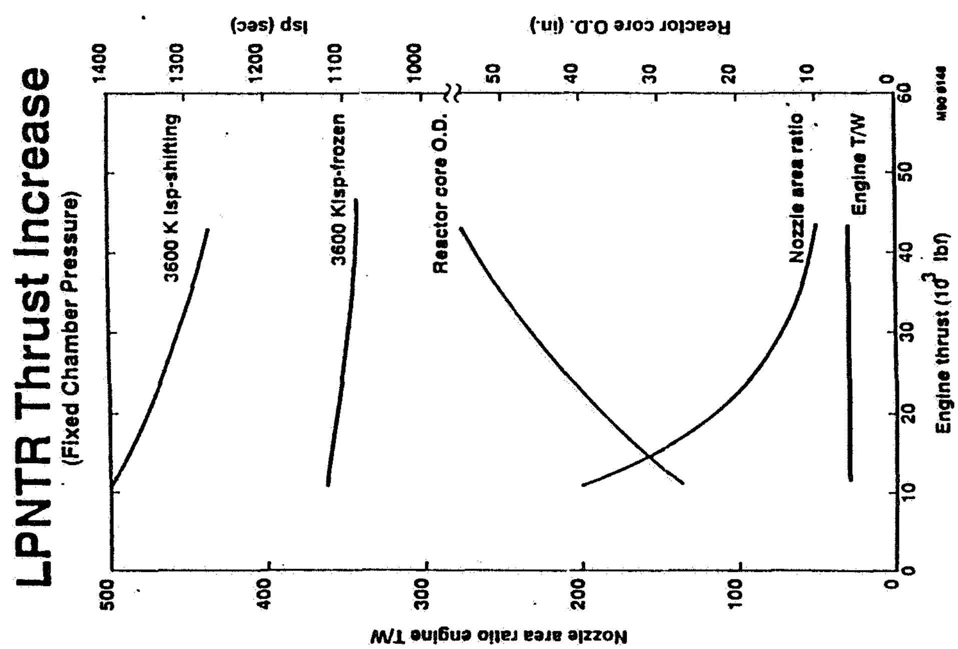


Figure 10

Estimated Augmentation Factor for Dissociation-Recombination Effects in Convective Heat Transfer to Hydrogen (Bussard 1965)

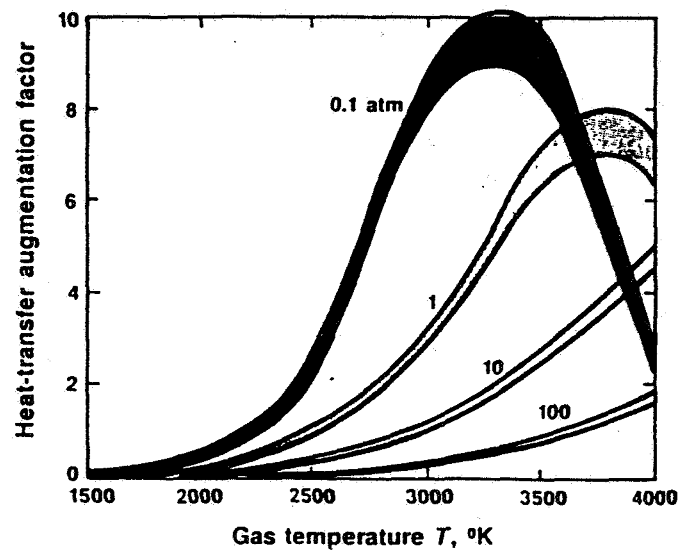


Figure 11

11,000 lbf Thrust LPNTR with Expansion/Deflection Nozzle

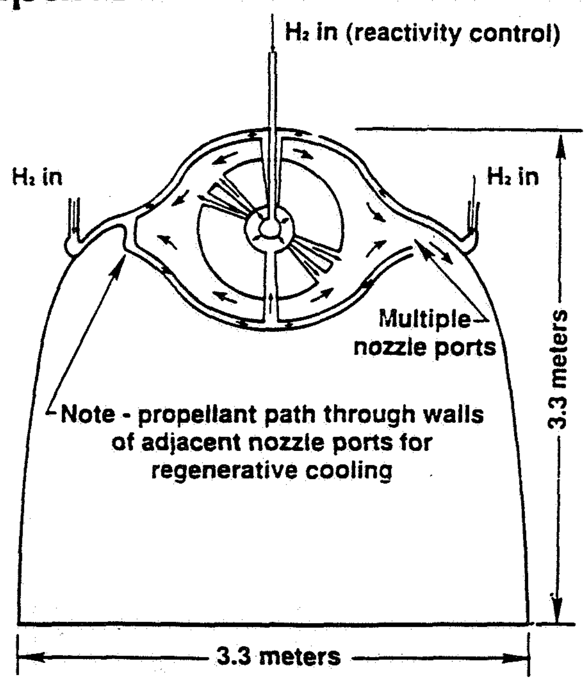
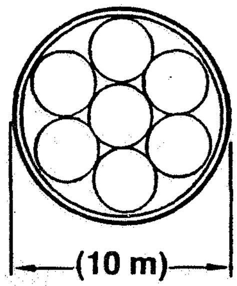


Figure 12

Multiple LPNTR Engine Concept



Assumptions

- Control thrust alignment with engine thrust
- Abort mission with any failures during perigee pulse
- Partial thrust for any two engine failures after Mars injection

Advantages

- No gimbles
- Ground assembly
- Small engines easier to develop and ground test
- Smaller clusters for Lunar and Planetary missions

Figure 13

LPNTR
Tank & Engine Configuration for Mission Analysis

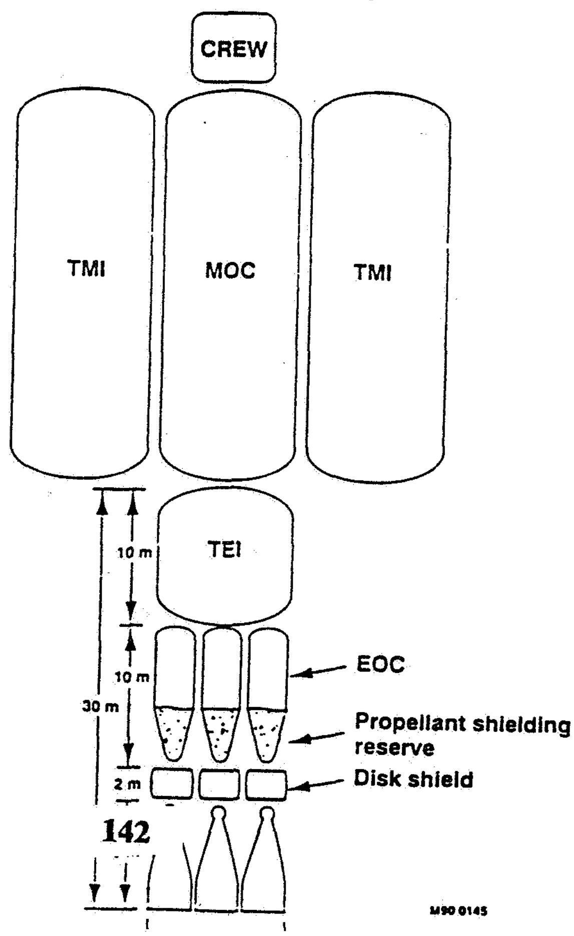


Figure 14

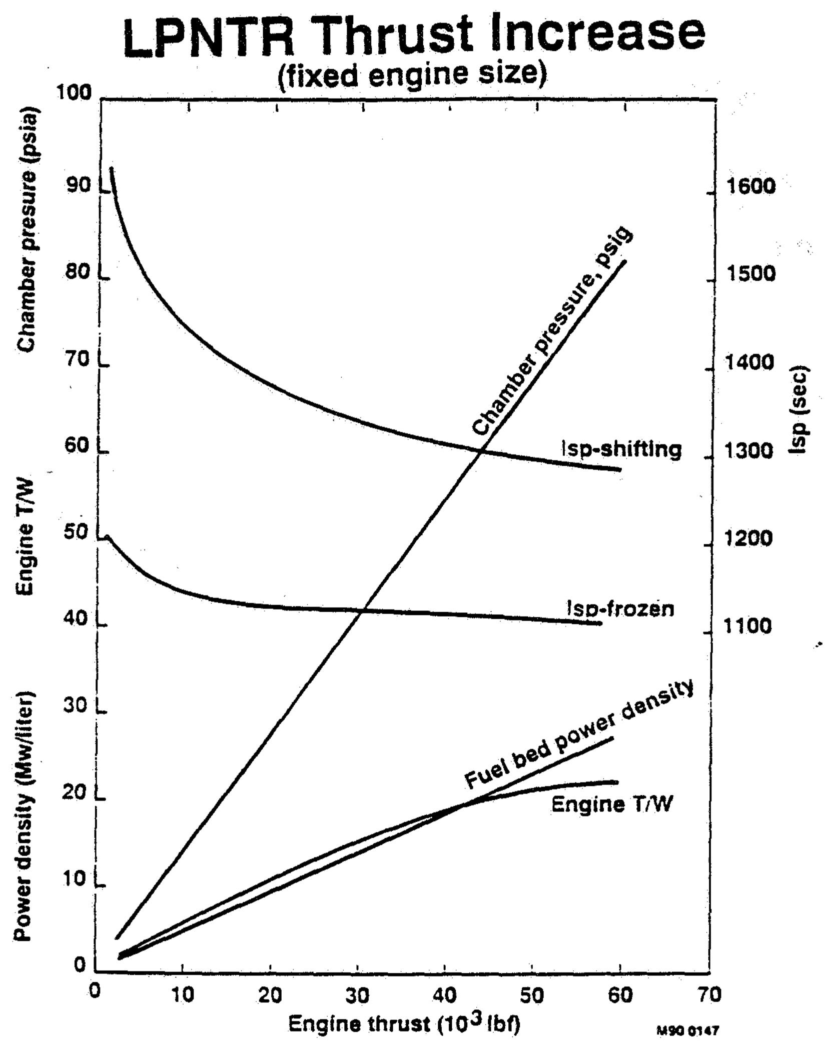


Figure 15

IMEO Advantate LPNTR

Engine	Isp	T/W: Engine	T/W: Engine plus shield	Mission: Ref / IMEO	Mission: 500 Km Earth orbit / IMEO
Ref (NERVA)	850	4	2.6	884	1400
Advanced NERVA	925	6	3.3	713	1037
LPNTR 3200 K	1050	6	2.2	603	814
LPNTR 3600 K	1210	6	2.2	485	611
LPNTR 3600 K Dual mode	1210 1350	6 1.25	2.2 0.44	440	534

Figure 16

LPNTR Reliability Potential

• Potential to reduce or eliminate troublesome components	Turbo pump — eliminated Control drums — eliminated Engine gimbal — eliminated Valves — reduced Reactor parts — reduced
• Small engine size gives 2 engine out capability	Any two failures of 7 engine configuration
• Low pressure reduces thermal problems	Improved core heat transfer — dissociation/recombination Lower nozzle heat flux

0-7786

Figure 17

LPNTR Reduces Susceptability to Safety Critical Failures

Explosive rupture	— No pumps - operates below tank pressure
Reactivity insertion	— Mechanical drums eliminated
Loss of flow	— Engines can be manifolded to get emergency flow from all tanks

0-7787

Figure 18

Safety Considerations

NERVA XE Safety Analysis Report: three major accidents

Control drum failure	50 pages
Liquid hydrogen insertion	14 pages
Loss of propellant flow	7 pages

∴ Elimination of control drum should eliminate many safety problems

0-7489

Figure 19

LPNTR Development Program

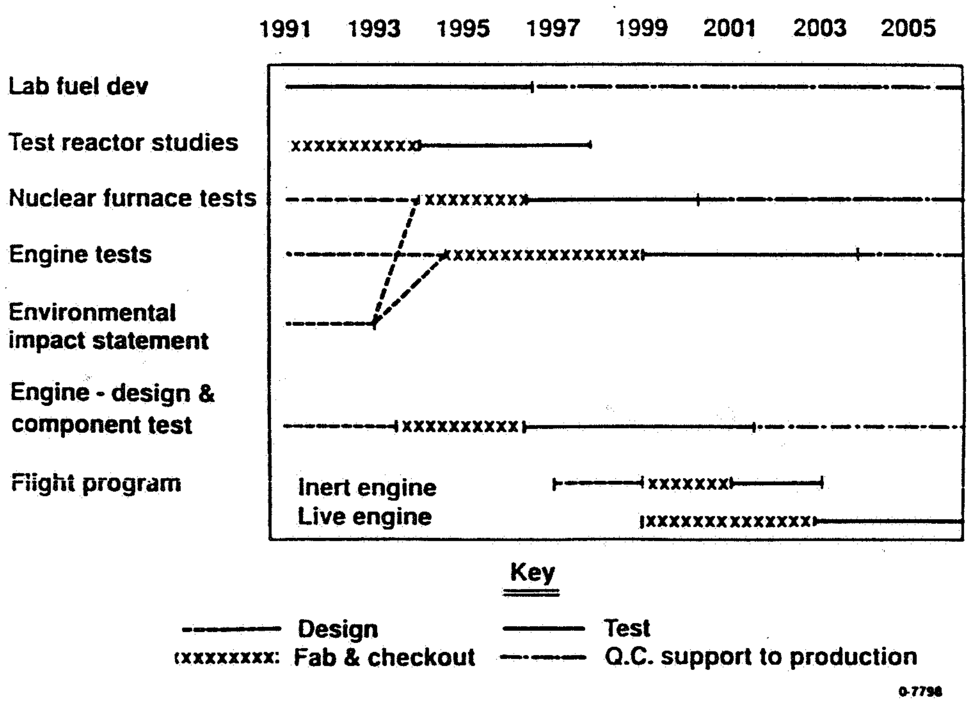


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Figure 20

LPNTR Program Initiation Costs

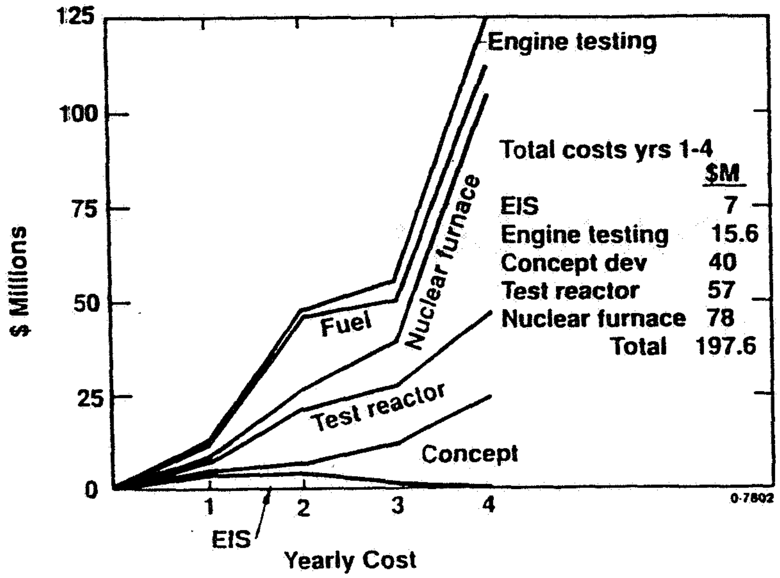

Figure 21

Total Program Cost 1991 - 2006

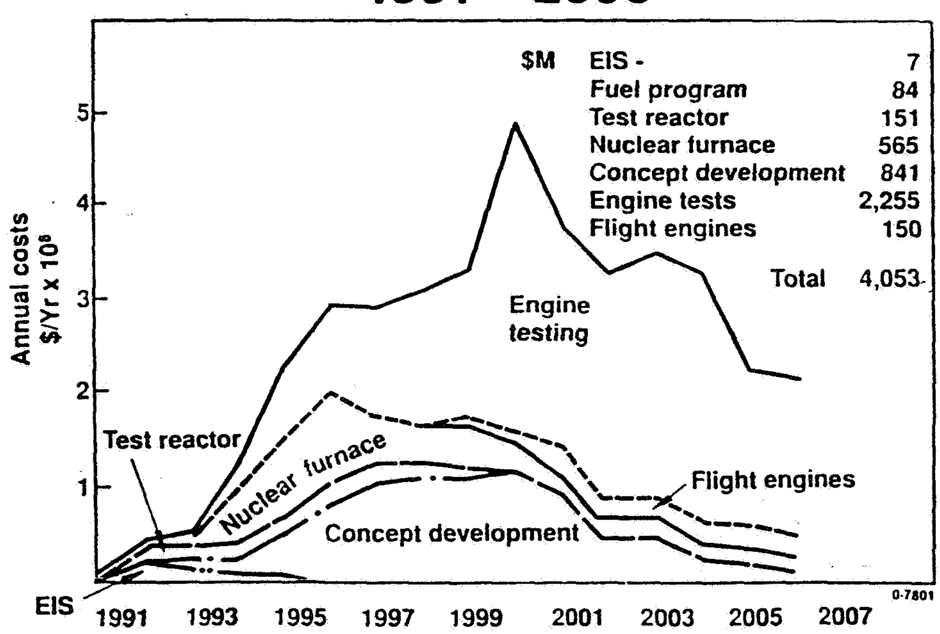

Figure 22

LPNTR Development
Two Qualified Flight Engines delivered to
Launch Site

All $ in Millions

Item	Cost	Output
EIS	$ 7m	Test site selected Program Environmental Approval obtained
Fuel Program	$ 84m	Developed fuel, electrical test facility 10 years of quality control check on fuel production.
Test Reactor	$ 151m	Hydrogen test loop; 3.5 years development testing of sub scale fuel assemblies; D&D of loop
Nuclear Furnace	$ 565m	- 60MW driver reactor; hydrogen test loop, prototype exhaust scrubber; four years full scale fuel assembly development testing; Six years of quality control testing of fuel assemblies.
Concept Development	$ 841m	Program Management, developed concept, qualification of non-nuclear components.
Engine Testing	$2,255	Three reactor development tests. Four engine development tests. Three engine qualification tests. Two quality control engine tests for flight support.
Flight Engines	$ 150m	Inert engine. Two qualified production engines.

Figure 23

Total Program Costs
$ Millions

	EIS	Design & Fuel Dev.	Component Qualification	Test Reactor	Nuclear Furnace	Engine Testing	Flight Engines	Total Annual Cos
1991	3	4	1	2	1	.3		11.3
1992	3	16	3	17	6	.3		45.3
1993	1	10	12	16	12	5		56
1994		9	24.	22	59	10		124.0
1995		8	50.2	22	83	65		228.2
1996		7	79.5	22	110	75		293.5
1997		3	103.0	22	54	105		287.0
1998		3	109.5	17	40	140		309.0
1999		3	109.0	11	40	170	10	343.5
2000		3	116		40	320	10	489
2001		3	91		20	264	30	408
2002		3	44		20	260.4	20	347.4
2003		3	44		20	260	20	347
2004		3	22.5		20	260	20	325.5
2005		3	18.5		20	160	20	221.5
2006		3	14.0		20	160	20	217.0
	7	84	841.2	151	565	2255	150	$4053.2

Approximately $4B

Figure 24

Engine Facility

	Facility Construction Cost	Operations Cost	Experiment Costs	
1991	.3			.3
1992	.3			.3
1993	5			5
1994	10			10
1995	60	5		65
1996	80	10	15	75
1997	70	20	15	105
1998	70	40	30	140
1999	50	60	50	170
2000	60	60	200	320
2001	4	60	200	264
2002	.4	60	200	260.4
2003		60	200	260
2004		60	200	260
2005		60	100	160
2006		60	100	160
	400	550	1,300	$2,255

Figure 25

Nuclear Furnace

	Reactor Design & Fab	Facility Design & Fab	Operations	Experiment Lab and Analysis	Total
1991	1				1
1992	5	1			6
1993	10	2			12
1994	50	3	3	3	59
1995	60	9	7	7	83
1996	60	10	20	20	110
1997	14		20	20	54
1998			20	20	40
1999			20	20	40
2000			20	20	40
2001-2006			10/Yr = 60	10/Yr = 60	20/Yr = 120
	200	25	170	170	565m

Figure 26

Program Management Engine Design
Non-Nuclear Component Qualification

	Project Management	Mission Analysis	Flight Safety	Stage Interface	Engine System	Nuclear Subsystem	Total
1991	.2				.5	.3	1.0
1992	.7	.1	.1	.1	1.3	0.7	3.0
1993	1.7	.1	.1	.1	5	5	12.0
1994	3.7	.1	.1	.1	10	10	24.0
1995	8	1.0	1.0	.2	20	20	50.2
1996	15	2.0	2.0	.5	30	30	79.5
1997	17	2.0	2.0	2.0	40	40	103.0
1998	18	1.0	2.0	8	40	40	109.0
1999	18	0.5	2.0	9	40	40	109
2000	19		7.0	10	40	40	116
2001	14		8.0	9	30	30	91
2002	7		9.0	8	10	10	44
2003	7		10.0	7	10	10	44
2004	2.5		8.0		6	6	22.5
2005	2.5		6.0		5	5	18.5
2006	2		4.0		4	4	14.0
	136.3	6.8	61.3	54	291.8	291	841.2

Figure 27

Fuel Development Funding

	Labor & Mat'ls	Test Facility	Facility Operations	Emp. Cost	D&D	Total
1991	4					4
1992	4	10	2			16
1993	8		2			10
1994	7		2			9
1995	6		2			8
1996	5		2			7
	34	10	10			54
1997-2006	2/Y = 20m		1 1/2- 10m			3/Yr = 30m

Total: = 84m

Test Reactor Funding

1991	2				2
1992	10		7		17
1993	4	5	7		16
1994		10	12		22
1995		10	12		22
1996		10	12		22
1997		10	12		22
1998		5	12		17
1999			6	5	11
	16	50	80	5	151

= $150

Assume $150m

Figure 28

LPNTR Major Technical Issues

1) Nozzle pressure vessel design to optimize performance
2) Flow/power match within fuel element and core
3) Cost of ground test facilities
4) Fuel form/maximum operating temperatures
5) Total pressure drop
6) Viability feedback power control

0-7793

Figure 29

LPNTR Technical Summary

- No problems identified which require technical breakthroughs
- Many engineering problems exist which could reduce performance
- Improved performance could be obtained with revised thrust chamber/nozzle configurations
- Performance, reliability, and safety makes LPNTR a promising candidate for early development
- Technology verification should initiate in FY91

0-7786

Figure 30

N92-11095

PARTICLE BED REACTOR NUCLEAR ROCKET CONCEPT

Hans Ludewig
Brookhaven National Laboratory

It is gratifying to see that we are not the only ones talking about the particle bed reactor anymore (Refer to concept just presented by J. Ramsthaler).

The concept (see Figure 1) consists of fuel particles, in this case (U,Zr)C with an outer coat of zirconium carbide. These particles are packed in an annular bed surrounded by two frits (porous tubes) forming a fuel element; the outer one being a cold frit, the inner one being a hot frit. The fuel elements are cooled by hydrogen passing in through the moderator. These elements are assembled in a reactor assembly in a hexagonal pattern. The reactor can be either reflected or not, depending on the design, and either 19 or 37 elements, are used. Propellant enters in the top, passes through the moderator fuel element and out through the nozzle.

Beryllium is used for the moderator in this particular design to withstand the high radiation exposure implied by the long run times.

As far as design philosophy is concerned, I would like to introduce another parameter (Figure 2). Stan Gunn talked about the importance of specific impulse. I would like to talk about the added importance of thrust-to-weight ratio as well. Mission analyses indicate that the thrust-to-weight ration should be above 4.0.

We looked at two reactor designs; one that tried to maximize the thrust-to-weight and one tried to maximize the specific impulse (Figure 3). To maximize the thrust-to-weight requires a high power density, high pressure, and high temperature. These requirements result in a small, high thrust reactor.

The high specific impulse design operates at reduced pressure to introduce some dissociation of the hydrogen and thus increase the specific impulse. A low power density is implied by operating at a low pressure. Because of the lower density of the gas, the engine becomes bigger, heavier, and the thrust is lower.

These are the parameters which were considered (See Figure 3). The engines range from 1,000 megawatts to 5,000 megawatts, in the high thrust-to-weight cases and 500 to 2,000 megawatts in the specific impulse case.

Power densities in the bed were also varied. This is not average power density of the core, but in the bed. The chamber temperatures range over 2,500K to 3,500 K and in the low pressure case we increased the temperature beyond from 3,000 K to 3,750 K.

The pressures range considered was 7 MPa - 14 MPa, depending on power density. At the higher bed power density, higher pressures are required. The low pressure case operated at a much lower pressure; 0.5 MPA.

We did full up analyses of these cores. These reactors were all found to be critical and coolable. We took into account pressure drops and heat transfer in the fluid dynamics analyses.

An important point I want to make here is that thrust to weight ratio drops (Figure 4) when comparing the two reactor design philosophies. These are unshielded and still within the limits of the baseline. However, as soon as one adds on a shield, and again this shield is a fairly cavalier design, one notices that the low pressure design drops way down and is below the baseline requirement.

Technology status (see Figure 5) is divided into analysis, proof of principle experiments and prototype experiments. As far as analysis is concerned, we use the Monte Carlo code (MCNP) that is standard in the industry.

In the case of fluid dynamics, we did have to generate our own codes. One cannot use an off-the-shelf fluid dynamics code and modify it. We made a 1-D survey code and transient code to study start-up. These were reported on at the Albuquerque meetings in 1987.
We use the standard Ergun correlation for pressure drop in the bed. There has been additional work by Achenbach that essentially confirms this work and that was reported in 1982 in Munich.

As far as the materials work is concerned, we have done various tests and the most significant had to do with the compatibility of zirconium carbides and hydrogen. Again, this was reported in 1985 in Albuquerque.

As far as the electrically heated tests are concerned, we built full diameter, half length fuel elements, and demonstrated that we can extract ten megawatts per liter from the bed.

In the case of fuel development, many people have looked at zirconium carbide coated fuel particles. I just refered to an ORNL report here, but work has gone on in this country. The Germans have looked at it, and so have the Soviets and Japanese. As for the UC/ZrC kernel, there is a reference that goes back to 1963 that reported manufacturing these. So I would put the technology readiness of this concept at around four.

The other item we were asked to address was the potential for new technology and safety requirements (see Figure 6). I think that for our concept, coatings are important. The mixed carbide coatings which have a melting point of about 4,000 K would really help.

Finally, enhanced light weight structures are important. Particularly if one can make them out of low Z materials in an effort to reduce the radiation heating, particularly if high power densities are required to maximize the thrust-to-weight. The platelet technology which Aerojet worked on for some time for reentry vehicles would be very useful in our moderators.

Safety issues are generic for most concepts (see Figure 7). Fuel element test reactor safety is uppermost in our work. The ETR (Element Test Reactor) will be used to develop the fuel element for the full scale reactor.

Ground test facilities are required to test several engines, to develop a reliable system. I would like to see a space craft with at least three engines on it, and that's where the high thrust-to-weight ratio requirements comes in. If one can design an engine that has a high thrust-to-weight ratio, one can afford to put several of them on the vehicle and still meet the thrust-to-weight goal.

Launch criticality and Earth reentry; these are standard accident scenarios that we all have to analyze.

Several energy release scenarios exist. Those associated with hydrogen deflagrations/detonations will probably be more important than those from nuclear events. I think we all know what is required there.

We think that we can propose multiple engines with our concept (see Figure 8). If we select a high thrust-to-weight ratio, small shields are implied. These would be smaller since they don't have to be shadow shields and they would also be easier to decouple, assuming that's a requirement.

The fuel particles are small and most particles in the bed are relatively cool. The only ones that are hot are the ones that are closest to the hot frit. Three-quarters of them will be cooler and thus failure and fission product release is expected to be low.

We have tried to make our designs using light weight materials with low Z to reduce the radiation heating effects. The thermal gradients are fairly moderate across most components, implying low thermal stress.

As far as key technology issues are concerned for high temperature particles, the erosion resistance is certainly important (see Figure 9). I would like to point out at this stage that the velocity of the coolant through the bed is of the order of 50 to 100 meters per second. Tests should be done on particles in hydrogen at about 7 MPa, at operating temperatures of about 3,000 K at that velocity.

Again, the same comments hold for the frit. The velocities are again the same since the coolant flows radially through the frit. The cold frit has to be manufactured, as was pointed out earlier, to have variable porosity to shape the flow.

We have a large selection of moderators at our disposal. In the current design, we use beryllium. However, various materials can be used, since the moderator operates at inlet temperature. Thus, we can use it to maximize whatever parameter we want to maximize.

It is important to carry out an integrated element test (see Figure 10). This should be done in a test reactor. We would test for cyclability, and also demonstrate that we don't have any auto catalytic failure modes.

As far as the rest of the engine is concerned, I think a radiatively cooled carbon/carbon nozzle should be developed. It has to be nuclear-radiation resistant, erosion resistant, and joined with the pressure vessel.

The key technology for the turbo pump, would be development of carbon/carbon rotors in order to reduce the heating and operate at reactor outlet temperature.

The schedule and costs have been divided into four major tasks before the year 2006: design analysis, technology development, engine test reactor system, and then the GTE, which would be the ground test system (see Figure 11).

The first task is a design analysis which continues through the CDR (Critical Design Review) for the flight test engine. Technology development would include tests, primarily on fuel, coating, and frit materials. The element test reactor would be used to carry out the integrated test on the fuel element.

We estimate that the entire program would cost one and a half billion dollars. Approximately a billion dollars would be required for the program to advance through to the ground test.

In the first year we will develop an engine design compatible with the mission (see Figures 12 and 13). In carrying out this task, we need to follow these philosophies: maximizing the thrust-to-weight or the specific impulse, depending on the system analysis; developing a plan to carry out the proof of principle test; and then of course starting the experimental work.

In phase one, the engine work will be continued. We will demonstrate high temperature particles to meet the mission, demonstrate that we can build hot and cold frits that would meet the mission cyclability, and operate full-scale elements in the test reactor. We would have to carry out a critical experiment. Nobody mentioned a critical experiment yet, but that's a physics test to make sure the physics methods are validated.

In order to develop the fuel element design, one would first carry out electrically heated tests and then eventually nuclear heated tests. Design of the ETR, which is the element test reactor, would be a major effort. There would have to be some work on the carbon/carbon nozzle. Finally the demonstration of carbon-carbon turbine rotors and mixer will be required.

For phase two, we have to select the site for the element test reactor and satisfy all safety requirements (Figure 14). We would prepare the site and then construct and carry out the test. I am sure that there are many other tasks in there, but that's approximately five years away.

As far as major facilities are concerned, critical experiments could be carried out at the available facilities; Los Alamos, or ANL (see Figure 15).

We would have to have a fluid dynamics test facility to check the two phase flow problems involved in start up. A large amount of hydrogen will be required and probably some of the NASA labs would be good candidates for these tests.

An ETR site would have to be selected. It is not clear where one would construct it. It might be concept-specific. I am sure that the test cavity in the middle of the reactor to test concepts would be different depending on the concept. Again, the site for the GTE would have to be selected. Of course, the GTE would be concept-specific, as well.

Finally the GTE might have to have an altitude chamber to simulate start up, particularly if one is going to have a regeneratively cooled nozzle, since the pressure drop must be simulated, implying a sufficiently large nozzle.

In conclusion, we feel that the PBR has several advantages for this mission (Figure 16). High heat transfer allows it to operate at very high power densities for a given total power. Thus we can design a very high-thrust, light-weight reactor. This would be useful if one wants to use redundant engines. Direct cooling of the particles enables one to operate as close as possible to the material limits of the coating. The coolant flow path ensures that all internal components of the reactor, moderator, control rods and so forth operate at inlet temperatures. This ensures reliable operations. And finally we feel that for solid core rockets, this concept would get the closest to the achievable limits, whether one wants to maximize thrust-to-weight or specific impulse.

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PBR Based Concept

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SCHEMATIC REPRESENTATION OF A PARTICLE BED REACTOR BASED ROCKET CONCEPT

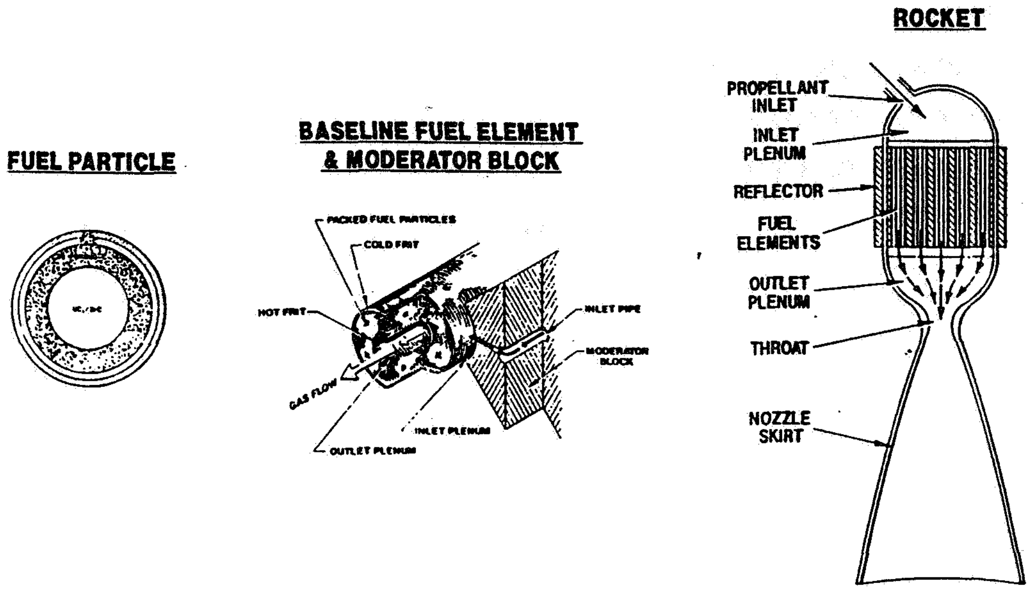

Figure 1

DESIGN PHILOSOPHY

- MAXIMIZE THRUST/WEIGHT
  - HIGH POWER DENSITY
  - HIGH PRESSURE
  - HIGH TEMPERATURE
  - SMALL SIZE
  - HIGH THRUST

- MAXIMIZE SPECIFIC IMPULSE
  - LOW POWER DENSITY
  - LOW PRESSURE
  - ULTRA HIGH TEMPERATURE
  - LARGE SIZE
  - LOW THRUST

Figure 2

ENGINE PARAMETERS

	HIGH THRUST/WEIGHT	HIGH SPECIFIC IMPULSE
POWER (MW)	1000 - 5000	500 - 2000
AVERAGE BED POWER DENSITY (MW/L)	20 - 80	5
CHAMBER TEMPERATURE (K°)	2500 - 3500	3000 - 3750
CHAMBER PRESSURE (MPA)	7.0 - 14.0	0.5
SPECIFIC IMPULSE (S)	850 - 1060	1000 - 1300
THRUST (N)	2.0 (5) - 1.0 (6)	6.0 (4) - 2.0 (5)

Figure 3

CALCULATED PARAMETERS

	HIGH THRUST/WEIGHT	HIGH SPECIFIC IMPULSE
TOTAL ENGINE MASS (W/O SHIELD (kg)	650 - 5500	2800 - 6000
THRUST/WEIGHT (W/O SHIELD)	20 - 35	4.0 - 7.5
SHIELD MASS (kg)	1300 - 6400	3700 - 7900
THRUST/WEIGHT (W/SHIELD)	8.6 - 14	2.0 - 3.2
MAXIMUM FUEL TEMPERATURE (K°)	2500 - 3650	3200 - 3900

Figure 4

STATUS OF TECHNOLOGY DEVELOPMENT

(BASED ON WORK CARRIED OUT FOR OTV AND MMW PROGRAMS)

	PHYSICS	FLUID DYNAMICS	HEAT TRANSFER	MATERIALS
ANALYSIS	EXPLICIT MONTE CARLO ANALYSIS - MCNP (LA-7396-M) (1986)	1-D SURVEY CODE 1-D TRANSIENT CODE (4TH SYM. ON S.N.P., ALB., NM) (1987)		--
PROOF OF PRINCIPLE EXPERIMENTS	--	PRESSURE DROP CORRELATION ERGUN (CHEM. ENG. PROG. 48:89-97) (1952)	HEAT TRANSFER CORRELATION ACHENBACH' (INT. HEAT TRANS. CONF. MUNICH) (1982)	COMPATIBILITY OF ZrC WITH H_2 (2ND SYM. ON S.N.P., ALB., NM) (1985)
PROTOTYPE EXPERIMENTS	--	ELECTRICALLY HEATED BLOWDOWN EXPERIMENTS (6TH SYM. ON S.N.P., ALB., NM) (1989)		ZrC COATED FUEL PARTICLES (HOMAN AND KANIA, ORNL/TM-9085, JAN. 1985), (UZr)-C FUEL PARTICLES (SYM. ON CARBIDES IN NUCL. ENG., HARWELL) (1963)

Figure 5

POTENTIAL NEW TECHNOLOGY AND SAFETY REGULATORY IMPACT

- HIGH TEMPERATURE COATING TECHNOLOGY FOR FRITS AND FUEL

- FIBER ENHANCED LIGHT WEIGHT STRUCTURAL MATERIALS
  - LOW Z TO MINIMIZE RADIATION HEATING

- PLATELET CONSTRUCTION OF COMPONENTS TO FACILITATE FLOW CONTROL AND COOLING.

Figure 6

SAFETY ISSUES TO BE ADDRESSED BY ALL NTR CONCEPTS

- FUEL ELEMENT TEST REACTOR SAFETY
- GROUND TEST FACILITY SAFETY FOR AN OPEN CYCLE REACTOR
- RELIABILITY/REDUNDANCY FOR SYSTEM MAN-RATING
- LAUNCH CRITICALITY ACCIDENTS
- EARTH RE-ENTRY ACCIDENTS
- ENERGY RELEASE OF POSSIBLE FAILURE SCENARIOS
- EXTENSIVE SAFETY REVIEW AND DOCUMENTATION EFFORT REQUIRED

Figure 7

POTENTIAL SAFETY ADVANTAGES OF CONCEPT

- COMPACT SIZE AND WEIGHT
  - MULTIPLE ENGINE REDUNDANCY POSSIBLE
  - EASIER TO SHIELD
  - EASIER TO NEUTRONICALLY DECOUPLE MULTIPLE ENGINES
- CONTAINMENT/CONFINEMENT CAPABILITY OF FUEL PARTICLES
  - REDUNDANCY
  - MOST PARTICLES ARE RELATIVELY COOL
- MOST CORE MATERIALS ARE COOL
- USE OF LIGHT-WEIGHT STRUCTURAL MATERIALS MINIMIZES RADIATION HEATING
- THERMAL GRADIENTS ACROSS MOST INDIVIDUAL COMPONENTS ARE SMALL

Figure 8

KEY TECHNOLOGY ISSUES

- HIGH TEMPERATURE PARTICLE/COATING
  - EROSION RESISTANT
  - NEUTRONICALLY BENIGN
  - COMPATIBLE WITH HOT FRIT

- HOT FRIT/COATING
  - EROSION RESISTANT
  - COMPATIBLE WITH PARTICLES
  - ACCEPTABLE MECHANICAL PROPERTIES

- COLD FRIT
  - MANUFACTURABLE WITH VARIABLE POROSITY
  - NEUTRONICALLY BENIGN

- MODERATOR
  - LARGE SELECTION OF MODERATOR POSSIBLE WITH PBR
  - SELECT MODERATOR WHICH WILL BE COMPATIBLE WITH MISSION PROFILE

Figure 9

KEY TECHNOLOGY ISSUES (cont'd)

INTEGRATED FUEL ELEMENT TEST

- DEMONSTRATE ABILITY OF FUEL ELEMENT AND THUS REACTOR TO REPEATEDLY CYCLE IN POWER FROM ZERO TO FULL POWER

- DEMONSTRATE MAXIMUM LIMIT IN ACHIEVABLE BED POWER DENSITY AND HOT CHANNEL FACTORS

- DEMONSTRATE STABLE OPERATION OF ELEMENT, NO AUTOCATALYTIC TEMPERATURE OR FUEL FAILURE MECHANISMS

- CARBON/CARBON NOZZLE - RADIATIVELY COOLED OPTION
  - EROSION RESISTANT
  - JOINT WITH PRESSURE VESSEL

- TURBO PUMP ASSEMBLY
  - CARBON/CARBON ROTORS FOR TURBINE

Figure 10

SCHEDULE AND COSTS

PHASE	I	II.	III	

FY: 90 92 94 96 98 00 02 04 06 08 10 12 14 16 18 20

ACTIVITY		COST ($M)
1. DESIGN AND ANALYSIS	1ST YEAR; CDR (ETR); CDR (FTE)	30
2. TECHNOLOGY DEVELOPMENT	COMPLETED TESTS	50
3. ELEMENT TEST REACTOR	CDR; SITE PREP.; TESTS COMPLETED	320
4. ENGINE DEVELOPMENT AND GTE	CDR; SITE PROPOSED; GTE MANUFACTURE; GTE TEST COMPLETED	500
5. SPACE QUALIFICATION	COMPLETE SPACE QUALIFICATION	600
		1500

Figure 11

CRITICAL TESTS/ACTIVITIES

- FIRST YEAR
  - DEVELOP ENGINE DESIGN COMPATIBLE WITH MISSION ANALYSIS
  - DEVELOP A PLAN FOR COMPONENT PROOF OF PRINCIPLE AND PROTOTYPIC EXPERIMENTS BASED ON ABOVE DESIGN
  - START EXPERIMENTAL WORK

Figure 12

CRITICAL TESTS/ACTIVITIES (cont'd)

- CRITICAL TEST - PHASE I
  - CONTINUE ENGINE DESIGN AND DEVELOPMENT
  - DEMONSTRATE A HIGH TEMPERATURE PARTICLE TO MEET MISSION NEEDS
  - DEMONSTRATE BOTH HOT AND COLD FRITS TO MEET DESIGN GOALS
  - OPERATE A FULL SIZE FUEL ELEMENT IN A TEST REACTOR (TREAT, ACRR)
  - CARRY OUT A CRITICAL EXPERIMENT
  - CARRY OUT PROTOTYPIC ELECTRICALLY HEATED FUEL ELEMENT FLOW EXPERIMENT TO DEMONSTRATE REPEATABLE, STABLE OPERATION AT MAXIMUM POWER DENSITY
  - DESIGN ELEMENT TEST REACTOR (ETR)
  - DEMONSTRATE CARBON/CARBON NOZZLE
  - DEMONSTRATE CARBON/CARBON TURBINE ROTORS
  - DEMONSTRATE MIXER FOR TURBINE FEED

Figure 13

CRITICAL TESTS/ACTIVITIES (cont'd)

- CRITICAL TESTS - PHASE II AND III
  - SELECT SITE FOR ELEMENT TEST REACTOR AND SATISFY ALL NECESSARY REGULATORY AND SAFETY AGENCY AND REQUIREMENTS
  - PREPARE TEST SITE FOR ETR AND GROUND TEST ENGINE (GTE)
  - CONSTRUCT AND CARRY OUT FUEL ELEMENT TESTS
  - DESIGN GROUND TEST ENGINE (GTE)
  - CONSTRUCT AND CARRY OUT GTE TEST PROGRAM

Figure 14

MAJOR FACILITIES REQUIREMENTS

- CRITICAL EXPERIMENTS (LANL, ANL (WEST AND EAST))
- FLUID DYNAMICS FLOW FACILITY TO VERIFY TWO-PHASE FLOW AND FLOW INDUCED VIBRATIONS EFFECTS DURING START-UP AND RUNNING
  - MUST HANDLE LARGE QUANTITIES OF HYDROGEN (NASA LABS)
- SITE FOR ETR - NEW
- ETR - NEW MAY BE CONCEPT SPECIFIC
- SITE FOR GTE (SAME AS FOR ETR (?))
- GTE - CONCEPT SPECIFIC
- GTE - ALTITUDE CHAMBER TO TEST START UP

Figure 15

CONCLUSION

- THE PBR HAS SEVERAL UNIQUE ATTRIBUTES WHICH MAKE IT ATTRACTIVE AS A PROPULSION REACTOR
  - HIGH HEAT TRANSFER AREA ENABLES REACTOR TO OPERATE AT HIGH BED POWER DENSITIES
  - FOR A GIVEN TOTAL POWER, THE HIGH POWER DENSITY RESULTS IN A SMALL AND THUS LOW MASS REACTOR - USEFUL IF REDUNDANT ENGINES ARE DESIRED
  - DIRECT COOLING OF PARTICLES RESULTS IN THE HIGHEST POSSIBLE GAS TEMPERATURE FOR ANY PARTICLE DESIGN - DESIRABLE FOR MAXIMIZING SPECIFIC IMPULSE
  - COOLANT FLOW PATH ENSURES THAT THE MODERATOR CONTROLS (INTERNAL OR EXTERNAL) AND MOST STRUCTURAL COMPONENTS OPERATE AT COOLANT INLET TEMPERATURES - ASSURES A WIDE SELECTION OF MODERATORS, ENSURES RELIABLE OPERATION OF CONTROL RODS AND STRUCTURAL COMPONENTS
- THESE ATTRIBUTES WILL RESULT IN A REACTOR DESIGN WHICH SHOULD APPROACH THE PRACTICALLY ACHIEVABLE LIMITS OF SPECIFIC IMPULSE AND THRUST/WEIGHT RATIO FOR A SOLID CORE REACTOR DESIGN

Figure 16

A CERMET FUEL REACTOR FOR NUCLEAR THERMAL PROPULSION

Gordon Kruger
General Electric

I want to talk to you about the cermet fuel reactor. I will discuss the work that was done in the 1960s. Very little work has been done since that time.

The cermet reactor work came out of both the ROVER program and the aircraft nuclear propulsion program (Figure 1). The 710 program was conducted at General Electric in Cincinnati while the nuclear rocket program was conducted by ANL; these programs were complementary. They both used the same kinds of fuel materials and both supported the same kinds of goals and objectives. The goals were to develop systems that could be used for nuclear rocket propulsion as well as closed-cycle propulsion system designs for ship propulsion, space nuclear propulsion, and other propulsion systems.

Part of that work involved fuel materials fabrication. There were reactor physics experiments, and there was an engineering analysis, and fuel test program.
What I would like to do is give you a little background on both the 710 program at GE, and then the ANL program so you will have an understanding of the work that has been accomplished so far.

At GE there were a number of different facets to the program (Figure 2). The 710 program goal was a 10,000 hour continuous operation design life for the closed cycle designs. They also had goals for a nuclear rocket. Design and control analyses were performed and fuel materials development was performed in the laboratories along with some fuel testing in reactors.

Fuel materials compatibility testing and clad compatibility testing were performed. A number of full-size fuel elements were fabricated and then tested up to 12,000 hours of operation. There were in-reactor radiation tests, and finally, critical experiments at GE.

At ANL, (Figure 3) the program focused on rocket propulsion areas and there were two specific designs that were prepared during that time period. For the 2,000 megawatt reference engine, cycle studies and core analysis studies and design studies were performed. Fuel materials work was performed in the laboratory for tungsten cermets with uranium oxide fuel. The assemblies were clad with tungsten. ANL developed a stabilized UO_2 fuel and investigated several different cladding techniques. ANL fabricated fuel elements and tested them statically as well as dynamically and then they also performed critical experiments.

Figure 4 is a comparison of the requirements for the NASA workshop here versus the ANL study which was done in 1960. The engine thrust was around 100,000 pounds. It was a single engine. Reactor power was 2,000 megawatts thermal. It was operating in a single mode. The engine thrust-to-weight turned out to be a factor of five. Specific impulse was 832 seconds. The nozzle expansion ratio was 50-to-1 as opposed to 100-to-1.

The system was designed for about ten hours of operation. It could withstand multiple startups and basically could meet the other goals shown in Figure 4.

Figure 5 illustrates the engine itself. It has a bleed cycle where the coolant comes from the source and then flows down through the nozzle, cooling the nozzle, and then flows through the reflector control drum segments and back into the entrance of the reactor and through the reactor.

Figure 6 shows some of the characteristics of the engine. This is a fast reactor; 2,000 megawatts thermal. It provides 832 seconds specific impulse, 100,000 pounds thrust, and operating time is about ten hours. It can restart up to about 40 cycles and uses liquid hydrogen as propellant with a flow rate of 120 pounds per second. The fuel was composed of 60 percent UO_2 and 40 percent by volume of tungsten, fully enriched fuel. The core itself is about 34 inches long and about 24 inches in diameter. There were 163 hexagonal shaped elements, 1.87 inches across the flats.

Figure 7 shows the core design with hexagonal shaped fuel elements that are suspended from a plate at the entrance of the reactor. There are 163 of these elements, which use a rather simple design, with only one support point at the inlet end. The reactor is controlled by beryllium control drums (Figure 7)

Figure 8 shows the fuel element. It consists of a hexagonal-shaped tungsten matrix with the fuel particles blended in with the tungsten and then compressed. There are coolant holes provided that allow the coolant to flow through the matrix.

The cermet is clad with a tungsten/rhenium cladding on the outside surface and also the inside of the tubes. This particular design uses a fuel segment region with beryllium oxide reflector region and an inlet end fuel support point.

The operating condition for the engine at full power produces an Isp of 832 seconds with 100,000 pounds thrust. The reactor outlet temperature is about 4,500 degrees Rankine.

One of the major program tasks involved developing fuel fabrication techniques for the cermet reactor. Figure 9 shows the process that was developed, basically starting with fuel compacts, which contained a dispersion of UO_2 fuel within a tungsten matrix. The compacts are combined with header plates that are drilled.

The fuel compacts were stacked. Then the tubes were slid through the fuel compacts and into the header. The header ends were welded. An outer hexagonal cladding unit was prepared and installed over the assembly. The cladding was welded to the header. Then the entire system was bonded so that the outer cladding and inner cladding would be bonded to the tungsten cermet. (Figure 10). These elements were very successful, very high quality, providing a very high-integrity fuel design.

Figure 11 shows an example of a fuel element that was built at ANL. It has 331 flow passages and it is designed for the nuclear rocket. It is an example of what can be done with the cermet fuel.

At GE, the fuel was tested extensively, both in-core and out-of-core as shown in Figure 12. 60 percent UO_2 and 40 percent tungsten cermet clad with the tungsten/rhenium cladding was used. The program was designed to demonstrate structural integrity of the fuel assemblies, high temperature performance, retention of fission products, compatibility of fuels and materials at high temperatures, dimensional stability and development of the manufacturing process.

All of these goals were achieved under the 710 program. Most of the testing was done at lower temperatures than we would expect to see for the nuclear rocket program, but ANL did additional tests on similar kinds of elements at higher temperatures.

There were some tests run at 2800 K, ex-pile, and these were run steady-state as well as at thermal cycles. The results demonstrated that the fuel was very forgiving under many thermal cycles. There were no breeches in the cladding.

Figure 13 shows the fuel development test program at ANL. They started off with some very simple wafers where they developed various coatings and claddings. In some cases the elements were clad, and in other cases they were vapor-coated with tungsten or tungsten uranium. They also developed a technique of coating the fuel particles before they were put into the matrix and then they would be clad, so you have basically a double barrier (Figure 14).

A VOICE: The particle would be coated with tungsten?

MR. KRUGER: Yes, the UO_2 coated with tungsten which was then clad.

These elements were run in a high temperature furnace (Figure 13). They were all run at about 2,500 degrees centigrade. They were then evaluated. The seven hole samples were fabricated and run through a temperature cycle furnace and finally through a small flowing loop hydrogen test. The 331 hole sample was manufactured but they never did get to the testing program because the program was terminated prior to the testing.

Figure 15 shows work that was done by ANL to develop a stabilized version of the UO_2; What they found was by adding a certain percentage of gadolinium to the matrix, they could prevent loss of fuel from the UO_2. These tests here were run for cases where there was no cladding on the fuel sample. You can see they were run at 2,500 C up to maybe a hundred cycles or more. Very good stability was demonstrated under those conditions (Figure 16).

The transient test was run in the TREAT facility with the cermet fuel (Figure 17). These were run with very high surface temperatures up to 2,750 temperatures centigrade, and also at very high rates of temperature change, up to 4,500, 6,000 degrees C per second. Because of the limitation on the facility, these were not maintained at temperature for very long, but they were run for a number of thermal cycles. This gave very encouraging results that the cermet fuel can take very severe transients and not fail; no failures were noted under these tests.

The cermet fuel was also being considered for use in a Brayton cycle with operation up to a year, and a number of tests were run in-reactor. Figure 18 shows the results of those test programs. The cermet fuel reached a burn-up of about half a percent with no fission product release. If accommodation was provided in the fuel matrix for fission products, even higher burn-ups could be achieved.

Figure 19 indicates the technology development for cermet fuel. We need to reinstate the cermet fuel manufacturing and qualification program, and there are several key areas of design and development testing required. First, we need to establish the fuel form that will be required through some system analyses or system development studies. Once that has been established, we will propose fabricating some small fuel samples and then verifying the material compatibility at temperature with the fuel stabilizer and the cladding. Then we would run small samples at temperature, conduct some irradiation, and run transient tests on the reference fuel form to demonstrate its capability. Finally, we would fabricate full-size elements and run those in full-flow transient tests to demonstrate stability needed to withstand the testing environment. This would then lead to a full-size reactor qualification test (ground test).

Most of the materials work has been accomplished as a result of the large data base developed for materials in the 1960s for tungsten and tungsten/rhenium alloys (Figure 20). There will be some additional materials testing that will be required and we would suggest that rhenium be considered as a possible candidate for fuel cladding because of its weldability.

For the reactor component development test, we would take maximum advantage of NERVA technology (Figure 21). We suggest that ROVER technology be used for reflector control drive development testing because similar drive systems are used. Of course, some reactor flow hydraulic testing is needed. The core mechanical support design needs to be verified and tested. The preheat zone just outside the reactor core may need testing. A review of data from the existing critical assemblies is needed to determine if any additional critical tests would be needed.

We believe that a full system ground test is needed in order to qualify the system for flight. (Figure 22). Of course, stringent safety precautions are going to be needed to prevent environmental releases during the ground test. One of the features of the cermet fuel is its inherent capability to retain fission products. It offers a very positive containment with essentially a zero-release to the environment. The ground test requirements may not be quite as severe for the cermet fuel as for other concepts.

Figure 23 presents a reasonable, although fairly aggressive schedule. It shows about nine years from the time of start until the time to launch. It also shows the flight option being initiated in parallel with the ground test. The key activities that need to be started right away would be mission studies and concept definition studies to define the reactor system and the fuel form. That information then would be fed down into development testing for the fuel.

At the same time, facility studies must be initiated so that the facility preparation could begin, leading to the ground test. Parallel with other activities we would have technology support as well as safety analyses and a rather rigorous safety program.

We need to take advantage of the technology that already exists. Both the NERVA and ROVER system experience can be applied to the cermet fuel reactor. Test facilities, support systems, the effluent cleanup systems, test operations, and all lessons learned could certainly be applied to the cermet reactor.

Safety is a paramount consideration (Figure 24). The cermet fuel offers some very definite safety advantages. It's a high-strength, very rugged fuel form that can withstand thermal transients and repeated rapid thermal cycles. It offers a positive way to retain fission products with essentially zero release, either on the ground or in space. It also provides very high strength for safe reentry and burial in the event there would be a launch abort accident. The tungsten/rhenium materials provide inherent safety in the event of a water immersion accident.

In conclusion, the cermet fuel work conducted in the 1960's has demonstrated that we can have excellent thermal and mechanical performance. Thousands of hours of testing were performed on the cermet fuel, both at GE and ANL, including very rapid transients and some radiation performance history. We conclude that there are no feasibility issues with cermet fuel. What is needed is reactivation of existing technology and qualification testing of a specific fuel form. We also believe that this can be done at minimum development risk.

A VOICE: One, you didn't mention the mass. Two, you didn't discuss the limitations of the fuel form.

MR. KRUGER: We haven't really optimized the mass, because what I have presented to you here is a study that was done by ANL back in the 1960s. The thrust-to-mass ratio is approximately five, which gives you a ballpark number. The limitation on fuel is temperature.

We believe that the fuel temperature can approach 3,000 K. The maximum fuel temperature was running around 2,700-2,800 degrees kelvin in these studies; the melting point of UO_2.

A VOICE: What is the fuel analysis lifetime?

MR. KRUGER: It depends on the temperature you operate at, of course, but under the case I showed here, it could be hundreds of hours.

A VOICE: What is your base design fuel loading?

MR. KRUGER: How much UO_2? 635 kilograms UO_2.

A VOICE: If the UO_2 is contained within the tungsten, why is the UO_2 melting a limiting criteria?

MR. KRUGER: It wouldn't necessarily have to be, if we could assure it could be contained in the tungsten/clad matrix.

A VOICE: What about the possibility of a UO_2-thorium mixture. It has a much higher melting point.

MR. KRUGER: Yes, that's true. UO_2-thorium has a much higher melting point and that could be a possible alternative. That was being considered in the 710 program at GE but had not been fully tested or developed.

A VOICE: What is the temperature limit on the operation if we simply consider the tungsten?

MR. KRUGER: Tungsten could go to much, much higher temperatures. I don't have a limit on that, but tungsten could go to much higher temperatures.

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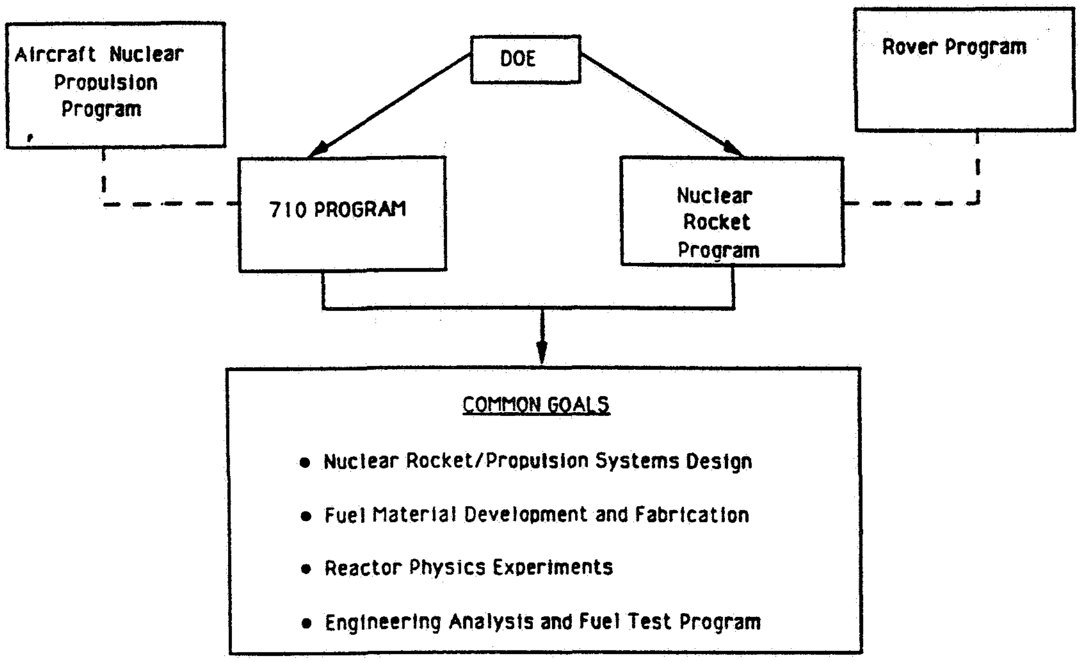
2. "710 High Temperature Gas Reactor Program Summary Report" GEMP-600 (six volumes) Nuclear Technology Department, Nuclear Energy Division, General Electric, Cincinnati, Ohio.

DIRECT NUCLEAR PROPULSION TECHNOLOGY DEVELOPMENT



Cermet Fuel Technology and Propulsion System Development Took Place During the Period 1962-1968

Figure 1

CERMET FUEL PROPULSION PROGRAMS IN THE 1960'S

710 PROGRAM AT GE

- Reactor Systems Design & Analysis
  - Liquid Metal
  - He
  - H_2
- 10,000 Hr Continuous Operation Design Life
- Nuclear Rocket
  - 30,000 - 200,000 lb Thrust
  - 10 Hr Full Power
  - 100-200 Restart
  - 850 - 870 Sec Specific Impulse
- Control Analysis
- Fuel + Materials Development
  - UO_2
  - W-Cermet
  - Mo Cermet
  - Clad
    - W-Re
    - W-Mo-Re
    - T-111
    - Mo-Re
  - Fuel/Materials Compatibility
  - Fuel/Clad Compatibility at 4700°F (2867K)
- Fuel Element Fabrication
  - Process Development
  - 19 and 37 hole full sized elements
- Fuel Element Testing
  - Non Nuclear Static/Dynamic
  - 3000° F (1922K)
  - Helium/Neon
  - Up to 12,000 hrs
- In-Reactor Irradiation Tests
  - Equivalent to 1 year operation
  - 2000 F (1367K)
  - Up to 5000 hrs in Reactor
- Critical Experiments
  - 9 Critical Experiment Configurations

Figure 2

CERMET FUEL PROPULSION PROGRAMS IN THE 1960'S

NUCLEAR ROCKET PROGRAM AT ANL

- Rocket Propulsion Design and Analysis
  - 2000 MW Reference Engine
  - 200 MW Alternate Engine

Elements

  - Cycle Studies
  - Core Design and Analysis
  - Control Studies
- Fuel and Materials Development
  - Fabrication Process Development
  - Fuel/Materials compatibility
  - UO_2
  - W Cermet

Assemblies

  - Clad

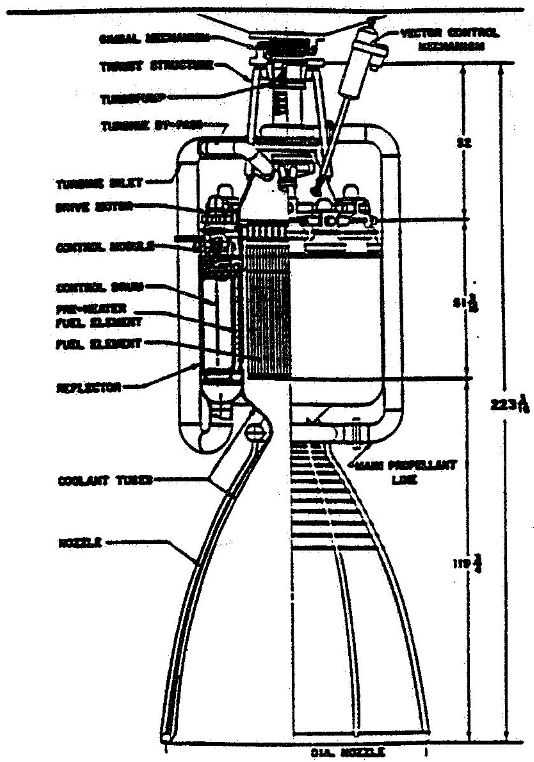
    W-Re
  - Stabilized UO_2
  - Pressure Bonded Cladding vs Vapor Deposited cladding
  - Material Property Testing
- Fuel Element Fabrication
  - Vapor Deposited Cladding and Pressure Bonded Full Size
- Fuel Element Testing
  - Non Nuclear Static/Dynamic
    - Up to 2600°C (2873K)
  - Reactor Dynamic Tests - Treat
    - Up to 2750°C (3023K)
    - 10,000° C/sec Transients
- Critical Experiments
  - Eight Critical Experiment

Figure 3

NUCLEAR THERMAL PROPULSION - REQUIREMENTS

REQUIREMENT PARAMETER:	UNITS:	BASELINE DESIGN - MANNED MARS	VARIATION FROM BASELINE:	CERMET CORE NTR CONCEPT (ANL STUDY)	CERMET CORE NTR VARIATION FROM BASELINE
Engine Availability	Year	2015	200-2017	Meet Schedule	Meet Schedule
Thrust Per Engine	klb(t)	75	25-250	100K	Design Accommodates Broad Range of Engine Thrust Requirements
Number of Engines	Number	1	Multiple	Single	Engine Concept is Feasible
Reactor Power (Thermal)	MW(t)	1500	600-5000	2000	No Limitation on Power Range
Dual Mode-Low Electric Power	kWe	0	25-50	Single Mode	Can Be Designed For Dual Mode High/Low Electric Power
Dual Mode-High Electric Power	MWe	0	1-5	Single Mode	
Engine Thrust/Weight	klb(F)/klb(m)	4	3-10	~5	Falls Within Range
Specific Impulse	Seconds	850	850-1200	832	Current Studies Indicate Range of SI 800-900
Nozzle Expansion Ratio	Ratio	100:1	100:1-500:1	50:1	Can Accommodate Greater Expansion Ratio
Propulsion Operating Time/Mission	Minutes	120	40-120	10 Hrs	Minutes to 10's - 100's of hours Can be Accommodated
Number of missions	Number	1	1-5	1	Can Accommodate Several Missions
Number of Startup Cyles/Lifetime	Number	6	1-30	Multiple	Can Accommodate Many Restart Cycle
Average Mission Duration	Days	434	270-600	Can Meet	No Limitations on Mission Duration
Reliability	Number	0.995	0.995-0.975	High Reliability	System Simplicity & High Strength of Cermet Fuel Provides High Reliability
Deployment Orbit	Kilometers	407	407-700	Can Meet	Can Meet Alternate Orbits
Maximum Crew Radiation Limits from Reactor Source	REM/Yr	5	0-5	Shielding Required Required	Optimized Shielding For Mission and Compact Core

Figure 4



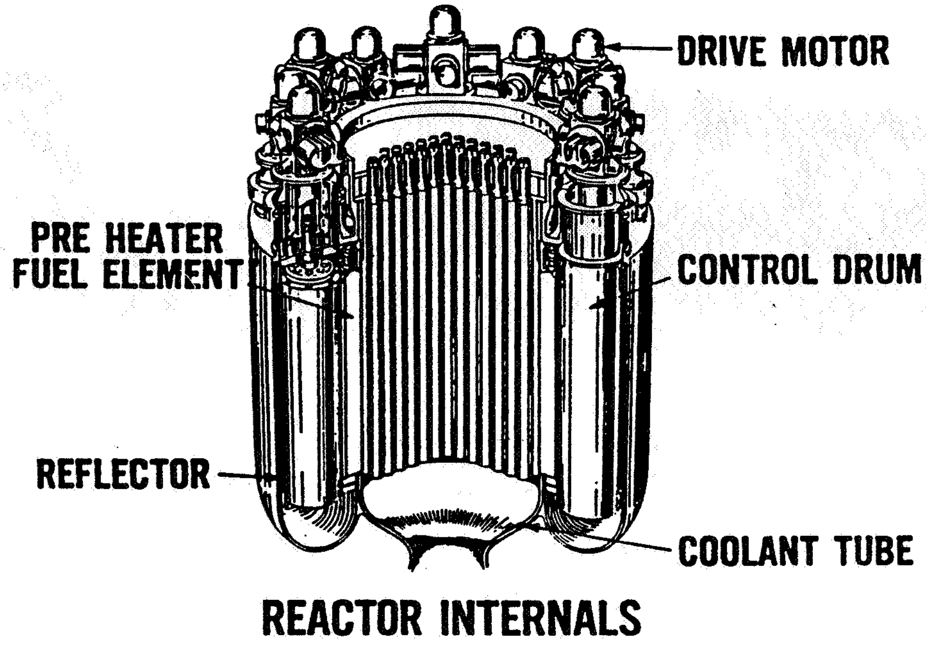
NUCLEAR THERMAL PROPULSION ENGINE
CERMET CORE 2000 Mwt

Figure 5

CERMET REACTOR FOR 2000 Mwt PROPULSION ENGINE

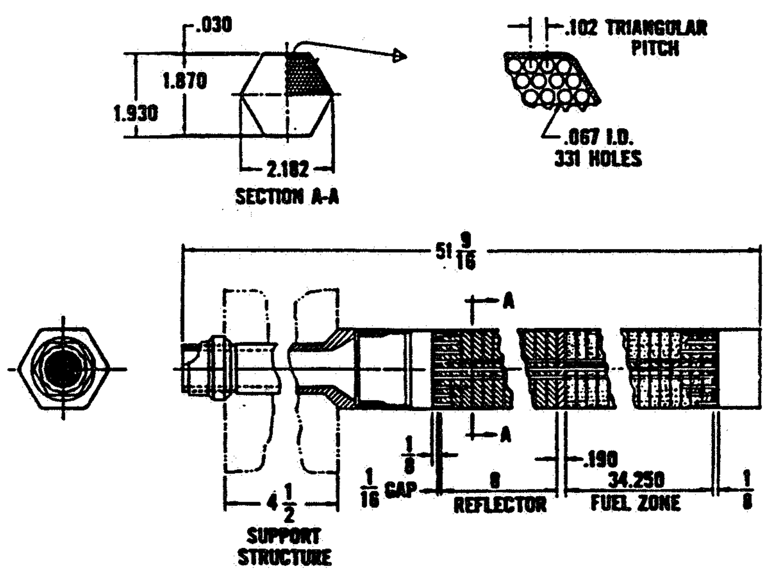
REACTOR ENGINE CHARACTERISTICS	
Reactor Type	Fast
Reactor Power	2000 Mwt
Specific Impulse	832 Sec
Thrust	~100,000 lb
Operating Time	Up to 10 hrs
Restart Capability	Up to 40
Propellant	Liquid Hydrogen
Flow Rate	120 lb/Sec
FUEL	
COMPOSITION	60 V/^{o} UO_2
MATRIX	40 V/^{o} W
U_{235} ENRICHMENT	93%
FUEL ELEMENT	
Length Active	34.25 in
Across Flats	1.87 in
No Assemblies	163
Fuel Clad	W-25 Re
Peak Fuel Temp	4912°R (2728K)

Figure 6



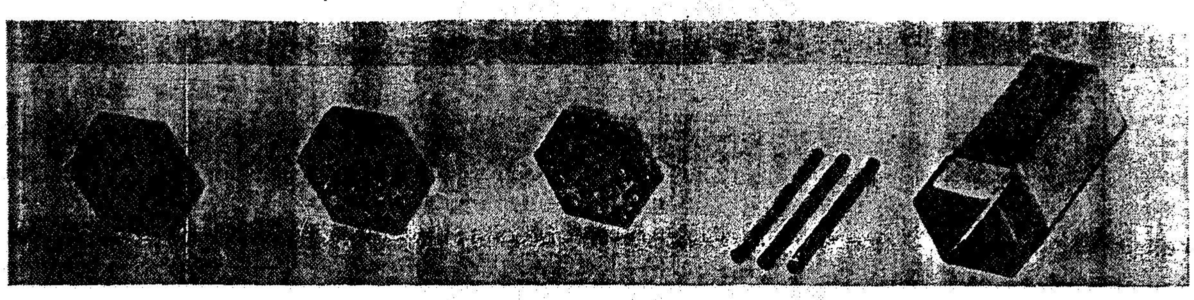
CERMET REACTOR CONCEPT FOR
2000 Mwt PROPULSION ENGINE

Figure 7



FUEL ELEMENT

Figure 8



FACE-GROUND HEADER BLANK
- Ultrasonic inspect.
- Cut rough hex
- Surface grind – opposite faces flat and parallel within 0.0002 inch

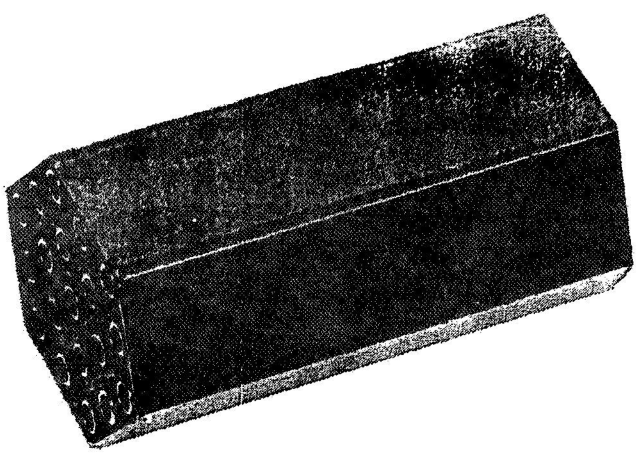
DRILLED HEADER
- C-2 carbide drills
- Torque controlled feed rate
- 19 holes
- Diameter tolerance: ± 0.001 inch
- Position tolerance: 0.001 inch radius from true position

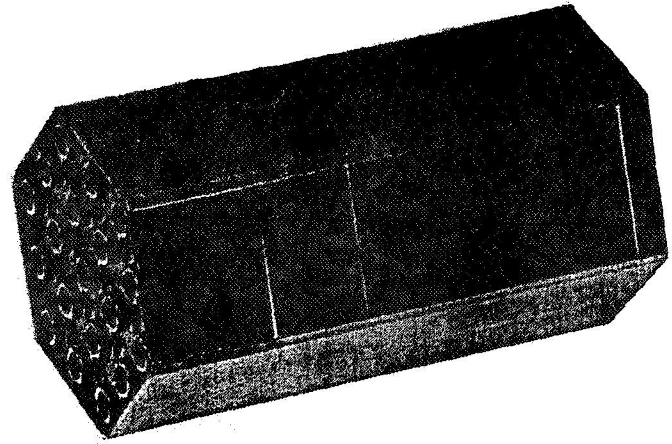
EMBOSSED HEADER
- Electric discharge machine (EDM)
- Gentrode 10 (graphite) electrode
- Chemically clean
- Hydrogen clean
- Vacuum clean (10^{-5} torr)

COOLANT TUBES
- Eddy current inspect
- Cut to rough length
- Grind to final length
- Re-eddy current inspect
- Chemically clean
- Vacuum clean (10^{-5} torr)

HEXAGONAL OUTER CLADDING
- Ultrasonic inspect
- Shear and grind to width
- Form to half-hexes and grind edges
- Chem. clean and heat treat
- Electron beam weld seams and X-ray
- Isostatically press to size
- Grind to length and leak check
- Chem. and vacuum heat clean

Figure 9





AS-ASSEMBLED AND SEALED
- Stack fueled segments
- Insert tubes
- Insert segments into hexagonal cladding
- Install headers
- Inspect
- Electron-beam weld hexagonal cladding and tubes to headers
- Leak check by helium mass spectrometer

AS HOT-GAS PRESSURE BONDED
- Load into bonding apparatus
- Pressurize, purge, and backfill with inert atmosphere
- Bond at 3180°F and 10,000 psig in He for 1.5 hours
- Leak check by helium mass spectrometer
- Bond checks: OD cladding by resonance frequency and pulse echo; ID cladding by through transmission
- Dimensions, weight, volume approx. 96–97% theoretical density O/U ratio – 2.00

Figure 10

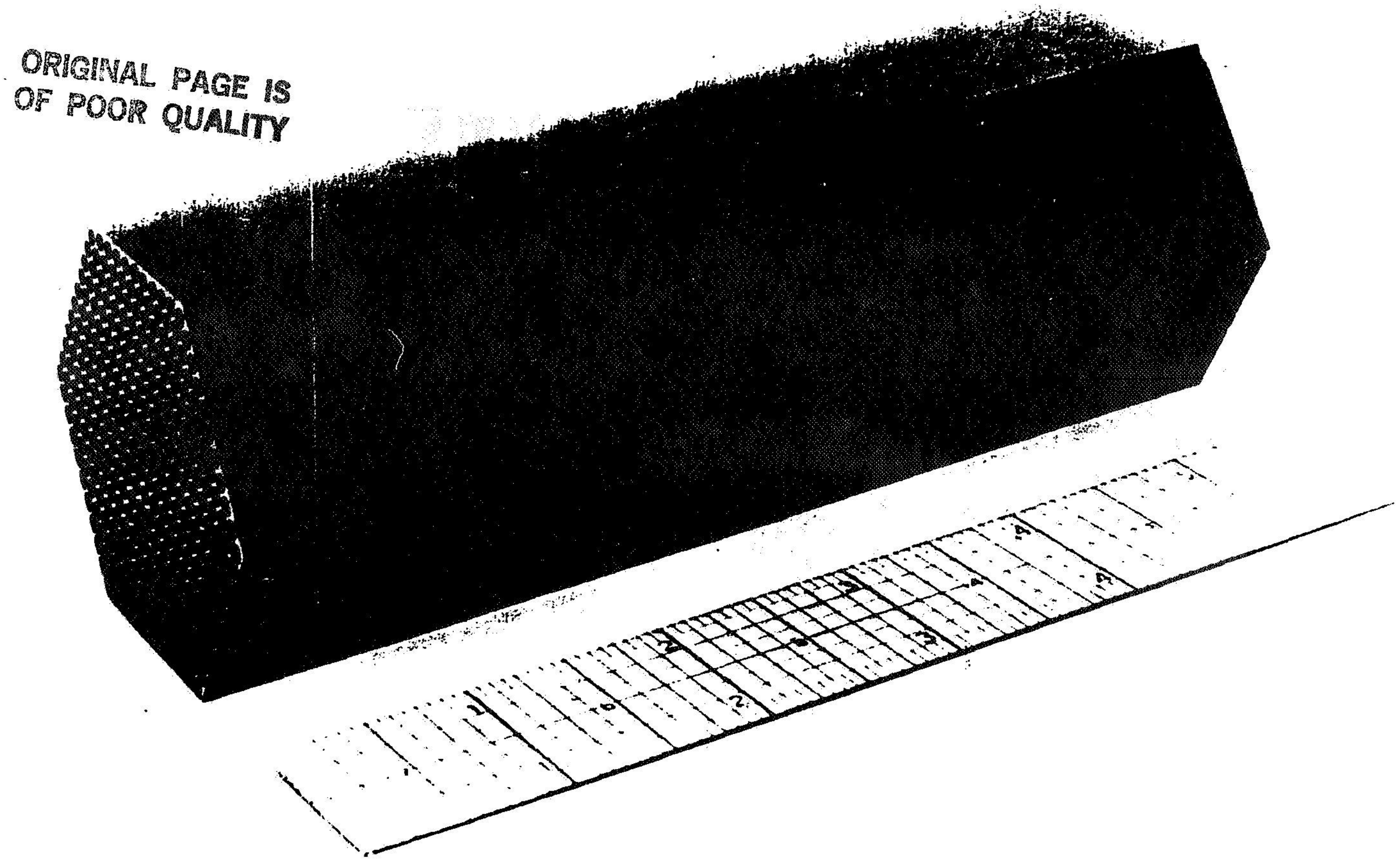
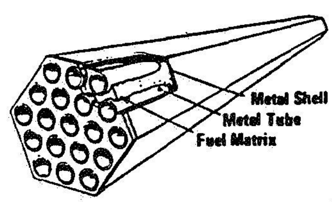
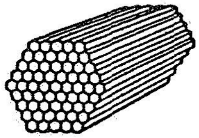


Figure 11

Fuel Element Sample, Large Nuclear Rocket



60 UO_2–40W CERMET
W–Re–MO CLAD



ACTIVE CORE ASSEMBLY

RESULTS OF CERMET FUEL TESTING - 710 Program

- STRUCTURAL INTEGRITY OF FUEL ELEMENT DEMONSTRATED UNDER STEADY STATE & TRANSIENTS
- HIGH TEMPERATURE PERFORMANCE ACHIEVED
- RETENTION OF FISSION PRODUCTS ACHIEVED
- FUEL AND MATERIALS COMPATIBILITY DEMONSTRATED
- DIMENSIONAL STABILITY DEMONSTRATED
- MANUFACTURING PROCESS DEVELOPMENT ACHIEVED

EXTENSIVE FUEL TESTING DATA BASE

HIGH TEMPERATURE EX-PILE STATIC/DYNAMIC TESTS

- 28 TEST ELEMENTS
- UP TO 90 THERMAL CYCLE RUNS/ ELEMENT BETWEEN 530K AND 1920K
- UP TO 12,000 HRS/ELEMENT

VERY HIGH TEMPERATURE DYNAMIC EX-PILE TESTS

- TO 2800K TEMPERATURE
- 193 THERMAL CYCLE RUNS
- 49 HRS TEST DURATION

BURST TRANSIENT TESTS IN TREAT

- SUCCESSIVE BURSTS TO 3020K WITH COOLDOWN
- EIGHT SPECIMENS
- UP TO 6 CYCLES EACH

HIGH TEMPERATURE IN-PILE QUALIFICATION TESTS – 710 PROGRAM

- 21 TEST ELEMENTS
- UP TO 1890K
- 10,000 HRS MAX. DURATION
- 0.5 AT % BU ACHIEVED - Porosity Requirements For Long Life Established
- UP TO 60 THERMAL CYCLES/ELEMENT

86-206 01

Figure 12

FUEL DEVELOPMENT TEST SEQUENCE

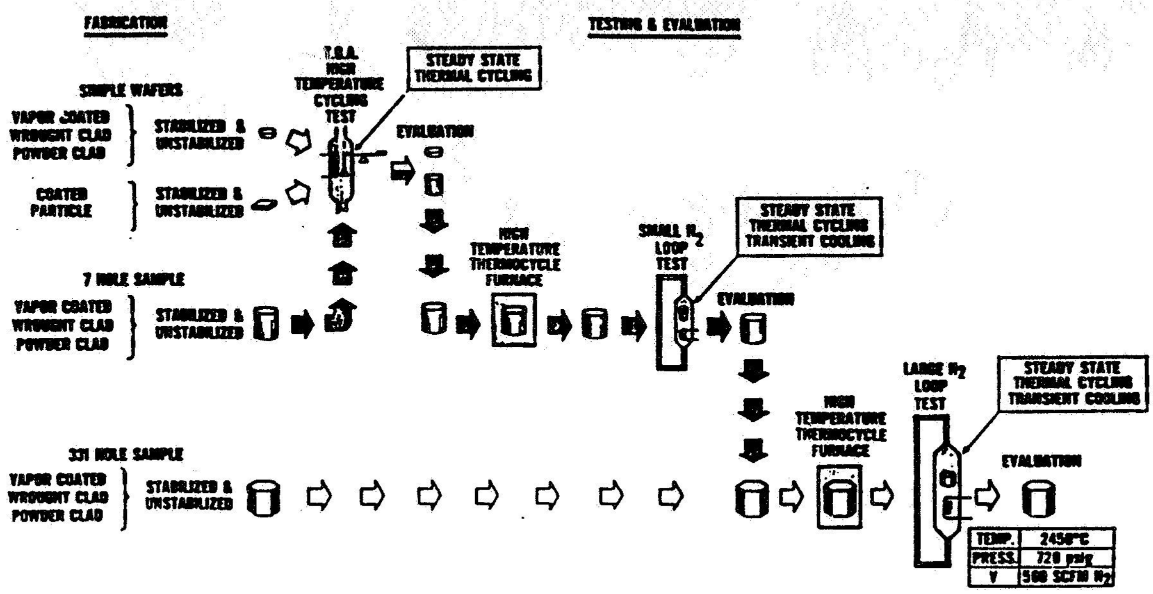


Figure 13

VARIOUS CLADDING TECHNIQUES

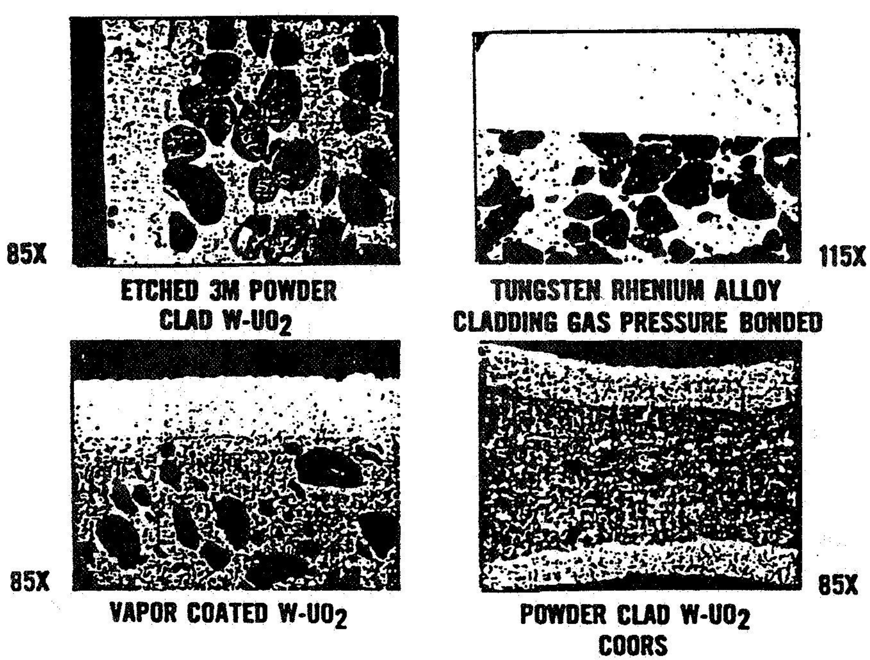
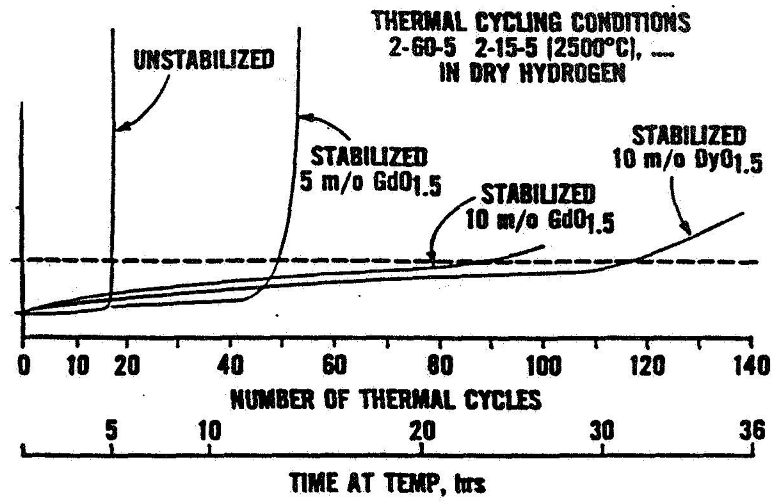


Figure 14

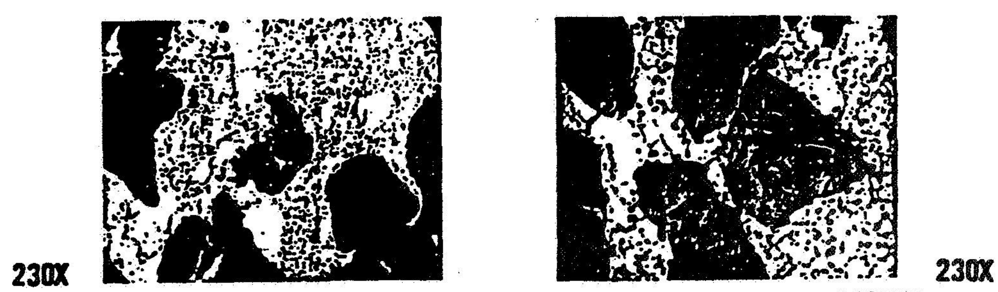
CERMET FUEL THERMAL CYCLING AT 2500°C

FUEL LOSS COMPARISON



Tests of Cermet Fuel with Ga Stabilizer Demonstrate Stability at Temperature and with Thermal Cycling

Figure 15



REJECTED U IN W - UO_2 (UNSTABILIZED)

Gd_2O_3 STABILIZED UO_2 IN W

EFFECT OF Gd_2O_3 STABILIZER

Figure 16

TRANSIENT TREAT TEST RESULTS

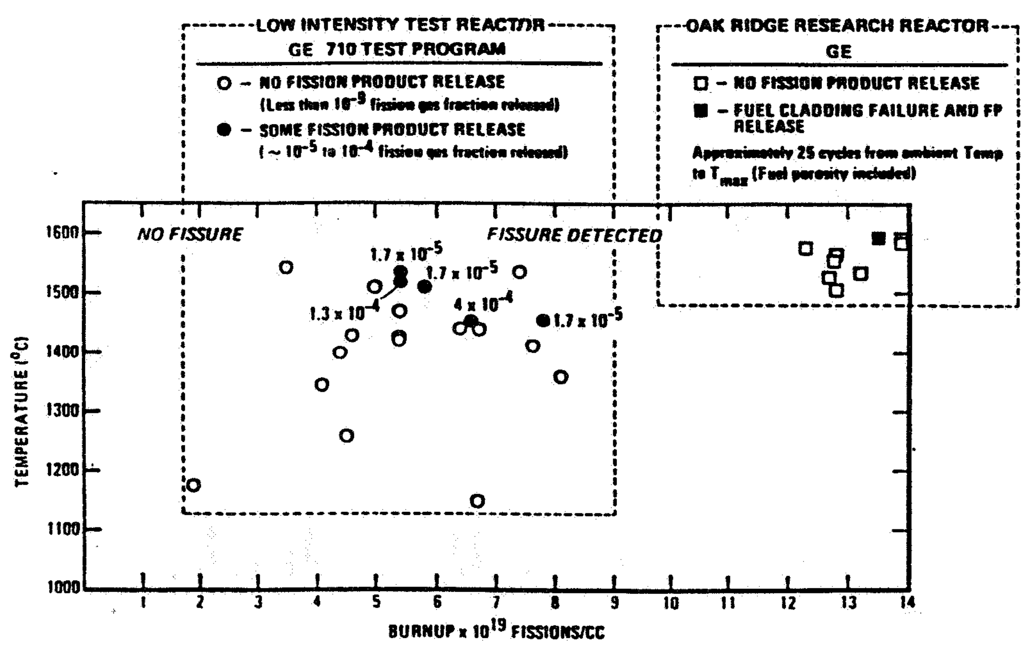
SAMPLE NO.	TRANSIENT DURATION (SEC)	REACTOR INTEGRATED (MW-SEC)	MAXIMUM RECORDED SURFACE (°C/SEC)	MAXIMUM RECORDED SURFACE TEMPERATURE (°C)
1	0.43	164	1,700	800
1	0.3	284	3,900	1,460
2	0.3	377	5,600	1,790
3	0.2	487	8,000	2,200
4	2.1(A)	332	800	1,460
5	0.2	540	2,000	2,600
6	3.0(B)	495	1,400	2,050
7(B)	0.2	523	4,500	2,750
8(C)	0.2	532	6,000	2,750

(A) "FLAT TOP" TRANSIENT
(B) SAMPLE GIVEN TWO ADDITIONAL TRANSIENTS OF SAME SEVERITY
(C) SAMPLE GIVEN FIVE ADDITIONAL TRANSIENTS OF SAME SEVERITY

NO FUEL FAILURES WHEN SUBJECTED TO SEVERE THERMAL TRANSIENTS

Figure 17

GE CERMET FUEL TEST PROGRAM BRAYTON CYCLE QUALIFICATION



ALL THERMAL CYCLING PINS THERMALLY CYCLED FROM 0.5 T_{max} to T_{max} DURING IRRADIATIONS – CYCLES RANGED FROM 33 to 88

Figure 18

CERMET FUEL KEY TECHNOLOGY DEVELOPMENT

- Reinstate Cermet Fuel Manufacturing Technology and Qualify the Specific Fuel Form for NTP
- Key Areas of Design/Development and Qualification Testing
  - Establish Fuel Form Requirements Through System Studies
  - Fabricate Small Fuel Samples for Testing and Select Reference Fuel Form
    - .. Verify Material Compatibility
    - .. Verify Fuel Stabilizer
    - .. Verify Cladding Approach
  - Conduct Irradiation/Transient Testing on Reference Fuel Form
  - Fabricate Full Size Fuel Assemblies
  - Perform Full Flow Transient Tests of Full Size Assemblies
- Conduct a Full Size Reactor Qualification Test (Ground Test)

Figure 19

MATERIALS DEVELOPMENT TASKS:

- The Fundamental Materials Database Was Developed for W, W/Re Materials in the 1960's
- Limited Materials Property Testing May be Required to Verify the Materials Database
- Rhenium Should be Considered a Possible Candidate for the Fuel Cermet Cladding to Provide Improved Weldability of the Clad Material

Figure 20

REACTOR COMPONENT DEVELOPMENT TASKS

- Utilize Modified NERVA Technology for Reflector Control Drive Development and Testing
- Reactor Hydraulic Flow Testing
- Reactor Core Mechanical Support Development and Testing
- Reactor Pre-Heat Zone Fuel Element Thermal/Hydraulic Testing
- Review Data from Existing Critical Assemblies to Determine if Additional Criticals are Required

Figure 21

CERMET FUEL PROPULSION GROUND TEST

- A Full System Ground Test is Necessary to Qualify the Cermet Fuel Propulsion System for Flight
- Stringent Safety Precautions and Environmental Release Requirements are Anticipated
- Cermet Fuel Offers a Positive Containment With Essentially Zero Release to Environment
- Ground Test Containment/Confinement May be Less Stringent Than for Alternate Concepts

Figure 22

PROGRAM SCHEDULE AND TASK SUMMARY

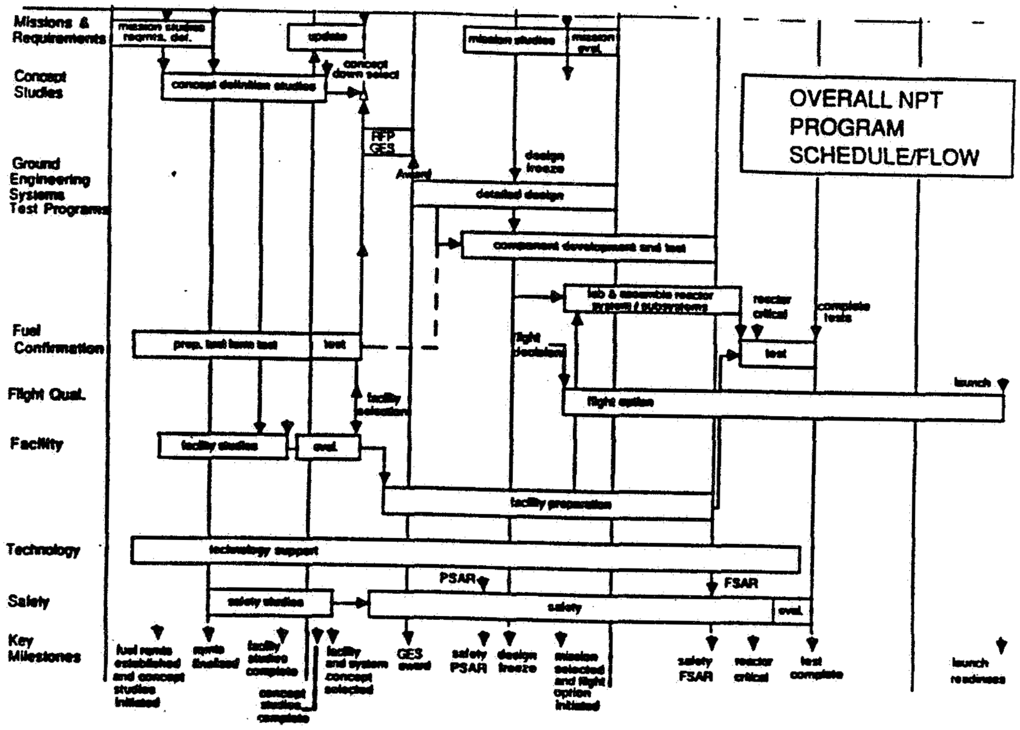


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Figure 23

CERMET FUEL REACTOR

SAFETY FEATURES

- Cermet Fuel is a High Strength, Rugged Fuel Form Which Can Withstand High Temperatures and Repeated Rapid Thermal Cycles
- Cermet Fuel Offers Positive Fuel Retention With Essentially Zero Fission Product Release to Environment
- Cermet Fuels High Strength Provides for Safe Re-Entry and Burial Configuration in the Event of a Launch Abort Accident
- Cermet Fuel Materials (W, Re) Provide Inherent Safety in Event of Water Immersion Accident

Figure 24

N92-11097

HYBRID PROPULSION SYSTEMS FOR SPACE EXPLORATION MISSIONS

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In the previous two presentations, you heard some very specific dual mode operations of the propulsion systems, which were referred to as hybrid systems. We felt that we should take a little broader look at the hybrid system, and give much broader top level characteristics of the various possibilities of combining the propulsion systems, and look at the constraints and the advantages.

Some information was presented on the NTP system in the previous two presentations, but since there was no information presented on the hybrid NEP system in the JPL conference, we want to provide a little additional basis for evaluation of the different concepts. I think it will be useful to see what the different technologies can bring, in terms of synergistic benefits, with respect to the other technologies, when you combine them together.

At the top of the chart (Figure 1) we have the chemical technology. Its' technology obviously is at hand, and it's a cliche to repeat the fact that it's limited in its performance by the specific impulse that it can provide. The next stage is a nuclear thermal, which has many advantages. First of all, it will improve upon the specific impulse problem that you have with the chemical system. It does have some negative aspects. It requires additional electrical power, and it has a requirement that no one has answered yet: how much hydrogen propellant you would need and how you would accommodate that and how much volume is associated with it. Then there is the question of our ability to ground test such a system.

Next there is a nuclear electric system, which has several positive points. The negative point obviously is the longer trip time, the associated long duration in the Van Allen Belt part of the orbit. So, since there are negative aspects with all of these systems, it benefits us to see if we can combine all these three and find the best hybrid system.

So we looked at all the possible combinations (Figure 2). First you can combine nuclear thermal with chemical, both being high thrust systems. You don't expect a tremendous amount of performance improvement there. Nevertheless, you will need some chemical propulsion for orbit capture, and definitely for altitude control, maneuvering and so forth, in addition to your NTP propulsion. Next is combining NEP with chemical. Again your performance could be limited depending on what your mass ratio requirement is, and then it could be limited in the chemical Isp that you can obtain in escaping Earth orbit. It also will be helpful, however, in achieving the orbit capture at Mars and on the

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return trip.

The NTP/NTP is basically a dual mode operation which you heard described earlier, and it has several plus points. But it makes the system complex, so you lose one of the key benefits of simplicity, and then also you have to account for the associated structural fraction penalty -- this being a high thrust system. By NTP/NTP I mean that you operate in a dual mode to generate high thrust and low thrust with one reactor. It's a matter of nomenclature.

The next one is NTP/NEP or NEP/NTP depending on what your inclinations are. One could be beneficial to the other, combining the high and the low thrust operation. High thrust phase is used for rapid Earth departure and the low thrust phase is used for the rest of the trip to reduce the trip time. One immediate drawback here is that two independent reactor technologies are required.

Finally, to compensate for NTP/NTP I also have NEP/NEP, wherein you would use a combined high and low thrust operation, but the same single reactor. It retains the Isp benefits of NEP throughout the mission. You operate at high thrust during departure from Earth orbit and low thrust subsequently. This approach has a single reactor technology, and there are some constraints associated with it that will be discussed later.

Figure 3 is basically the high and low thrust profile mission, wherein you rapidly come out of Earth orbit using the high thrust and spiral in to Mars.

Figure 4 shows the impact of the thrust-to-weight ratio, for example, on the velocity requirement. You can see in the thrust-to-weight ratio that at lower orbits you pay additional penalty because of the Earth gravity. In addition to providing that additional Delta V, you also have lower acceleration, which results in longer trip time. On the right, going from LEO to GEO orbit you have Delta-V versus thrust-to-weight ratios. In the range to the left, the Delta-V requirement is nearly constant. However, as you increase thrust-to-weight ratio, you can reduce your trip time going from LEO to GEO. When you consider just going from the LEO to, let's say, outside the Van Allen Belt, you can significantly cut down the trip time.

Now, to look at some of the concepts that were generated combining the high and low thrust, consider the earlier SNAP concepts. You can consider this in many different ways. This could be a dual mode operation, wherein you have an NTP system combined with a nuclear electric power producing system. It's not for nuclear electric propulsion, but is combined with an electric generator to produce utility power. It could use the same reactor or it could use another reactor to produce power. For example, you can replace the power generation part with the SP-100 to deliver 50 kilowatts, or whatever the power requirement is to the crew module. That is another item that hasn't been discussed yet, but I think it is an important item in the NTP system, namely, how do you provide this power that would be needed for the crew during the multi-year time of their living on the spaceship. In another concept, high thrust NTP chemical propulsion system is combined with nuclear electric propulsion. After coming out of the Earth orbit, the high thrust part is ejected and the rest of the spacecraft goes on its trip to Mars. Figure 5 shows a combination of NTP and NEP. In this variation the nuclear electric propulsion part of the system is boosted by the high thrust NTP. After it has come out of the Earth orbit, the system is deployed as shown in the figure; spacecraft panels are extended and the high thrust part is ejected. The whole system can also be given an artificial gravity by rotating the spacecraft. In the variation shown in Figure 6, the high thrust part is achieved by chemical propulsion. Figure 7 is a little better picture of the same spacecraft showing a deployed configuration. In this case the high thrust system is not discarded, but is available throughout the mission.

Finally, you have an NEP/NEP configuration (Figure 8) wherein the same reactor is used to generate the high- and low-thrust propulsion, using the same electrical output. With this you have a single reactor, and you have a low thrust engine, for example, an ion propulsion engine, combined with high thrust MPD thrusters to provide a combination of thrusts.

There are several advantages of the system (Figure 9). It can significantly reduce the duration in the Van Allen Belt and it avoids the need for the crew rendezvous in the high Earth orbit if you can do the mission in a single spacecraft. There is significant reduction in the power level to achieve the trip time. I think this is an important factor in the NEP system, because it relates to system reliability, launch and assembly constraints, and it avoids the need for the development of two independent reactor technologies. It's better to have a single reactor that achieves the same goal.

We went through several preliminary analyses, which I will discuss in another session. But to summarize, one particular case using NEP/NEP for example, to go from NSO, nuclear safe orbit, to approximately 10,000 kilometers using 1200 Isp (Figure 10), you need about 28 days. Obviously this probably is not short enough, but you can trade that off against the power requirement, and also the amount of shielding that you will need.

At the moment we don't have any definition on the amount of shielding that we will need for the crew protection from solar flare, for example. It may turn out that the shielding requirement for the solar flare may overshadow the shielding requirement that you need for the Van Allen Belt, so this trip time may not be too far out.

Basically the bottom line here is that you need a total mass of 650 metric tons, and that compares very well with the reference design. Also, although there may be differences in the launch dates and trajectories, it is comparable with the 680 metric tons that you need for the NTR system (Figure 11).

So, looking at the features of the hybrid propulsion system that affect the SEI mission (Figure 12), it pays to follow the high thrust with a low thrust when the leftover Delta V that you have to work with after the high thrust is of the same order or greater than the high thrust Delta V. This definitely is the case when you are escaping the Van Allen Belt radiation. You are not escaping the entire Earth orbit but you are simply trying to get out of the Van Allen Belt. For comparable initial mass in Earth orbit, the NEP/NEP hybrid system can substantially cut down the Earth escape time. For comparable trip-time, the power requirement is substantially lower than for the pure NEP system. Alternatively, for the same power level, trip time can be shorter. As another important consideration, when you combine the high thrust and the low thrust system, the structural part of the spacecraft becomes very critical. Figure 13 shows the effect of structure fraction on payload fraction. It is seen that the payload fraction drops off rapidly as the structure fraction increases for a given Delta V. This is important in a purely high thrust system. You must design the spacecraft for the high thrust, which may have a very hefty structural requirement. So it's important to have a graceful structural integration between high and the low thrust requirements, for example, by not going at a wide range of Isp, but keeping it closer together. A high-thrust phase could then be available at any point in the trajectory to achieve mission resiliency.

Figure 14 summarizes the status and need for hybrid system technology. It is concluded that hybrid systems do offer many advantages, and I think they should be considered, should be looked at much more closely, and should be compared with the other innovative technologies that we are looking at.

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Propulsion System Technology

Chemical
- Technology In Hand
- Specific Impulse Limits Performance
- Large Amount of LO_2 and LH_2 Must be Transported and Stored or Manufactured at LEO

Nuclear Thermal
- NERVA ReactorTechnology Ground Tested In Early 70's
- Specific Impulse Double That of Chemical
- Requires Large Amount of LH_2 to be Transported To LEO
- Requires Additional Electrical Power Source
- Requires Technology Revitalization - Ability For Future Ground Testing To Be Established

Nuclear Electric
- Electric Propulsion Engine Development In Embryo Stages
- Space Reactor Power Technology Validation in Progress
- High Specific Impulse Promises Heavy Cargo Delivery Capability But Results In Longer Trip Time
- Long Duration Van Allen Belt Exposure
- Relatively Easier Storage and Transportation of Xenon Propellant
- Design Modulatiry/Resiliency to Meet Broad Range of Propulsion Requirements
- Commonality With Proposed Lunar Mission Requirements

Hybrid
- Development Paced as Above
- Potential to Overcome Shortcomings of Above

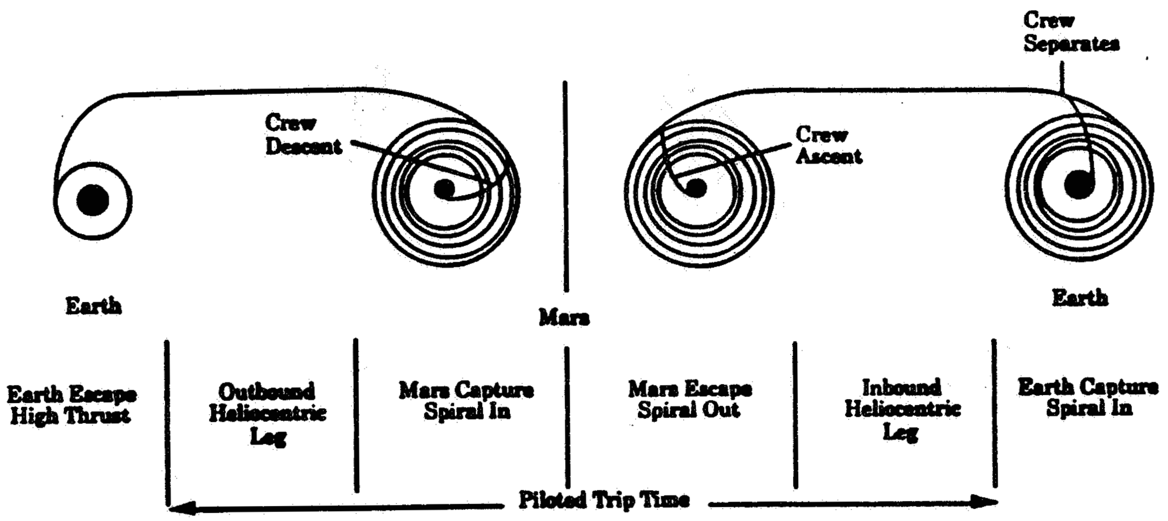
Figure 1

Characteristics Of Hybrid Propulsion Systems

NTP/Chemical	• Very Limited Performance Improvement Can Be Expected • May Be Necessary For Short Burn Orbit Capture And Trim Control
NEP/Chemical	• Usefullness Depends On Mission Applications - Limited By Chemical ISP And Required Mass Ratio • Helpful In Orbit Capture
NTP/NTP	• Dual Mode Operation - Electric And Thermal • Available Throughout Mission • Substantially More Complex And Expensive
NTP/NEP or NEP/NTP	• Combined High/Low Thrust Operation • High Thrust Phase Used For Rapid Earth Departure • Low Thrust Phase Used During Otherwise Coasting Period • Two Independent Reactor Technologies
NEP/NEP	• Combined High/Low Thrust Operation • Retains Benefit Of NEP Throughout Mission • Single Reactor Technology

Figure 2

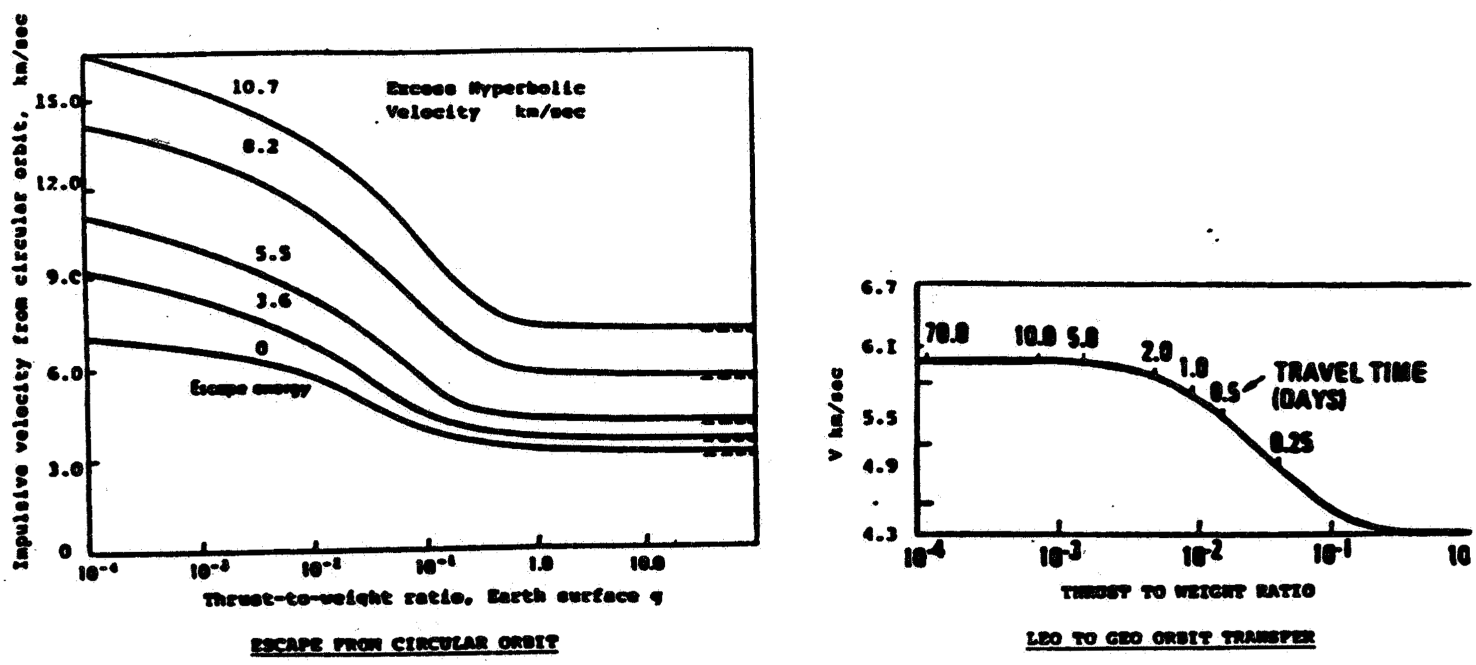
Typical Combined High/Low Thrust Profile



HIGH THRUST STAGE JETTISONED AFTER TRANS-MARS INJECTION BURN
NEP SYSTEM USED FOR REST OF MISSION

Figure 3

Impact Of Thrust To Weight Ratio On Delta V Requirements



Higher Thrust To Weight Ratio Can Assist Near Earth Orbit Transfer And Escape

Figure 4

NTP/NEP CONFIGURATION WITH ARTIFICIAL GRAVITY

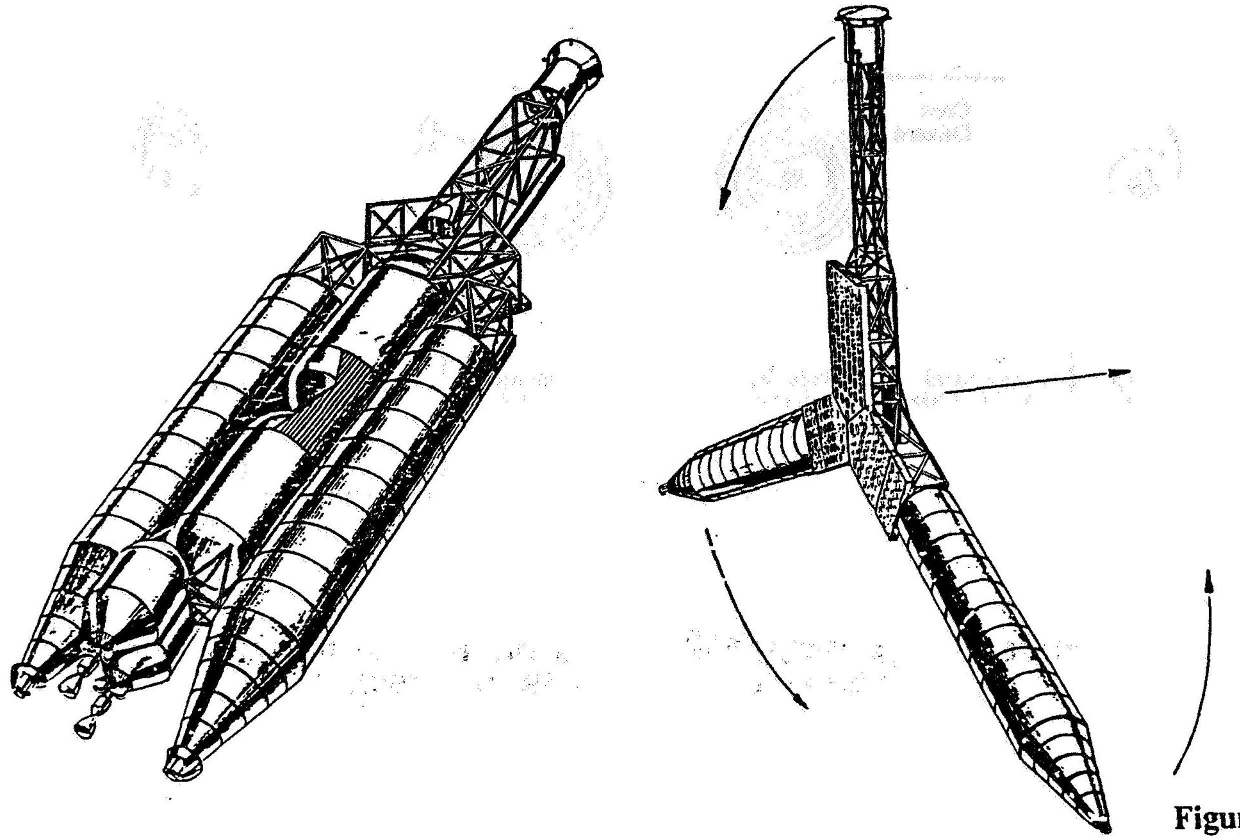


Figure 5

A Conceptual Design of Hybrid NEP/Chemical Propulsion System

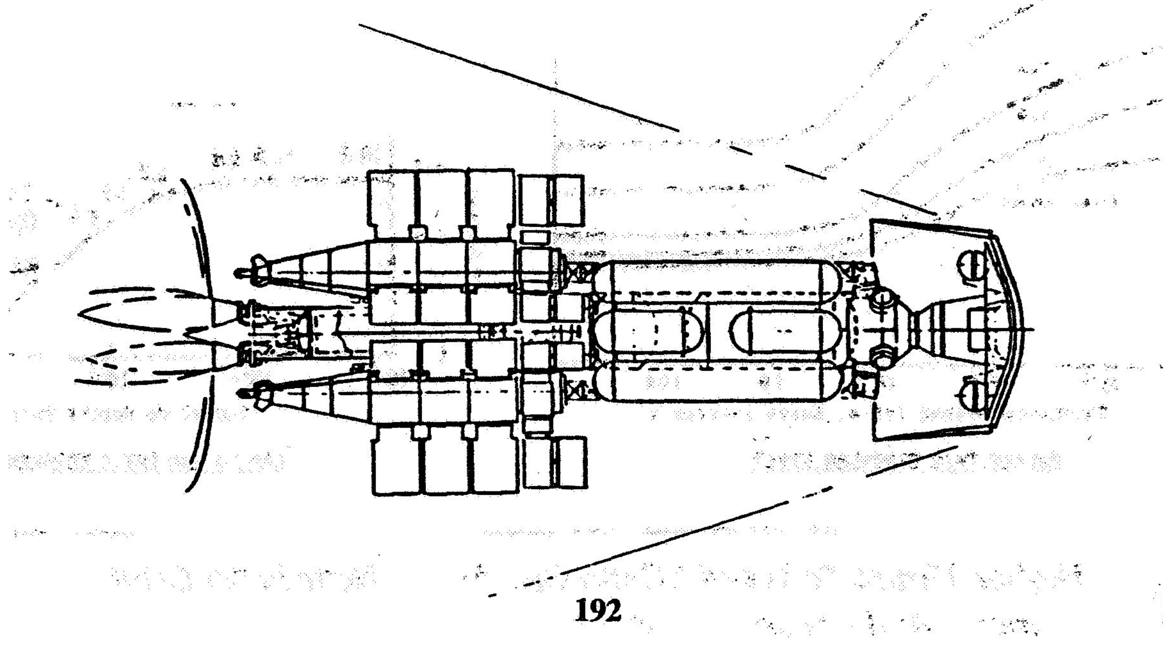


Figure 6

A Conceptual Design of Hybrid NEP/Chemical Propulsion System

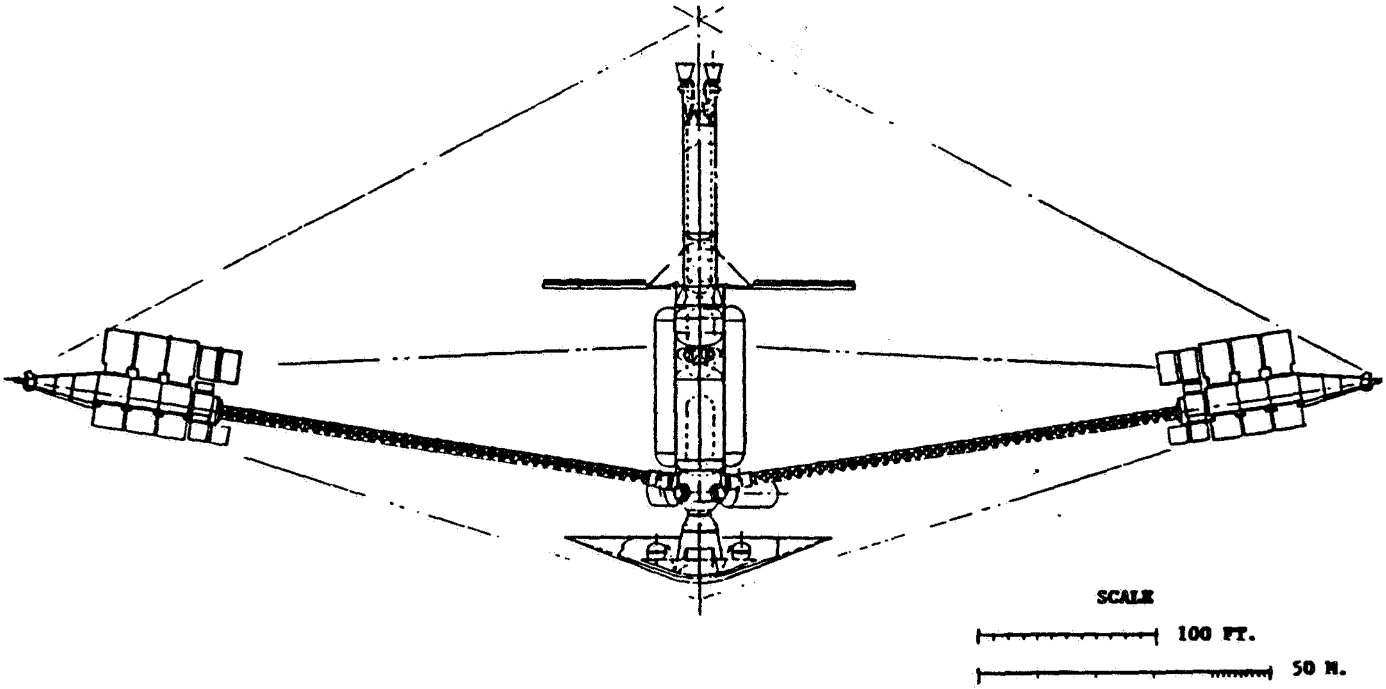


Figure 7

A Concept Of Hybrid NEP/NEP System

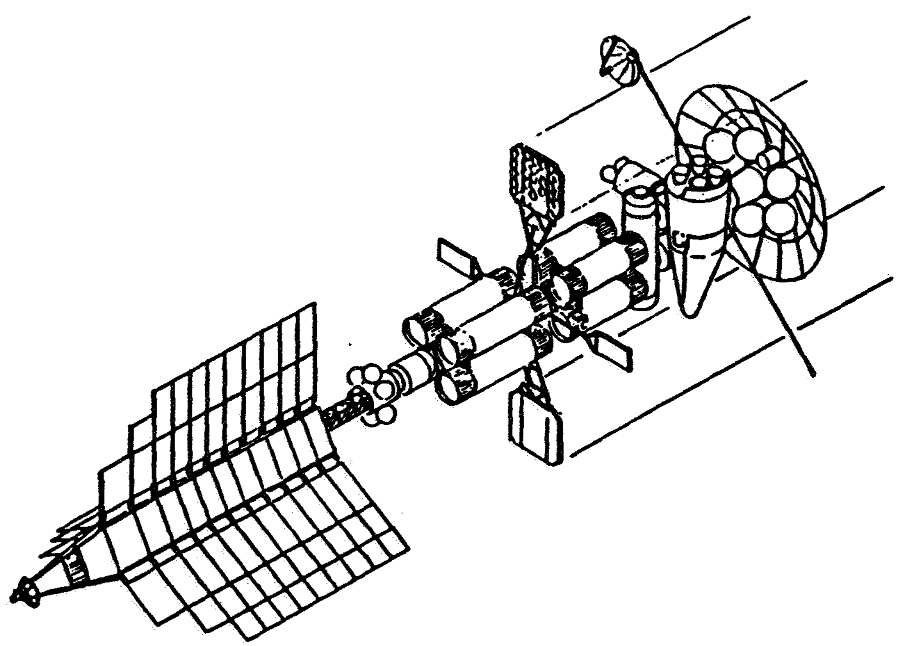


Figure 8

Combined High/Low Thrust NEP System

- *Use ARC Jet/MPD Engines For Initial High Thrust Stage From NSO to 10000 km Orbit*
- *Use Ion Engines For The Remaining Low Thrust Leg Of The Journey To Mars*
- *Use Low Thrust Spiral And Crew Separation For Return Earth Capture*

Advantages

- *Significantly Reduce The Duration In Van Allen Belt*
- *Avoids Need For Crew Rendezvous In HEO*
- *Significant Reduction In Power Level To Achieve TripTime*
- *Lower Power Level - Essential To Meeting Overall System Reliability And Launch And Assembly Constraints*
- *Avoids Need For The Development Of Two Independent Reactor Technologies*

Figure 9

Hybrid NEP/NEP System Characteristics

Trajectory Phase	Thruster	ISP(Sec)	One Way Trip Time (Days)	Power (MWe)
NSO To ~ 10,000 km	Arc Jet/MPD(NH3)	1200	28	10
10,000 km To Mars Orbit	ION (Xenon)	5000	150	10

Mass Estimate		
Delivered Payload Mass	Mt	125
NEPS (Dry) (Includes High And Low Thrust System)	Mt	200
High Thrust Propellant Mass	Mt	175
Low Thrust Propellant Mass (Two Way)	Mt	150
Total Mass	Mt	650

Figure 10

Performance Of Alternative Propulsion Systems

Evolutionary Mars Exploration

Propulsion System	IMLEO. t		Percent of Chem/AB IMLEO	
	[a]2004	[b]2011	[a]2004	[b]2011
Chem/AB	573	662	100	100
Chem/AP	3800	3141	663	475
'72 NTR	1133	933	198	141
'89 NTR	1031	857	180	129
Advanced NTR	787	680	137	103
NTR/AB	380	443	66	67

[a]2004: First Flight, Opposition-Class Mission
[b]2011: Fifth Flight, Conjunction-Class Mission

Ref: Borowski S. K. et. al. Paper IAF-89-027

Figure 11

Features Of Hybrid Propulsion System Affecting SEI Missions

- *It Pays To Follow A High Thrust With A Low Thrust When Performing Fast Interplanetary Transfer When The Remaining ΔV Is Of The Same Order Or Greater Than The High Thrust ΔV*
- *Such Is The Case When Escaping Most Severe Portion Of The Van Allen Belt*
- *For Comparable Total IMEO NEP/NEP Hybrid System Can Substantially Cut Down The Earth Escape Time*
- *For Comparable Trip Time Power Requirement Can Be Substantially Lower Than NEP System Alone*
- *Alternatively For The Same Power Level Trip Time Can Be Shorter*
- *Structural Requirements Can Be Gracefully Tailored Between High And Low Thrust Requirements*
- *High Thrust Phase Also Usable At Any Point In The Trajectory To Achieve Mission Resiliency*

Figure 12

Effect Of Structure Fraction On Payload Fraction

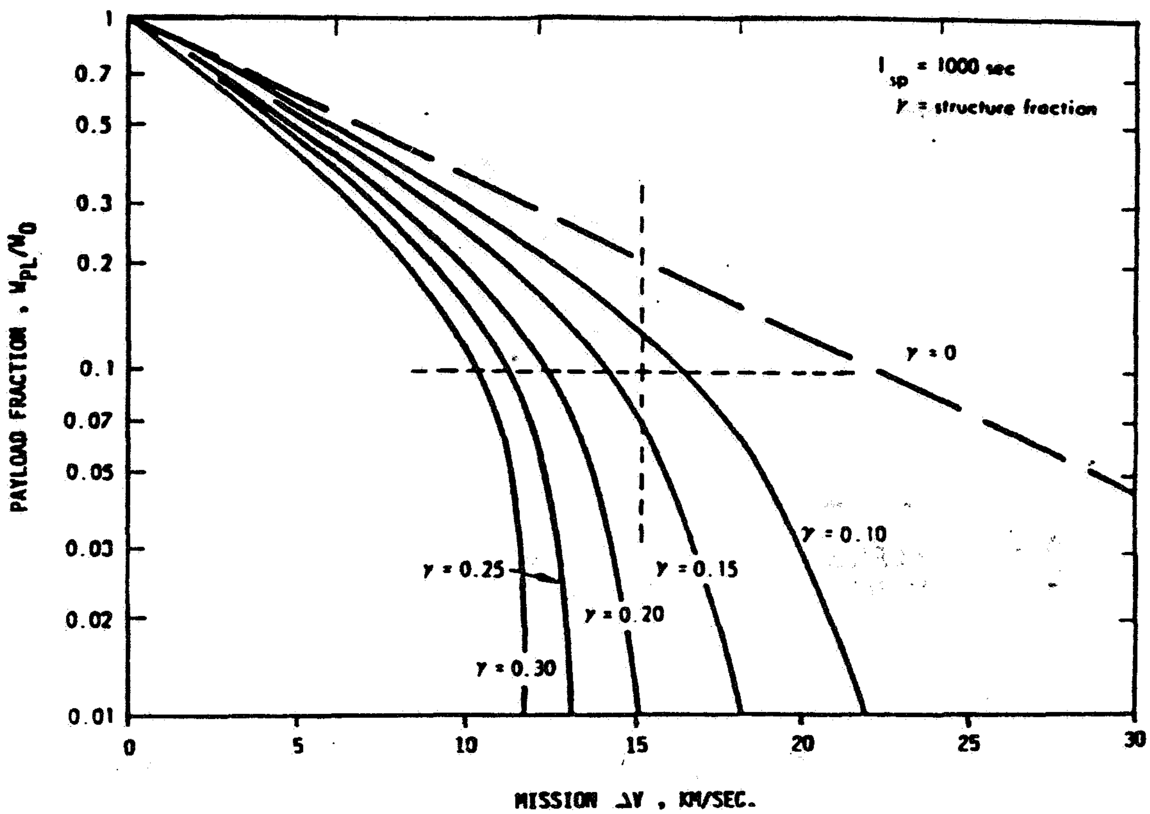


Figure 13

NEP Hybrid System Technology Status And Needs

- ***Conceptual Spacecraft Design Using Arc Jet As Active Load For 100-kWe SP-100 Has Been Studied In Detail For SDI-Missions***
- ***MPD Thrusters More Suitable To Meet High Power Needs***
- ***Shielding Requirements For The Crew Compartment Not Defined Yet***
- ***Shielding For Solar Flare May Overshadow The Need For Short Duration Van Allen Exposure (Can Be Traded Against Power Needed)***
- ***Spacecraft Concepts For Combined High And Low Thrusts Required***
- ***Study Of Best Combinations And Optimization Needed***
- ***Direct Comparison Required With Best NTP System***

- ***Understanding Of Hybrid Systems Is Important To Evaluate System Flexibility To Meet Potential Requirements***
- ***Hybrid High And Low Thrust NEP System Can Provide Cost Effective Way Of Meeting Requirements For Manned Mars Mission***

Figure 14

N92-11098

NUCLEAR ROCKET USING INDIGENOUS MARTIAN FUEL NIMF

Robert Zubrin
Martin Marietta Astronautics

The concept that I am going to be talking about has the endearing name of NIMF. It's a little bit different than the other concepts that have been and will be presented at this meeting because the NIMF is not primarily a space transportation technology. It has an impact on space transportation requirements, but fundamentally the NIMF is a different way altogether of making use of nuclear thermal rocketry through enhanced capability for Mars missions and other kinds of planetary missions.

As everyone here knows, in the 1960's we had the NERVA and ROVER programs, and they developed hydrogen-fueled NTR engines (Figure 1). They were hydrogen fueled in order to have the maximum specific impulse, and the reason why you wanted maximum specific impulse was to lower the mass of the manned Mars missions by increasing the efficiency of space transportation.

There have been innumerable trade studies done of NTR propulsion that show benefits on the order of a factor of 2 for reducing the initial mass in LEO of the manned Mars mission.

But there is a different potential capability of NTR engines. Rather than attempt to exploit them for their potential performance, let's attempt to exploit them for the potential versatility inherent in the concept. What I mean by that is that there is the possibility of designing NTR engines that can use propellants other than hydrogen, in particular propellants that are volatiles indigenous to an extraterrestrial body. If you can do that, you can have tremendous enhancement of the mission capability because you can endow the mission with global mobility at the target planet.

In particular, in talking about Mars, it's quite clear what the optimum indigenous propellant is. The Mars atmosphere is 95 percent carbon dioxide (Figure 2).

You've got a vehicle that comes in and lands on Mars with just enough propellant to set it down, perhaps after a parachute assisted landing (Figure 3). Now it's sitting on the surface of Mars with no propellant in its tank. Then, run a pump and acquire Martian CO_2.

With the temperatures that exist on Mars, CO_2 can be liquefied without refrigeration. It can be liquefied simply by putting it under about 100 psi pressure. So you run a pump, you fill a tank with liquid CO_2, and then, when you want to fly, you just run it through the NTR, heat it to a high temperature-vapor, and shoot it out the rocket nozzle and away you go.

And the performance, while modest by rocketry standards, is good enough to get you back up to orbit, or, what is far more important, to be able to hop from one point on the surface of Mars to any other point on the surface of the planet in a single hop, at which point you can land again and refuel. So you have unlimited global mobility. It means a manned Mars mission can visit ten sites instead of one. So we are talking about an order of magnitude increase in the exploratory capability of a Mars mission by exploiting this potential.

Figure 4 shows a concept of what a NIMF vehicle might look like. You have the astronauts on the control deck, and an additional habitation deck. The pumps are actually much smaller than shown. You only need about 25 kilowatts of pumping power to do the job, which is like a 30 horsepower pump.

Here's the tank of propellant and the NTR engine with a shadow shield above it. A coaxial tank wrapped around the reactor provides supplementary shielding when we are on the surface. So when the reactor is being fired, the crew is up here and they are protected by the shadow shield, by the enormous mass of propellant in the main tank, by miscellaneous equipment, and by a second shield, which is positioned right under them.

The second shield protects them against possible reflected radiation that comes during landing, which is the most critical point of the mission from the shielding point of view. There is more that could be said about vehicle design.

If we are talking about alternative propellants, Figure 5 may be of interest because we always talk hydrogen. These are ideal Isp's. The figure shows infinite expansion ratio Isp's with no nozzle losses included. If you were to include that stuff and had an expansion ratio of say 100, you would be talking about 93 percent of these numbers as realistic performance numbers. So, if we talk about 2800 K, we are talking about 265 seconds Isp with CO_2. Now, with water we are up in the mid 300's, with methane in the high 500's. But water is only available on Mars in the form of ice or permafrost, and so it's much more difficult to access. Methane would require chemical synthesis which makes it still more difficult to access. So, CO_2 is the one that's important.

We can acquire it with simple pump compression (Figure 6). The energy cost of acquiring the CO_2 is very low because it's a simple physical acquisition process. It's on the order of 80 kilowatt hours per ton. That is about 2 orders of magnitude less than the energy cost required to manufacture a propellant; for example, by electrolyzing water and liquefying it or dissociating CO_2 into CO and O-2 and liquefying them.

Since we can use the volatile in its raw form, and the energy comes from the reactor, we have a device that can make its own fuel. The energy costs are so low that the propellant acquisition system can travel with the vehicle, which is not true for a system that would have to synthesize chemical fuel. As I say, the performance is in the mid to high 200's, but that is good enough to attain highly energetic orbits around Mars.

But there is a sticking point. The CO_2, when elevated to high temperatures, becomes an oxidizing medium. It would not be compatible with the fuel elements that were developed for the NERVA program. So, if we are to utilize CO_2 in a nuclear thermal engine, we have to master a new engine chemistry for oxidizing media.

You have a number of options for the propellant acquisition system (Figure 7). You could use a dual use reactor which would allow you high power levels, perhaps a hundred kilowatts. This would allow for rapid refueling. With a hundred kilowatts we could fuel this thing to fly up to its maximum orbit in just 12 days. But you have an issue with the shielding of a critical reactor on the Martian surface for an extended period of time.

We could use solar arrays. They would be set up by the astronauts with a couple of days work on the surface. That could be done. That will work. It's more massive than the other alternatives, but you could do it.

The one that I like the best and which I selected in the NIMF design study that we did (for NASA Headquarters) at Martin was a dynamic isotope power source. It's less than half the mass of a solar array, producing the same amount of power on the Martian surface. We don't have the problem with a critical reactor on the surface. But all three options are viable.

The key issues that define the feasibility of the concept, include the need for a high thrust-to-weight engine (Figure 8). The use of CO_2 as your propellant helps. It degrades your specific impulse, but it increases the thrust for the same energy density of the reactor, so we are talking about triple the thrust of hydrogen at the same power level.

In order to get high thrust, I think we need a high pressure engine, though numbers greater than 800 psi no longer scare people in the NTR communities, so that's not that big a deal.

For high heat transfer area, this would mean that concepts such as particle or pebble bed, where you maximize the heating area of the fuel elements, are most promising for the NIMF. Also, obviously, a small reactor eases the shielding problem, and if the NIMF is going to be used as a manned vehicle, that would also help a lot.

We require fuel materials or coatings that can withstand corrosion by hot CO_2. With a hydrogen NTR, you want 2500-2800 K because you are in direct competition with chemical aerobrake. Unless you have those rather high temperatures, you can't demonstrate a performance advantage of significance.

With the NIMF, that's not the case. There is nothing in competition with it. There is no other enabling technology for global mobility on Mars. If it works at any level of performance, it does the job.

The high temperature 2800 K is desirable because it would enable you to go from the surface of Mars to extremely energetic, highly elliptical orbits around Mars. But frankly, a suborbital vehicle that could hop around the planet, which would require fuel temperatures on the order of 1200 or 1400 K, would represent a tremendous increase in our capability on Mars.

So the most promising appears to be mixed thoria/urania oxide fuel pellets coated by zirc oxide, which might reach the 2800 K temperature. Beryllium oxide was used in the Pluto program. It's a lower temperature material, maybe 2400 K. If we are under 1900 K or so we could talk about urania/carbide fuel elements coated by silicon carbide, which after all resists oxidation in air on space shuttle tiles at that kind of temperature, and air is a more serious oxidizer than CO_2.

The data in Figure 9 was compiled by people at NASA Lewis working on resistojets. As you can see, the zirc oxide was good up to 2700 K in oxygen. So that really might get us close to where we want to be.

There needs to be a serious program of engine chemistry to determine the optimum materials and test them, and this can be done at fairly modest cost, near-term, in electric furnaces.

Figure 10 shows what the propellant temperature does. As you can see, if you are interested say in attaining low Mars orbit, and if the vehicle can have a mass ratio up to 8, which is reasonable because CO_2 is a high density propellant, even 2000 K does it.

If you want to attain a highly energetic elliptical orbit, you better have 2600 K. And if you want to do a direct trans-Earth injection from the Martian surface, you better be over 2800 K. So depending upon what you want to do, the temperature requirement that you have to be able to attain is determined.

The ballistic NIMF is probably the more promising one (Figure 11). It's lighter and can do more, and you can see that this was designed at 2800 K and it could attain the highly elliptical Mars orbit. But even as low as 2000 it was still getting to low Mars orbit. That's consistent with what I mentioned before.

Sometimes in the past people have proposed using a carbon monoxide/oxygen bipropellant hopper as the basis for Mars global mobility.

Since Mars atmosphere is CO_2, people have proposed making bipropellant out of it. The problem is that the energy requirements for propellent productions are 100 times greater than for the NIMF (Figure 12). What that means is that the carbon monoxide hopper (CMH) has to have a fixed base. Therefore, to explore a particular site, it has to do twice the Delta V as the NIMF because it has to hop there, land there, and then hop back to the base. Additionally, the Isp is almost the same on the CMH as on the NIMF.

You can get the mass ratio up a little more because the thing is lighter, but fundamentally the doubling of the Delta V is the dominating factor here. You can see that CMH loses it at around the 1300 kilometers range, whereas the NIMF can just hop up to orbit and come down anywhere on the planet. So the NIMF has global mobility and a chemical hopper simply does not have global mobility. That's all there is to it.

The NIMF can also be used to deliver cargo (Figure 13). With 10 tons we can still make it back up to orbit, but with 40 tons we could hop 4000 kilometers, which is roughly the distance from the Martian pole to the equator.

So if you had a base at the equator where there is more solar energy and warmth and so forth, but no water, you could send the NIMF up to the pole, scoop up 40 tons of water from the polar cap, and hop back to the base with it. Also obviously you could hop around the planet depositing science payloads in various places and setting up a global science network.

It can take 40 tons 4000 kilometers, or it could take 100 tons 1000 kilometers. If we have a base on Mars, there will always be some raw material which isn't situated right where you are and it would really be useful to have this capability to move payloads around the planet.

Now, I did not analyze the NIMF using the same mission plan that was used as the standard mission for the other concepts at this conference. The reason for that is twofold. First of all, the NIMF completely changes what the payloads are that you would send to Mars, so you are changing the manifest: the comparison goes out the window.

The other thing is that I think that the mission plan that was chosen for this conference doesn't have any merit because it spends 400 days in transit and only 30 days at Mars. That's a very inefficient way to try to explore Mars.

Figure 14 shows a variety of propulsion options: Chemical propulsion, chemical with an Aerobrake, NTR, NTR with an Aerobrake, carbon monoxide hopper and NIMF.

As an example, say NTR all propulsive, on the first mission where the NIMF or the CMH have to both be transported to Mars, the mission masses are not too different. But on the second mission, CMH & NIMF halve the mass in LEO.

Even if you were to average this over a five mission sequence, they would be roughly a factor of 2 lower in LEO than the conventional approach. The CMH and the NIMF are about the same mass-wise. However, the CMH can only visit one site whereas the NIMF has global mobility.

Figure 15 shows the figure of merit I use for a manned Mars mission. Figure 16 shows that the NIMF mission has about a factor of 30 greater figure of merit than the conventional lander, and a factor of 10 greater than the CMH.

If we assume all NTR, all propulsive for the space transfer, and you want to conduct a program of Mars exploration incorporating landing at 50 discrete locations on the surface of Mars, Figure 17 shows the total mass of the NTR mission with and without the NIMF. The total mass with a conventional lander is 11700 tonnes, using the NIMF reduces this to 640 tonnes. Using the NIMF shows a factor of 20 benefit. This is much greater than would be afforded by any advanced space transportation propulsion technology.

Figure 18 depicts a manned Mars mission being launched using a NIMF and one launch of a heavy lift launch vehicle.

One early possible application of the NIMF would be unmanned as a Mars Rover sample return mission (MRSR). The Centaur throws the NIMF to Mars where it lands on Mars, it hops around, visits ten sites. The unmanned NIMF collects samples from ten sites, then ascends to orbit. It then shoots the samples back in one of these sample return vehicles (Figure 19). Now, we may discover that site Numbers 3 and 8 were the interesting ones. So, we send the NIMF back there and get a second consignment of samples and fire them back.

The comparison between this and a conventional MRSR mission is quite profound. We are able to do it in one launch instead of several. We return 220 kilograms of samples instead of five, 22 times more sample payload. They come from at least ten sites instead of one, and there are two sample shipments allowing some degree of feedback in the mission, instead of none.

It's possible to extend the NIMF concept to other destinations in the solar system (Figure 20). There is water ice on the moons of Jupiter, Saturn, and Uranus. There is methane on Titan, and we could actually envision performing sample return missions from these bodies using this sort of approach, though there would be some technological change.

Envision an unmanned sample return mission to Titan. It would use methane as propellant. It uses on NTR to kick itself out to Titan where it aerocaptures. Once it's going slow in Titan's atmosphere, it unfolds wings.

Titan has four times the atmospheric density of the Earth and 1/7th the gravity, so it's the aviation paradise of the solar system. A vehicle with wings can remain airborne flying at a speed of 25 miles an hour in Titan's atmosphere.

When it's all done doing its low-level aerial reconnaisance of Titan, which is necessary because Titan is clouded over, you tank up with methane from Titan's atmosphere. Then you either do a big Delta V and go back to Earth, or you could actually fly from Titan to any one of Saturn's other moons (except for Mimas), land, collect some samples, go back to Titan and refuel, and then jet back to Earth. So it opens up the capability for some rather spectacular unmanned outer planetary missions.

In conclusion, the NIMF technology offers extremely high leverage in increasing the cost effectiveness of missions to Mars and the outer solar system (Figures 21).

It reduces the IMLEO of a given Mars mission (if we figure it as part of the sequence of even three or four missions) by about a factor of 2, regardless of the propulsion technology, simply because you are reducing the payload manifest. It enables a manned Mars mission in a single HLV launch.

It increases the number of sites visited per mission by a factor of 10 or more, and that is really what counts. That's the big leverage. It enables global transport on Mars. It increases the science return of a Mars Rover sample return by an order of magnitude, and extensions of the technology could enable sample return missions to the outer solar system.

Therefore, I maintain that the NIMF offers greater leverage for Mars exploration than any other advanced propulsion concept.

The NIMF is not a trivial technology challenge. I would say this concept is at technology level 2; we have to demonstrate new engine chemistry. However, there are no fancy physics here. It's the same kind of thing we did with the NERVA or other NTR concepts except we're doing it in a different context. It's just a solid core reactor with a different propellant.

What we recommend is this: The immediate focus should be a NERVA derivative or other solid core hydrogen fueled NTR system (Figure 22). The number two priority should be the development of a CO_2 NIMF because, even though the chemistry is different, the people, the test facilities, a lot of the computer codes and so forth that are used in the hydrogen NTR program could be shifted over later to the NIMF.

Once the NERVA clears the test facilities, we could put the NIMF in there. So I see it as an evolutionary program. I would say that the NTR development evolving towards NIMF can enable a much more capable and cost effective program.

A VOICE: Have you received any feedback, that, by operating a nuclear propulsion system in the atmosphere of Mars, you may be disturbing the ground which you are striving to gather and study by neutron activation?

MR. ZUBRIN: Oh, well, you would collect the samples from an adequate distance from the landing site.

A VOICE: You could be a kilometer away and still have a significant aggravation.

MR. ZUBRIN: Oh, I don't think so. The activated materials will be very easy to identify as such since we are quite clear that these things don't exist in neutron-activated short half-life form on the surface of Mars.

A VOICE: I think you have a potential problem here that the science folks will really have a problem with.

MR. ZUBRIN: Well, actually our strongest support has been from the science house in NASA.

A VOICE: Those are mission planners, those aren't the guys that get the samples back.

MR. ZUBRIN: The samples can be collected from sufficient distance from the landing site.

MR. ZUBRIN: Let me just take one more question.

A VOICE: What's the probability in your mind that this ability to hop around will be a mission requirement, either initially or second or third mission?

MR. ZUBRIN: Well, I don't know if it will be a mission requirement for the manned mission, but it's extremely desirable from the point of view of being able to carry out effective science.

Initially, we may have a small unmanned NIMF which acts as an auxiliary for the manned crew. They can send this thing hopping around the planet, which also gets you around a number of shielding problems on the vehicle so it can fetch and bring, collect samples.

That might be an initial way to implement it; prove the technology in an unmanned mode. But in terms of whether JSC all of a sudden will come out and say that is a requirement, I couldn't predict that.

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Martian Propellants

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Introduction

In the 1960s, Nuclear Thermal Rocket (NTR) engines were developed and ground tested capable of yielding Isp of up to 900 s at thrusts up to 250 klb.

Numerous trade studies have shown that such traditional hydrogen fueled NTR can reduce the IMLEO of Lunar missions by 35% and Mars missions by 50 to 65%.

The same personnel and facilities used to revive the hydrogen NTR can also be used to develop NTR engines capable of using indigenous Martian volatiles as propellant.

By putting this capability of the NTR to work in a Mars Descent/Ascent Vehicle, the NIMF (Nuclear rocket using Indigenous Martian Fuel) can greatly reduce the initial mass in LEO of a manned Mars mission, while giving the expedition unlimited planetwide mobility.

Figure 1

The Martian Atmosphere

Carbon Dioxide	95.00 %
Nitrogen	2.70 %
Argon	1.60 %
Water	0.30 %
Oxygen	0.13 %
Carbon Monoxide	0.07 %

Figure 2

Nuclear Rocket Utilizing Indigenous Martian Fuel (NIMF)

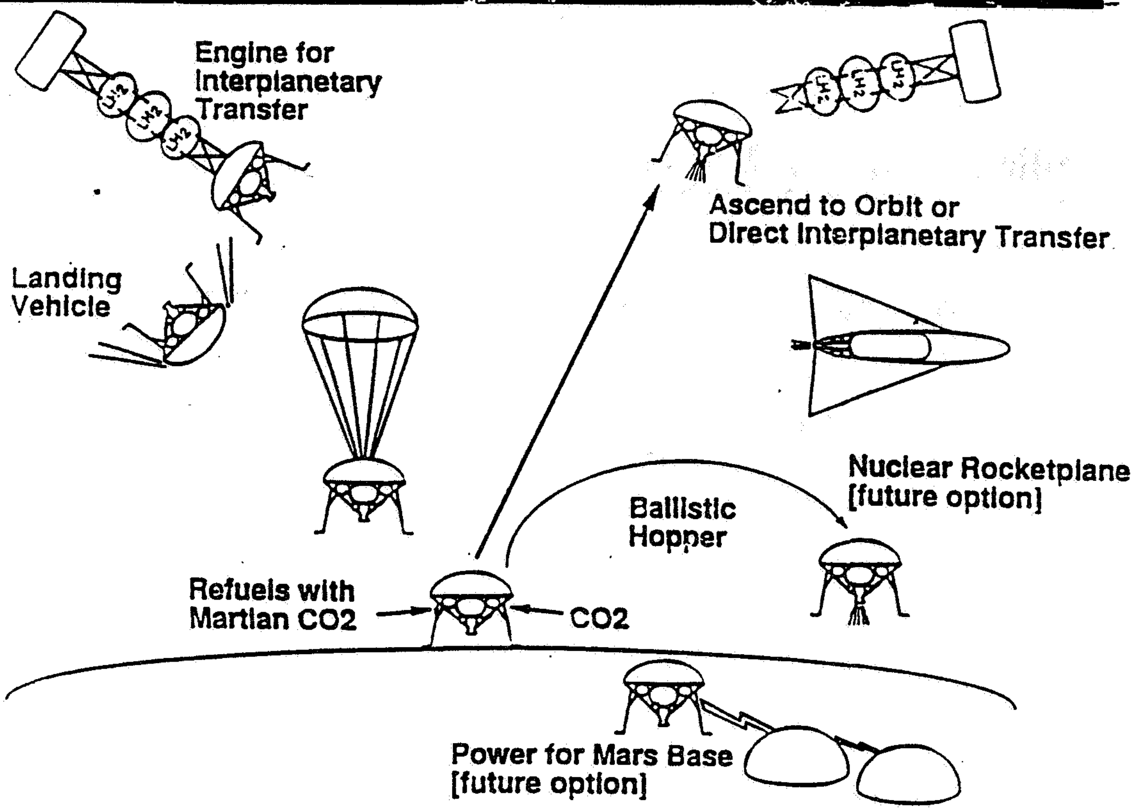

Figure 3

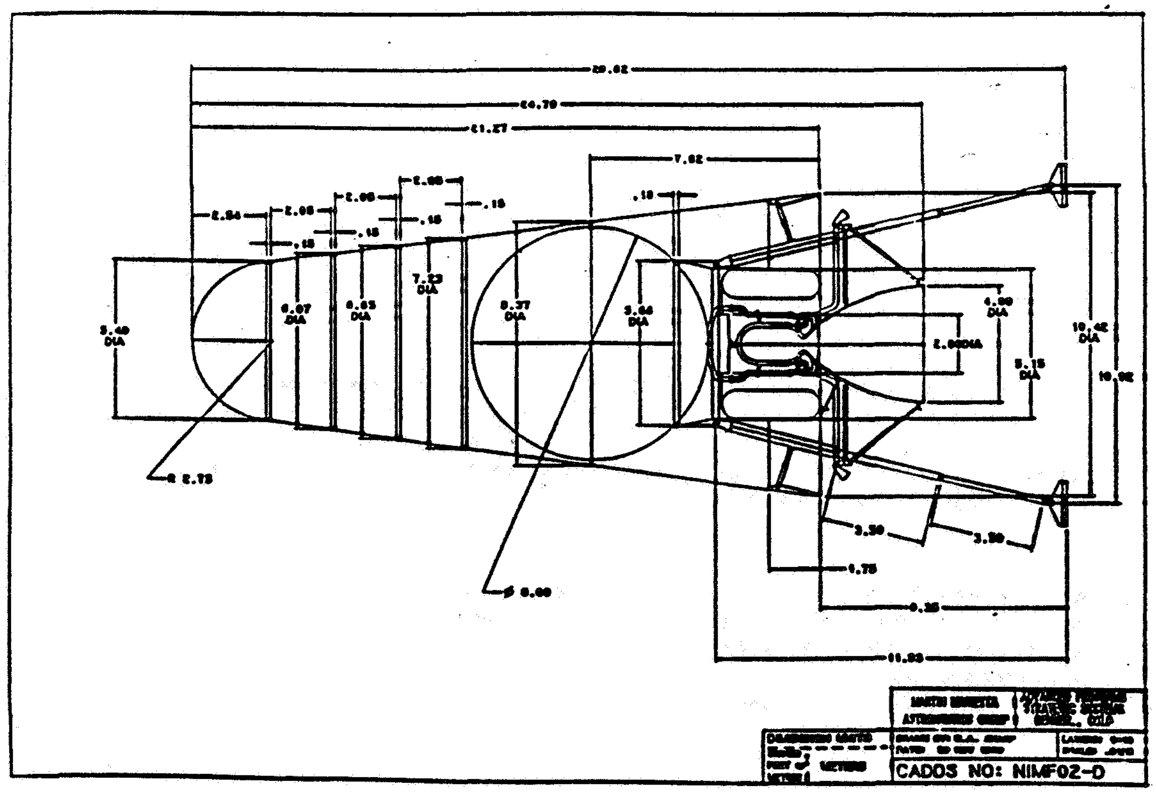

Figure 4

Ideal Specific Impulse of Martian Propellants

Temperature	CO2	Water	Methane	CO or N2	Argon
1400 K	162	222	460	162	110
2800 K	283	370	606	253	165
3000 K	310	393	625	264	172
3200 K	337	418	644	274	178
3500 K	381	458	671	289	187

Figure 5

NIMF Propellants

Carbon Dioxide

- Most readily available propellant on Mars. Can be aquired by simple pump compression at an energy cost of 84 kW-hrs per metric ton.
- Storable liquid at 233 K under 147 psi pressure, Density is 1.16 that of water.
- Modest performer. Isp = 280 sec. Sufficient for ascents to high orbits.
- Requires the development of oxide fuel elements.

Water

- Aquisition requires the melting of ice or permafrost. Reactor heated CO2 or steam can be used to do this.
- Propellant tanks must be insulated or heated to avoid freezing under martian conditions.
- Good performer. Isp = 350 sec. Sufficient for direct ascent to Trans-Earth Injection.
- Widely available on moons of outer planets, and possibly on Phobos and several asteroids as well.
- Requires the development of oxide fuel elements.

Methane

- Aquisition requires melting of ice or permafrost, and using reactor heat to crack CO2 and drive synthesis.
- Mild cryogen. Liquid at 135 K under 74 psi. Density is 0.46 that of water.
- Excellent performer. Isp = 560 sec. Sufficient for direct ascent to high energy Trans-Earth Injection orbits.
- Available on Titan and Triton.
- Can use conventional NERVA carbide fuel elements. Coking may be a concern.

Figure 6

Propellant Acquisition System Options

(1) Dual Use Reactor.
- 100 kWe possible.
- Allows flight to max orbit in 12 days refueling.
- Shielding of critical reactor on surface an issue.

(2) Solar Arrays
- 25 kWe average (round the clock) power requires 3500 m^2 array.
- Such an array would mass 8.8 tonnes and take 3 astronauts 2 days to set up.
- Solar option appears feasible but unattractive.

(3) Dynamic Isotope Power Source (DIPS)
- 30 kWe DIPS would mass 4 tonnes.
- Allows fueling for flight to maximum orbit in 50 days.
- No major operational issues.
- Selected.

Figure 7

Key Issues Defining NIMF Feasibility

- Requires high thrust to weight NTR engines

  - Use of CO_2 propellant helps. Provides triple the thrust of hydrogen NTR at the same power level.

  - High pressure (> 800 psi) engines appear desirable to increase the power density.

  - High heat transfer area concepts such as the particle or pebble bed appear most promising.

- Requires fuel materials or coatings that can withstand corrosion by hot (> 2200 K CO_2).

  - Prime options for high temperature operation include coatings of either ThO_2, ZrO_2, or BeO around UO_2/ThO_2 fuel pellets. Operation with UO_2/ThO_2 fuel pellets coated by ZrO_2 as high as 2800 K may be feasible.

  - Experience base exists for high temperature BeO (Pluto program). May enable operation as high as 2400 K.

  - Possible alternatives for lower temperature (< 1900 K) operation include UC_2 fuel coated with either SiC or NbC. Would enable use of NERVA/ROVER fuel technology in suborbital hopping vehicle.

Figure 8

MATERIAL	REACTIONS	EVAPORATION RATE (gms./cm^2 sec) temperature	mass loss
Hafnium Carbide	1000°C Nb,Ta,W,N_2,H_2 1700°C O_2,(air) 2000°C Mo	2500°C 2700°C	$1x10^{-6}$ $8x10^{-6}$
Zirconia Carbide	1000°C Nb, N_2,H_2 <1200°C O_2(air) 2000°C Mo	2500°C 2700°C	$2x10^{-8}$(calculated) $3x10^{-6}$(calculated)
Tantalum Carbide	1000°C Nb,H_2,W	2700°C 2900°C	$1x10^{-6}$ $6x10^{-6}$
Titanium Carbide	1000°C Nb, N_2 1200°C CO_2 2000°C Mo 2400°C H_2 <2000°C O_2(air)	2200°C 2300°C	$3x10^{-5}$ $1x10^{-4}$
Niobium Carbide	1600°C W, H_2 1800°C Nb,Ta,Mo	2500°C 2700°C	$7x10^{-7}$ $7x10^{-6}$
Zirconia Oxide	1900°C NH_3,C 2400°C H_2,O_2(air)	2000°C 2400°C	$2x10^{-7}$ $2x10^{-5}$
Thoria Oxide	2600°C O_2,(air)	2000°C 2400°C	$2x10^{-6}$ $3x10^{-4}$
Magnesia Oxide	?°C CO, H_2O	similar to zirconia	

Figure 9

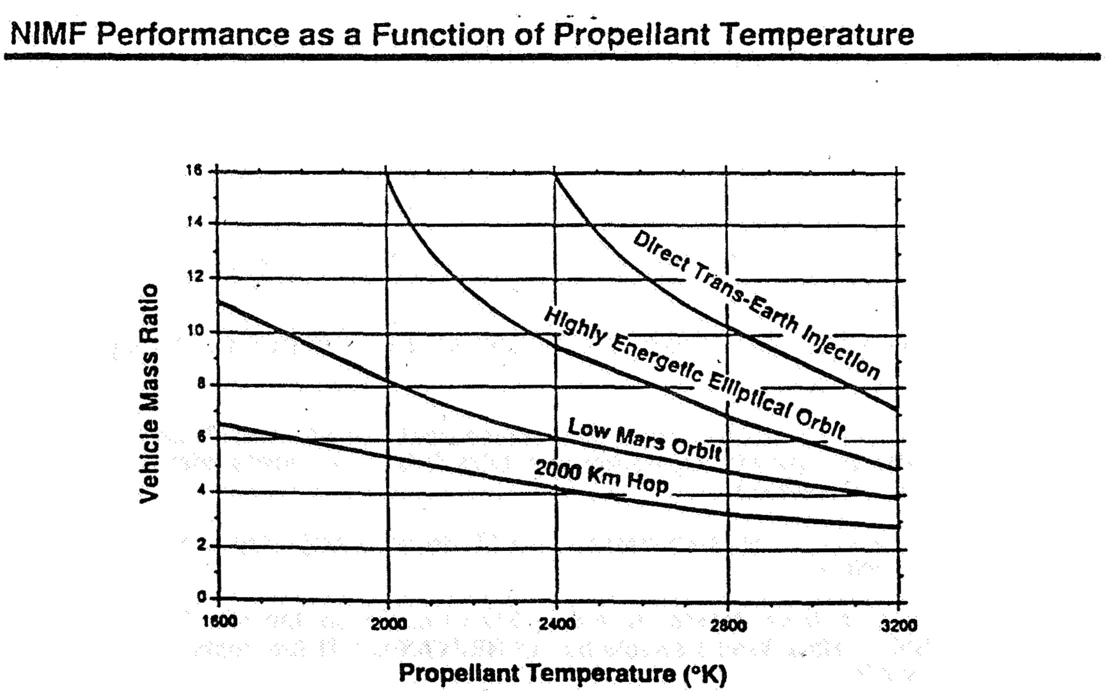

Figure 10

Performance of CO_2 Propelled NIMF

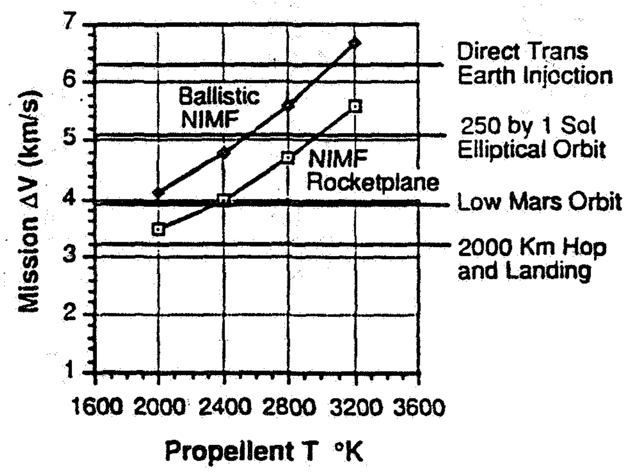

Figure 11

NIMF and CMH Mass Ratio vs Hop Range

Comparison of Mobility of Ballistic NIMF and Carbon Monoxide Hopper (CMH)

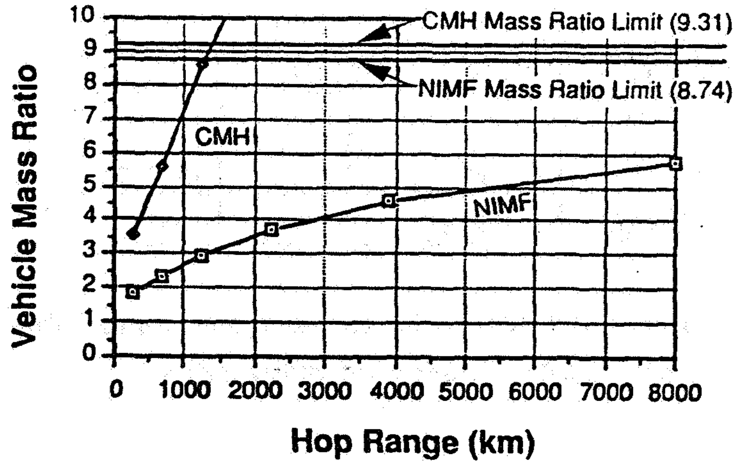

Figure 12

Cargo Capability of the Ballistic NIMF

Cargo(tonnes)	Mass Ratio	Delta-V(Km/s)	Range(km)
10	7.03	5.047	Orbital
20	6.04	4.65	8500
30	5.23	4.28	6000
40	4.61	3.96	3920
50	4.13	3.67	3000
60	3.73	3.41	2280
70	3.41	3.17	1800
80	3.13	2.96	1450
90	2.90	2.75	1220
100	2.70	2.57	1000

We thus see that the ballistic NIMF can transport cargos of up to 40 tonnes over distances of 4000 km, and cargos of up to 100 tonnes over distances of 1000 km across the Martian surface.

The NIMF requires no propellant producing infrastructure at either end of the route to accomplish the cargo transport. To achieve a comparable performance, a chemical vehicle would require propellant producing base facilities at both ends of the route.

Figure 13

ETO Masses of Manned Mars Missions (tonnes)

Propulsion	CMD/AV	CMH	NIMF
	1st/2nd	1st/2nd	1st/2nd
cryo	616/591	546/399	551/363
cryo/AB(E)	431/406	360/214	365/178
cryo/AB(EM)	355/330	297/174	301/142
NTR	258/228	220/124	223/104
NTR/AB(E)	226/201	188/ 93	184/ 74

The NIMF and CMH ETO masses are comparable, but the NIMF can visit 10 times as many sites.

Figure 14

Merit Factor for Manned Mars Missions

$$m = (n)(t)(p)$$

m = dimensionless merit factor. Should be made as high as possible.

t = time efficiency = (time on Mars)/(time in transit)

p = (dry payload on Mars)/(ETO mass)

n = number of discrete landing sites visited by drymass payload.

Figure 15

Merit Factor "m" for All Manned Mars Mission Options

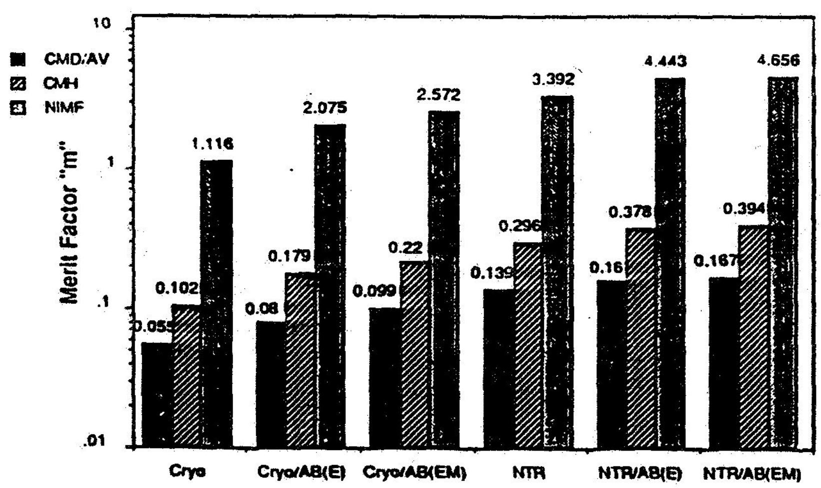

Figure 16

ETO Mass Required for Manned Mars Landings at 50 Sites

Conventional Lander	11700 tonnes
NIMF	640 tonnes

• Assumes NTR all propulsive for space transfer.

Why 50 landings? Mars is a big place. Assuming the use of ground exploration vehicles with a 1000 km one way range, 50 widely separated landings will only provide one-time access to 27% of the martian surface.

The use of the NIMF thus reduces the ETO mass required to support a program of Mars exploration by a factor of 20.

This is much greater leverage than that afforded by any advanced space transportation propulsion technology. Even a hypothetical perfect (i.e. infinite T/W, infinite Isp) space transportation engine using a conventional lander would still be outclassed by an NTR/NIMF combination by a factor of 5.

Figure 17

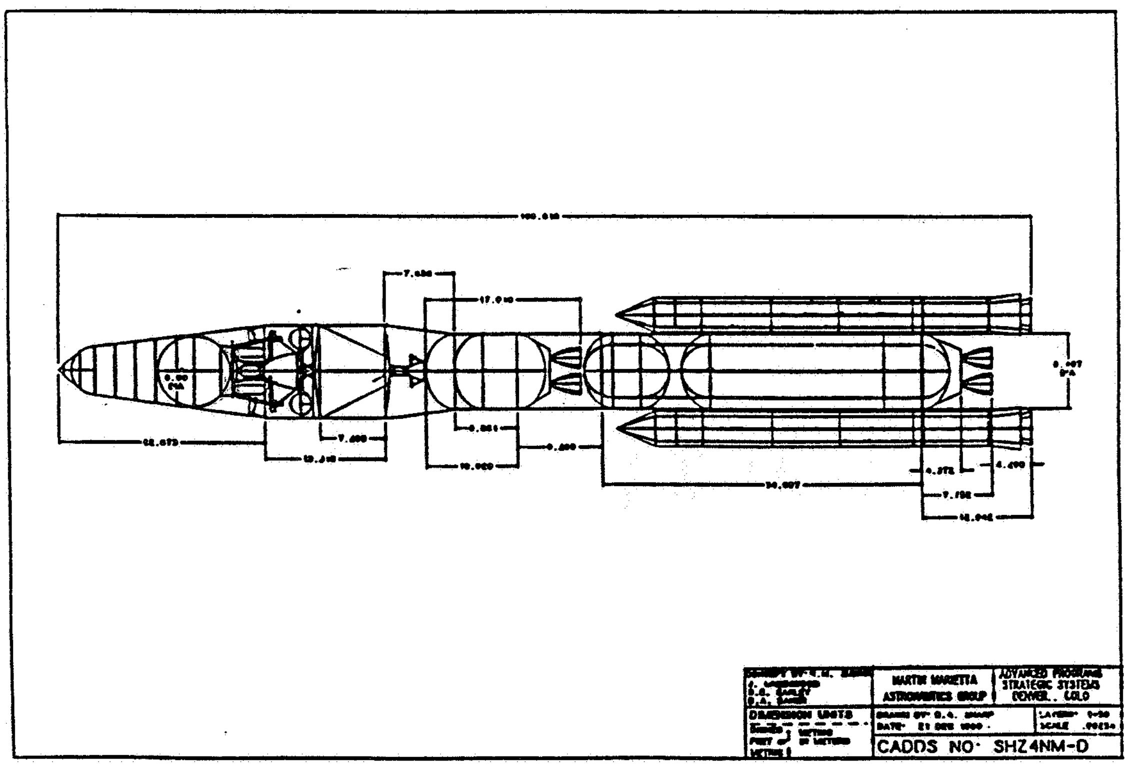

Figure 18

Mars Rover Sample Return Utilizing a NIMF

Mission Parameters

NIMF mass	5.30 tonnes	
SRVs (2)	0.56 tonnes	
Science payload	1.00 tonnes	
Cryogenic stage	14.40 tonnes	Isp= 460 s
Sample returned	0.22 tonnes	

Launch Vehicle	Titan IV
Number of Sites Visited	10
Number of Sample Shipments	2

(allows redirection by scientists after the first SRV returns)

The NIMF MRSR mission returns 45 times as much samples from 10 times as many sites as a conventional MRSR mission.

Figure 19

Exotic Missions Made Possible By NIMF Propulsion

In addition to its primary purpose as facilitating technology for manned and large scale unmanned Mars missions, the NIMF engine can also enable a number of exotic missions. Some of these exotic missions include:

- Multiple sample missions from all the moons of the major planets. Ice is available on several of the moons of Jupiter, Saturn, and Uranus; methane is available on Titan and Triton. A methane fueled NIMF could use Titan as a base for repeated sorties to each of Saturn's moons. Water fueled NIMFs could use the ice worlds as bases.

- Prospecting the asteroid belt with water fueled NIMFs. Ice is available on Ceres and several Trojan asteroids.

- Venus surface sample return carried out by a winged automated NIMF using CO2 propellant.

- Atmospheric sample return from Saturn, Uranus, or Neptune carried out by a winged NIMF using hydrogen propellant and airborne aquisition.

- Comet core sample return, using the comet ice itself for return propellant. Possible mission to the Oort Cloud, using comets as bases.

- Shuttle service between the Earth's Moon and Phobos, using lunar SO2 and Phobos water as propellants.

Figure 20

Conclusions

• NIMF technology offers extremely high leverage in increasing the cost-effectiveness of missions to Mars and the outer solar system.

•• Reduces the IMLEO of a given Mars mission by a factor of 2.
•• Enables a Manned Mars mission in a single HLLV launch.
•• Increase the number of sites visited per mission by a factor of 10.
•• Enables global mobility on Mars.
•• Creates the capability for global transport of cargo (essential for settlement)
•• Increases the science return of MRSR by an order of magnitude.
•• Enables sample return missions to the outer solar system.

The NIMF offers greater leverage for Mars exploration than *any* other advanced propulsion concept.

• NIMF technology poses a development challenge more formidable than the revival of NERVA, but less than that of the exotic NTR propulsion concepts.

•• New engine chemistry must be mastered.
•• But no "fancy physics" is required.

Figure 21

We therefore Recommend:

• That the immediate focus for advanced propulsion development be an updated hydrogen driven NERVA derivative;

• That the development of a CO_2 propelled NIMF be made the number 2 program priority.

• Thus, as the NERVA derivative moves through various phases of its maturity, the capabilities associated with earlier phases of its development be rescheduled to support the development of the NIMF. Such capabilities include:

•• Preliminary design, engineering, and test personnel.
•• Thermal hydraulics, shielding, and neutronics codes.
•• Test facilities.

• NTR development evolving towards NIMF technology will thus enable a much more capable and cost effective Space Exploration Initiative.

Figure 22

N92-11099

WIRE CORE REACTOR FOR NTP

R. B. Harty
Rocketdyne/Rockwell International

I am going to talk about the wire core concept. This is not a new concept. It originated primarily by GE in the aircraft nuclear propulsion program, and this was a concept that they determined was the best for that particular application. However, the program was canceled, and AI (Atomics International) picked it up and did a fairly complete conceptual design study from 1963 to 1965, and even made some fuel. Nothing more has been done since that particular time.

You will notice there are some things that are missing. One is development planning. During this period, there was no development planning activities. Also, very little was done on safety. That does not imply that this concept is not safe. There are some very good safety features, but it does need to be updated to the current safety criteria.

The wire core is a system that has a thrust of 205,000 pounds -- we did not have the time or resources to characterize a system in the 75,000 pound category. A wire core consists of a fuel wire with spacer wires (Figure 1). It's an annular flow core. It has a central control rod. There are actually four of these, with beryllium solid reflectors on both ends and all the way around.

Figure 2 shows some details of the wire core. The wire diameter is 34 mils; about the size of a paper clip. The cladding, which is a tungsten rhenium alloy, varies in thickness from 2 to 7 mils depending on where it is in the radial direction of the core. The spacing is about 70 mils but that is a also variable. The fuel used was uranium nitrate. Figure 3 is a cross-section of the core just to show the size for this 205,000 lbs. of thrust. The core diameter is 24 inches with an 8-inch diameter central hole.

Figure 4 is the sketch of the overall engine. Most of the work during this study was done on the reactor and not on the engine. It is a bleed cycle with the hot gas driving the turbine. The outlet gas temperature here is 5000 degrees fahrenheit. Figure 5 is a summary of the performance with a block diagram of the engine cycle. The thrust is 205,000 lbs., which consists of thrust from the main nozzle, plus the exhaust from the pump nozzle. The actual specific impulse is 930 seconds.

Figure 6 is a breakdown of the reactor-only weight, and it includes the shield. You can see a significant portion is the gamma shield material. The total mass of this particular reactor here is 11,000 pounds.

Figure 7 shows engine weight vs. thrust. As previously mentioned, most of the work was performed for the reactor, so a range of values is shown. At 75,000 pounds thrust, the engine will weigh between 7000 and 9000 pounds.

The fuel element is a wire configuration. Figure 8 shows a technique used to fabricate this particular fuel element. They started with an 8 mil tungsten wire, and braided the wire into 125 mil diameter tubes. Then they packed 4 mil-sized UN fuel particles, coated with tungsten rhenium, inside the tubes. The tube was vapor-deposited with tungsten so that all the pores in the wire mesh were filled with tungsten. This tube was then swaged from 125 to 75 mils and drawn down to 35 mil diameter wire.

The wire core reactor has a high power density at the inlet of the core where the hydrogen is very cold. By controlling the spacing with the cladding thickness, one can obtain very low Delta-T's within your fuel element. The cladding thickness varies from 3 to 7 mils with a total diameter of the wire of 35 mils.

One of the problem areas of a radial flow reactor is flow distribution. In this particular reactor, half the flow comes in at the bottom and half at the top. It then turns and goes radially through the core.

If a reactor were built with constant axial spacing, one would obtain an axial temperature distribution as shown in Figure 9. This is the actual Delta-T divided by Delta-T average over the axial location. There are peaks up to 1.5, which is clearly unacceptable, while something in the order of 1.05 is required. The wire core reactor can be designed to control the Delta-T. Figure 10 shows what could be done to obtain a perfectly flat Delta T in the axial direction.

Figure 11 reviews some of the advantages of the wire core reactor. It has a very large heat transfer area. There are 570 square feet per cubic foot that can be compared to 120 for a typical NERVA reactor. The wire core reactor also has very large heat transfer coefficients. Radial flow also provides flow divergence, so when the gas is becoming hotter there is a much larger flow area. Separation of fuel and structure relies on the wire cladding for strength, not the fuel. There is a short heat path in the wire source, since the wire is only 35 mils in diameter. Compatibility of the fuel cladding and propellant is very important. The UN, tungsten and the hydrogen are all compatible at these high temperatures.

With radial flow the gas loads cancel in all directions. Also, high specific impulse, 5000 degree fahrenheit temperature capability, and restart capability are other advantages. One of the areas requiring development is the fuel element. There has been a lot of work on rhenium and uranium nitrate fuel and a review of this information is required to derive an adequate development program. Fuel fabrication development is also required, as is more work on safety.

WIRE CORE REACTOR

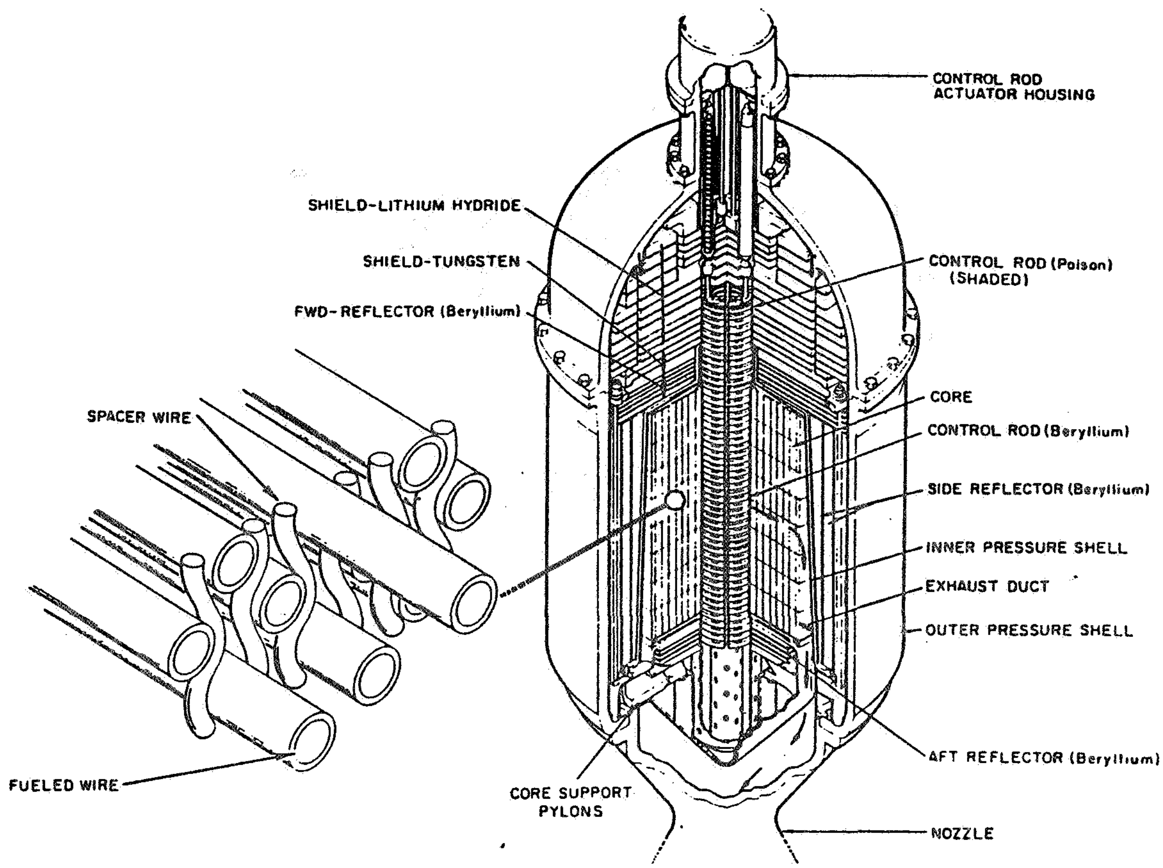

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Figure 1

REACTOR FUEL MATRIX

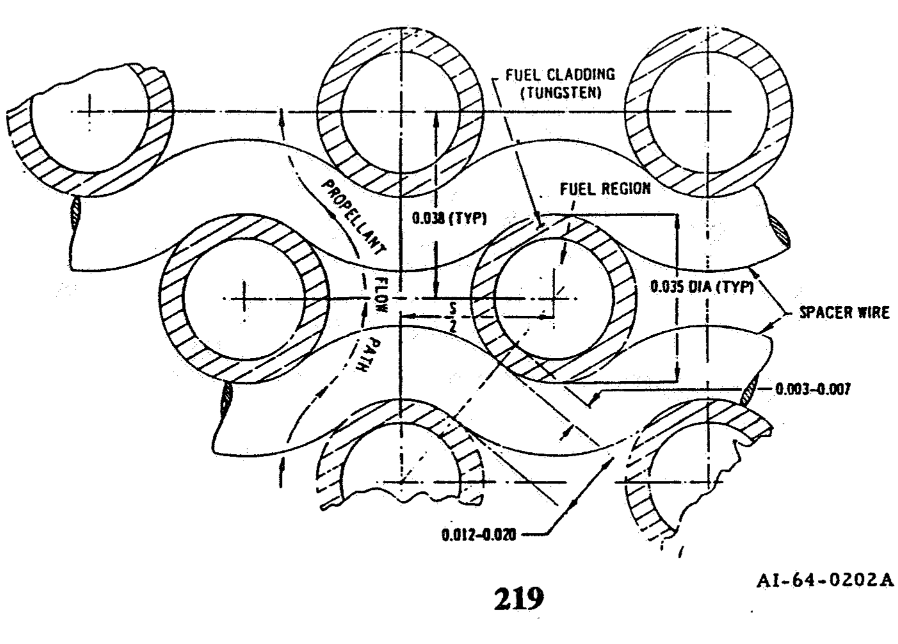

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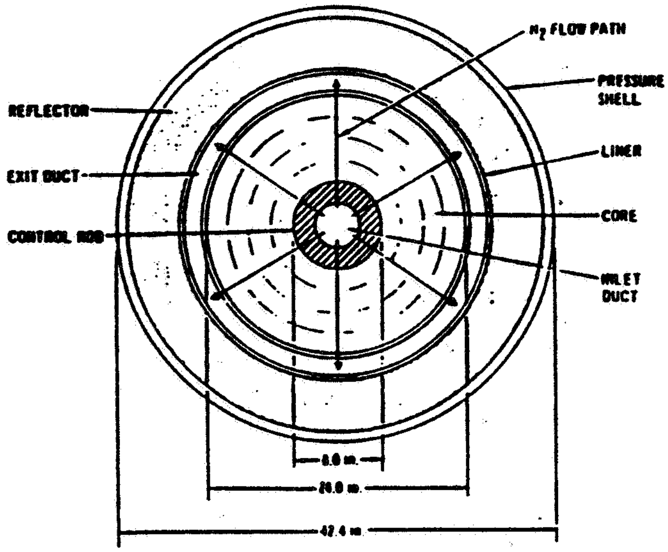
Figure 2

REACTOR CROSS SECTION



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Rockwell International
Rocketdyne Division

Figure 3

NUCLEAR ROCKET SCHEMATIC FLOW DIAGRAM

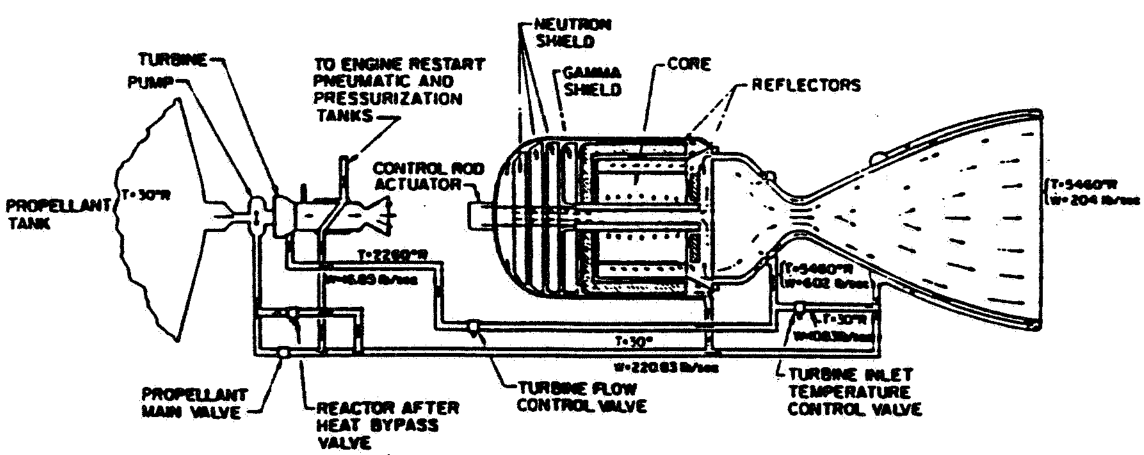

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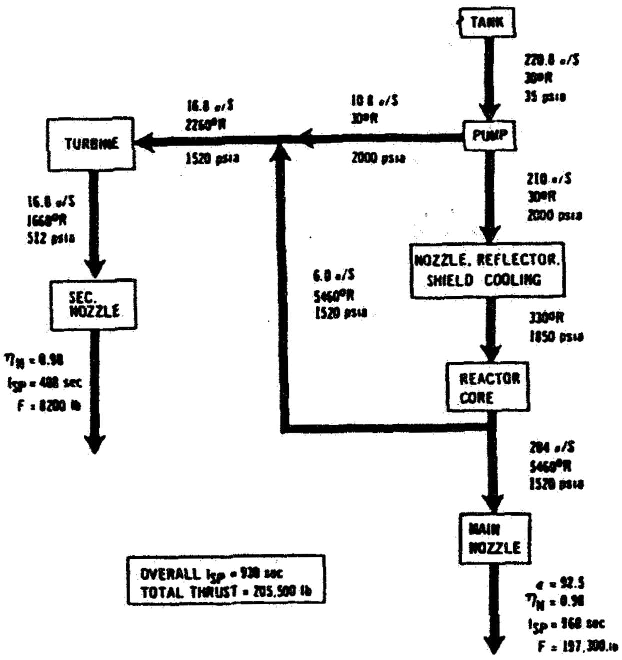
Figure 4

NUCLEAR ROCKET PERFORMANCE CHARACTERISTICS



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Thrust (lb)	205,500
Specific Impulse (sec)	930
Core Inlet Temperature (°R)	330
Core Exit Temperature (°R)	5460
Core Inlet Pressure (psia)	1850
Core Exit Pressure (psia)	~1500
Core Propellant Flow (lb/sec)	210
Core Thermal Power (Mw)	4400
Pump Cycle	Bleed
Turbine Flow (lb/sec)	17
Exit Area/Throat Area	92.5
Nozzle Efficiency	0.98

Figure 5

REACTOR/SHIELD ASSEMBLY WEIGHT BREAKDOWN

205,500 LB THRUST

	Pounds
Active Core	3,440
Reflectors	2,000
Gamma Shield	2,330
Neutron Shield	540
Control and Actuators	420
Outer Pressure Shell	935
Inner Pressure Shell	560
Core Rear Support	250
Core Front Support	350
Core Sheath	360
Total Weight	11,185

Figure 6

PREDICTED ROCKET ENGINE WEIGHT VS THRUST

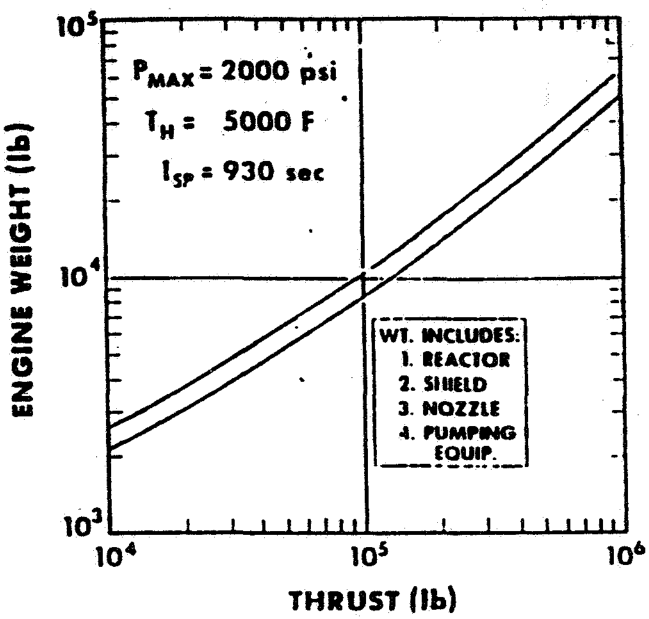
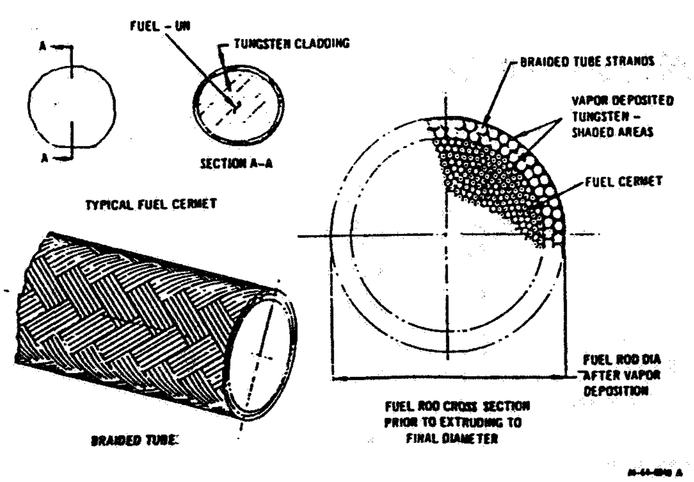


Figure 7

WIRE FUEL FABRICATION

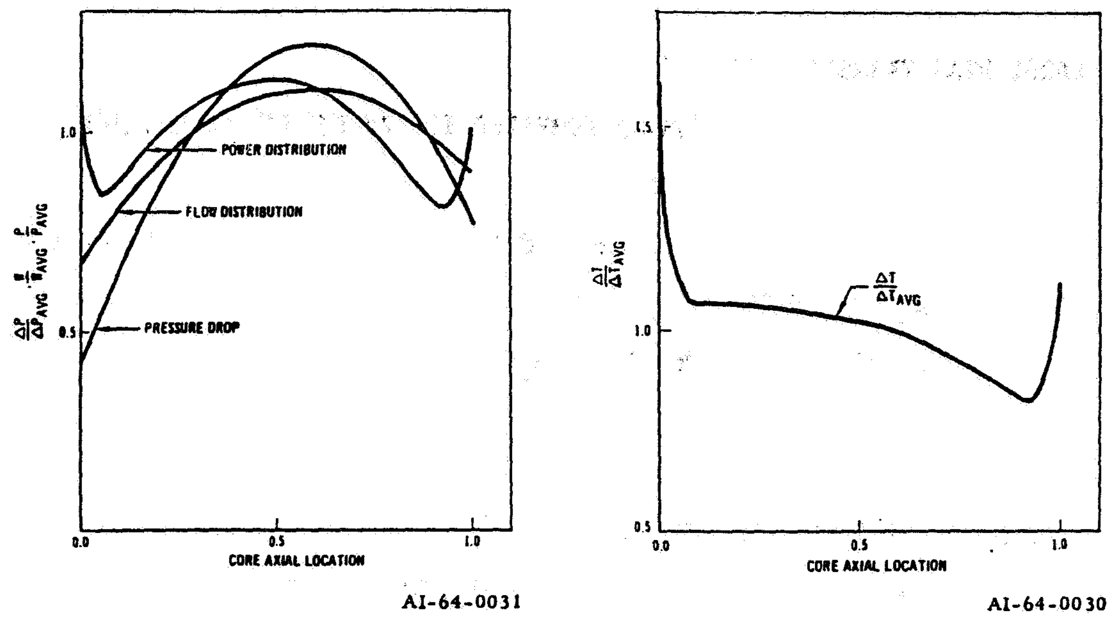


MAKING FUELED WIRE

- Obtain 8 mil Tungsten Wire
- Braid Wires Into 125 mil diameter Tube
- Obtain 4 mil Size UN Fuel Particles
- Coat Fuel Particles With Tungsten
- Fill Braided Tube With Coated Fuel
- Vapor Deposit Tungsten on Filled Tube
- Swage Tube From 125 to 75 mils
- Draw to 35 mil diameter Finished Wire

Figure 8

CORE POWER, FLOW, PRESSURE DROP, AND TEMPERATURE DROP DISTRIBUTION



Rockwell International
Rocketdyne Division

Figure 9

CORE FUELED WIRE SPACING AND SOLIDITY VARIATION FOR UNIFORM H_2 TEMPERATURE RISE

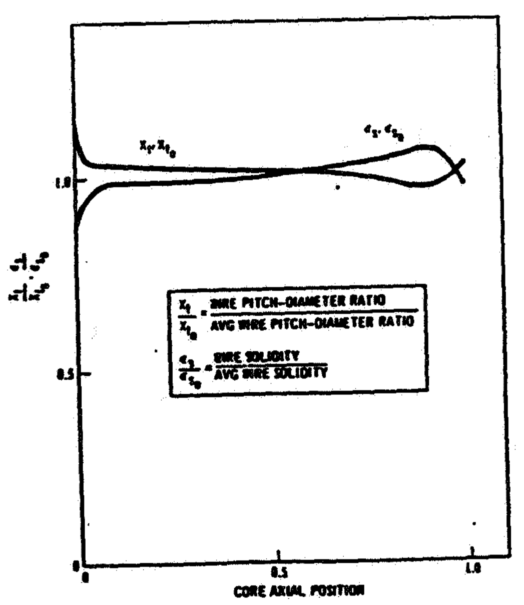


Figure 10

ADVANTAGES OF WIRE CORE REACTOR

- LARGE HEAT TRANSFER AREA
  - 570 FT^2/FT^2 (19 CM^2/CM^3) COMPARED TO 120 FT^2/FT^3 (4 CM^2/CM^3)
- LARGE HEAT TRANSFER COEFFICIENT
  - AVERAGE FILM COEFFICIENT = 8500 BTU/FT^2-HR-·F (4.8 W/CM^2-·C)
- RADIAL FLOW DIVERGENCE
  - MOVE HEAT TRANSFER TO OUTER HOTTER WIRES
- AXIAL POWER SHAPING
  - AXIAL SPACING BETWEEN FUELED WIRES
- SEPARATION OF FUEL AND STRUCTURE
  - RELIES ON THE WIRE CLADDING FOR STRENGTH (NOT THE FUEL)
- SHORT HEAT PATH IN WIRE
  - LOW CENTER TO SURFACE TEMPERATURES
- COMPATIBLE FUEL, CLAD, AND PROPELLANT
  - UN, W, AND H_2 COMPATIBLE AT ELEVATED TEMPERATURES
- GAS LOADS CANCEL
  - RADIAL GAS FLOW RESULTS IN CANCELING OF GAS LOADS
- HIGH SPECIFIC IMPULSE
  - ISP = 930 SEC
- RESTART CAPABILITIES
  - METALLIC CONSTRUCTION INHERENTLY RESISTANT TO THERMAL SHOCK

Figure 11

N92-11100

DUMBO: A PACHYDERMAL ROCKET MOTOR

Bill Kirk
Los Alamos National Laboratory

Since ROVER/NERVA technology has been ably covered by other speakers, and since the intention of this workshop was to cast a rather wide net, I thought it might be useful to tell you a little bit about a lesser known chapter of nuclear rocket history for a couple of reasons. First, perhaps we might learn something from that history, and second, under certain circumstances it might provide an alternative that would be useful in helping to develop a higher performance nuclear rocket engine.

Dumbo goes back to the very beginning of nuclear rocket technology. The first report on Dumbo, from which I have stolen this title, was written by B.B. McIntyre, R.M. Potter and E.S. Robinson in 1955. In 1957, a somewhat larger report was issued, with roughly the same authors.

My first point is that really there are only very few basic concepts in almost any field, in particular in nuclear rocketry, and we come back to the same things over and over. I think it is worthwhile taking into account the lesson from that, that sometimes it pays us to take a different way of looking at the world. Even though ideas are not new, maybe we can learn something from looking at them in a little bit different way.

Dumbo, like several of the reactors you have already heard about, is what I call a folded flow reactor (Figure 1). While it's not my term, it's one that I like to use because it describes very well the idea that the propellant comes in axially and leaves axially, but during some part of its passage through the reactor it flows in a radial direction.

Figure 1 is one of the very early Dumbo pictures. You may not be able to tell from the figure but this particular design had the cold gas flowing inside the cylindrical fuel sections and flowing radially outward through the fuel and then exiting through the annuli around the various cylinders.

Figure 2 is a typical picture; you have seen similar ones earlier today for related concepts. The reflector is a little bit unsophisticated, being flat plates of beryllium, but in any case, we are going to use one of the series of hexagonal magic numbers of fuel cylinders, 1, 7, 19, 37, 61, 91, etc. in almost any reactor that we put together. This is similar. The hexagons are zirconium hydride and the little double circles in the centers depict an annulus of fuel.

Figure 3 is a similar picture showing the full system with some detailing of the reactor components. The Dumbo system started out with very, very thin fuel elements. In fact, there was talk of using 3 mil corrugated foils, made in the shape of a washer, that were going to be stacked together to form the fuel elements, with flow to be metered by the size of the corrugations between the fuel. People learned fairly quickly that you can't make something like that very well or very repeatably. As a consequence, during the program the fuel elements grew to be a lot coarser as the design progressed.

Figure 4 shows a fuel geometry that we were looking at fairly closely at the end of the program. Each fuel washer would be a few tens of mils thick, separated by what we called the spider made of an unloaded material. For the first planned reactor experiment, the fuel material for Dumbo was to be the more easily fabricated Molybdenum UO_2 cermet, with the premise that later we would be able to use tungsten UO_2 fuel, and have similar geometry advantages with higher temperature materials. Others today have pointed out the advantages of radial flow reactors as compared to axial flow reactors.

The key characteristics of the system (Figure 5) are some that I have mentioned earlier; folded flow, use of fuel washers, large flow area, large surface area, small fuel volume, hydride moderator, and cermet fuel. I am going to be talking to you a little bit later about adapting uranium carbide-zirconium carbide to this particular geometry.

The Dumbo project was canceled in 1959. Figure 6 is an excerpt from the progress report that described the cancellation. Basically what it says is we didn't see a heck of a lot of advantages as compared to the axial flow system. We thought it was going to be very complicated, tough engineering problem to develop the folded flow reactor. We had to put our resources either one place or the other and we chose to put them into the axial flow carbon-based systems.

Let me add as a historical footnote, that the small engineering design team, some three of us, who were working on engineering the Dumbo system, then went to work on other geometries, first on axial flow tungsten UO-2 systems and then began looking at new fuel geometries for carbon-based systems. By 1960, we had defined the parameters for the 19-hole fuel element was that the basis for the rest of the nuclear rocket program.

Let me now suggest some reasons why one might want to go to a Dumbo type system, which I will define as a folded flow washer type fuel system. This is a curve (Figure 7) Gerry Farbman showed you earlier. My version has bands on it rather than single lines. It also has the word on the right that I would ask you to look at very carefully: it says "preliminary" in talking about the possibilities for carbide fuel. But there is a lot of space between the predicted temperature capability of carbide fuel and that of composite and graphite fuel. That space I think forms the carrot that's involved in going to carbide fuel. However, there is also a stick, which has to do with thermal stress and consequently with power density that you can get from a carbide system. These limitations becomes important as the composite fuel fraction of UC-ZrC is increased, and as you go to 100 percent UC-ZrC, the thermal stress resistance decreases further and further.

There is also quite a problem in fabricating uranium carbide-zirconium carbide. Our answer to that back in 1970 to 1972 was to use this particular fuel element (Figure 8), a single hole fuel element as opposed to a 19 hole fuel element for a couple of reasons. First of all it makes it smaller just from a fabrication point of view; it is about 1/12th the size of a 19-hole fuel element. Also it enables reducing the total thermal stress by reducing the element to one flow passage, and one set of fuel meat to go with that congruent passage rather than having several.

We did test a few of these carbide elements in the Nuclear Furnace. Two cells in the Nuclear Furnace held carbide elements; they fragmented rather badly under the power density of the Nuclear Furnace. However, it isn't clear that the fragmenting is a show-stopper.

One reason we worried about thermal stress for the graphite and composite elements is that any thermal stress fracture was a new path for corrosion. With the carbide fuel that's not so much a worry because the carbide fuel has an intensive resistance to corrosion. On the other hand, if the fractures in the fuel elements disturb the flow geometry, there may be real problems. That becomes potentially damaging to the entire core by changing the flow patterns in the core. Certain parts of the core are going to get cooler and certain parts are going to get hotter. You are either going to have to shut down (if you know this is happening through instrumentation readings), or if you don't know it's happening, you are probably going to melt out some parts of the core.

In considering the possible geometries for using uranium carbide-zirconium carbide we can include using it in a folded flow geometry as a washer. I have listed here just a few ideas (Figure 9) about what the fuel elements might look like. These are certainly nothing definitive, because at the time we were working on this before, we were looking at a cermet system, which has different properties. But you can think of a lot of ways that such a fuel might be defined. It's going to depend on interactions between the fabrication and design issues that come up, so that one can choose something that will work in both respects.

I put together a comparison for a 1500 megawatt, that is, a 75 K thrust reactor of some for the characteristics of a carbide Dumbo system and a couple of other systems (Figure 10); one a Rover fuel and the other a particle bed fuel. The assumptions that I made are listed at the bottom. I don't claim any great precision for these numbers; I think they reflect the assumptions that were made. But they give you some idea of the kind of characteristics that will be typical for these fuel geometries.

Based on the assumptions, fuel volume is different because we assume higher power density for the Advanced Dumbo and for the particle bed. On the other hand, the surface area, the heat transfer surface area, is increasing to the right in the figure because the surface-to-volume area of the fuel is increasing as we go to the right. Also, there is some variation in the flow area in the system; the flow area per unit volume of fuel is increasing. It doesn't necessarily always increase, because the volume of fuel depends on the power density assumptions. If you believe, as I do, that the temperature/lifetime performance of a nuclear rocket engine is limited by mass loss from the fuel, not from an absolute temperature limit (but from mass loss either associated with corrosion or with evaporation of the various components), you might conclude that there could be an optimum fuel surface-to-volume ratio. This would be one that gives you the closest match between maximum temperature of the fuel and the exit gas temperature, but that limits the surface area that is exposed to corrosion because, as far as we know, total corrosion rates depend on surface area.

The corrosion can be characterized as a loss rate per unit surface area, and of course, this may depend on the surface temperature or the interior temperature of the fuel at a given location. I don't assert that that's true; I just say that it's a possibility that we need to look at in optimizing the design for the UC-ZrC system.

So the characteristics of the Advanced Dumbo (Figure 11) are that it offers an alternative fuel geometry. By having a higher surface to volume ratio of the fuel, it offers reduced thermal stress, as other people have suggested for this kind of geometry. I say it eases fuel fabrication for the UC-ZrC system with a question mark, because I'm not sure that it does. Compared to a particle bed system, it has a defined fuel passage; that is, once the coolant gets into the fuel it has only one place to go. And also compared to particle bed system the fuel is radially self-supporting.

I am a little amused that here I am saying that the fuel supporting itself is an advantage and other people are saying the fuel not supporting itself is an advantage, and I don't know which one of us is going to turn out to be correct.

The Advanced Dumbo system also has certain key design issues (Figure 12); I use the word issue to mean problem. Flow balancing has been talked about before, Dumbo is going to have the same need for an orificing system at the inlet of the fuel that any other fuel system or any other fuel geometry does. Folded flow systems, I think, have a considerable amount of engineering complexity compared to axial flow systems. Fuel fabricability in this particular geometry is an issue. As to thermal stress, I'm not sure that this geometry generally solves the thermal stress problem. And then axial support, particularly at the hot end, is going to be a problem, if indeed we are to be able to get a very, very high temperature propellant out of this system. These problems, of course, are shared to various degrees by similar concepts.

I mentioned earlier that there was another historical footnote, after the end of the nuclear rocket program. There was a rather bitter article in the December 1975 <u>Analog</u> science fiction magazine, in an article called "Atomic Rockets." It pointed out how stupid and conservative the administrators and engineers in the NERVA program had been for refusing to take the Dumbo system as the route to develop and said our lack of imagination had led to the termination of the nuclear rocket program. This article is one of my prize possessions, and while I would be glad to let you look through it please leave it when you do. By the way, I don't mean to imply that I agree with any part of the article.

MR. ZUBRIN: As I recall, the original Dumbo had some very exotic microdynamics of the propellant going to plates and so forth,
and they maintain the laminar flow --

MR. KIRK: The velocity was slow because you have a very, very high flow area compared to an axial flow system. By going folded flow you multiply the flow area by a fairly large factor.

MR. ZUBRIN: You are also in the laminar flow regime.

MR. KIRK: I don't know. It doesn't matter a heck of a lot to me whether it's laminar flow or turbulent flow. I am always going to have to have metering in the front end in order to get the right amount of fluid into the right passages.

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Bill Kirk
<u>Dumbo</u>

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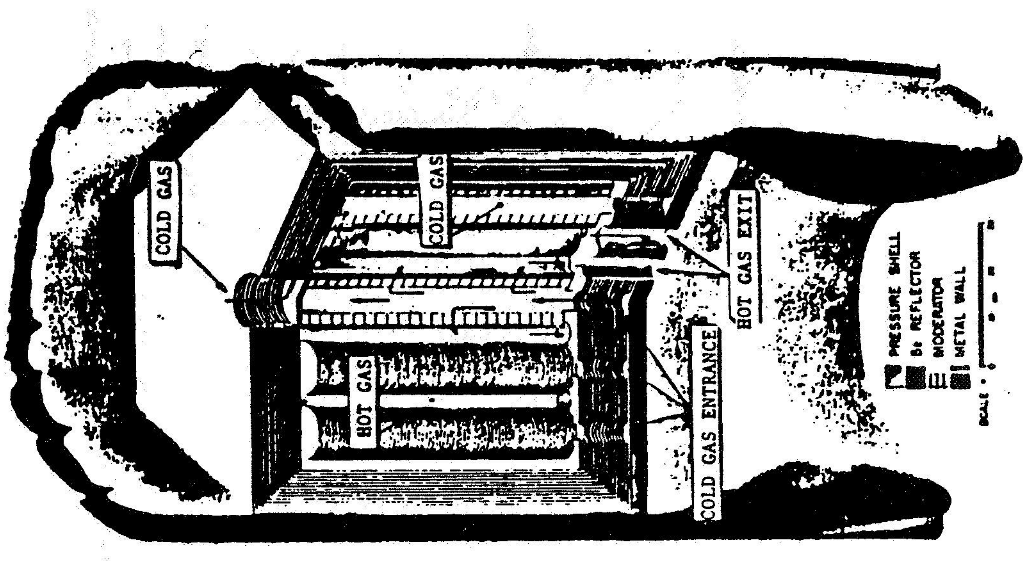


Typical Dumbo reactor

Figure 1

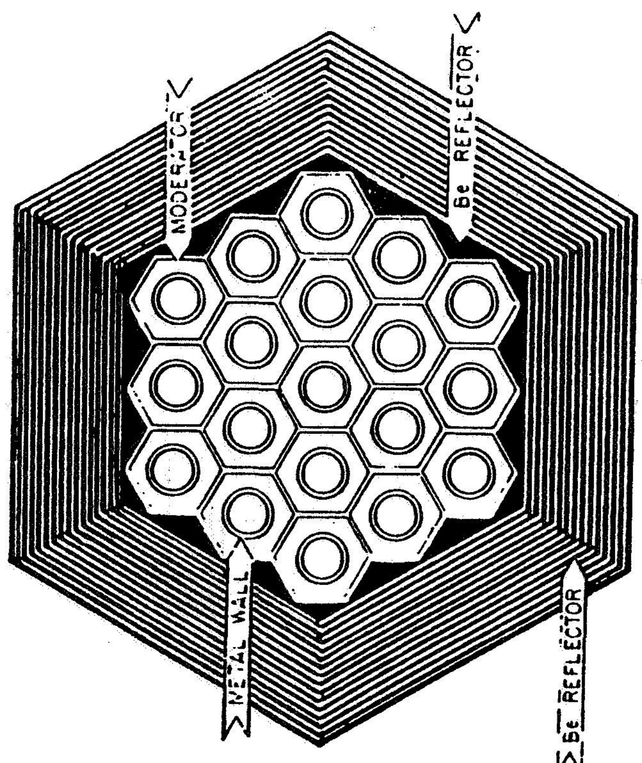
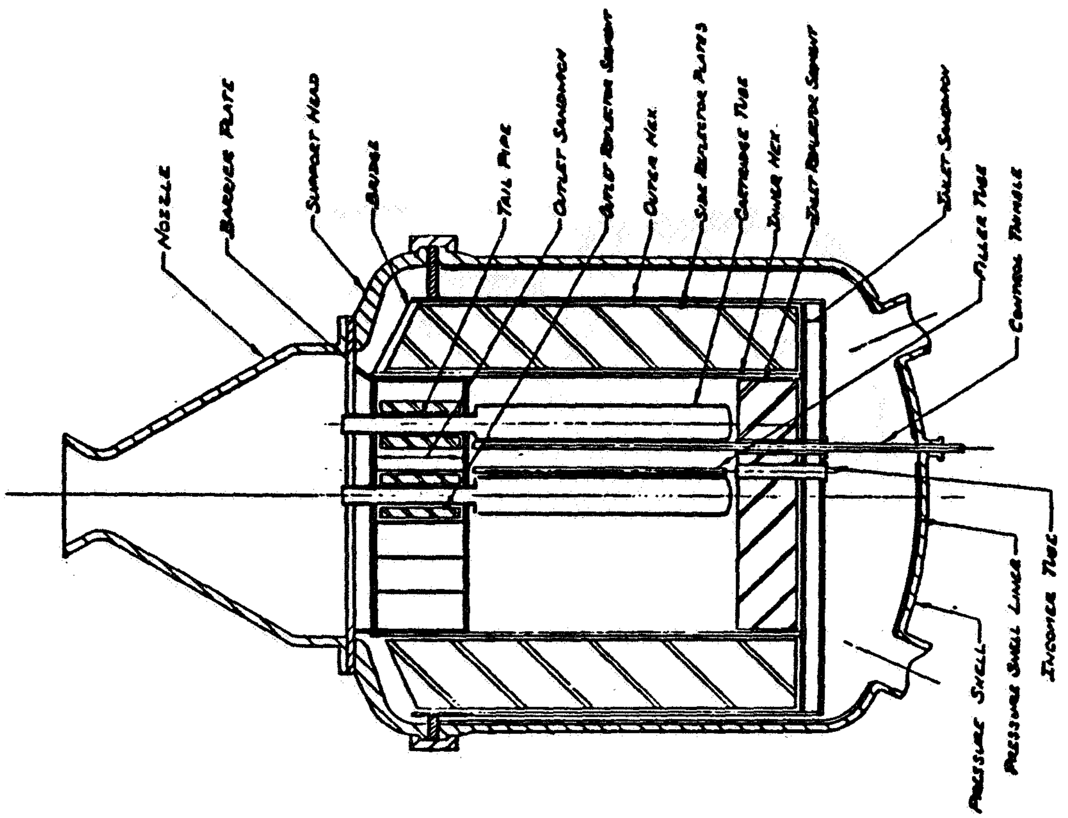


FIGURE 9-8
MODEL C DUMBO

SCALE 0 10 20 30 cm.

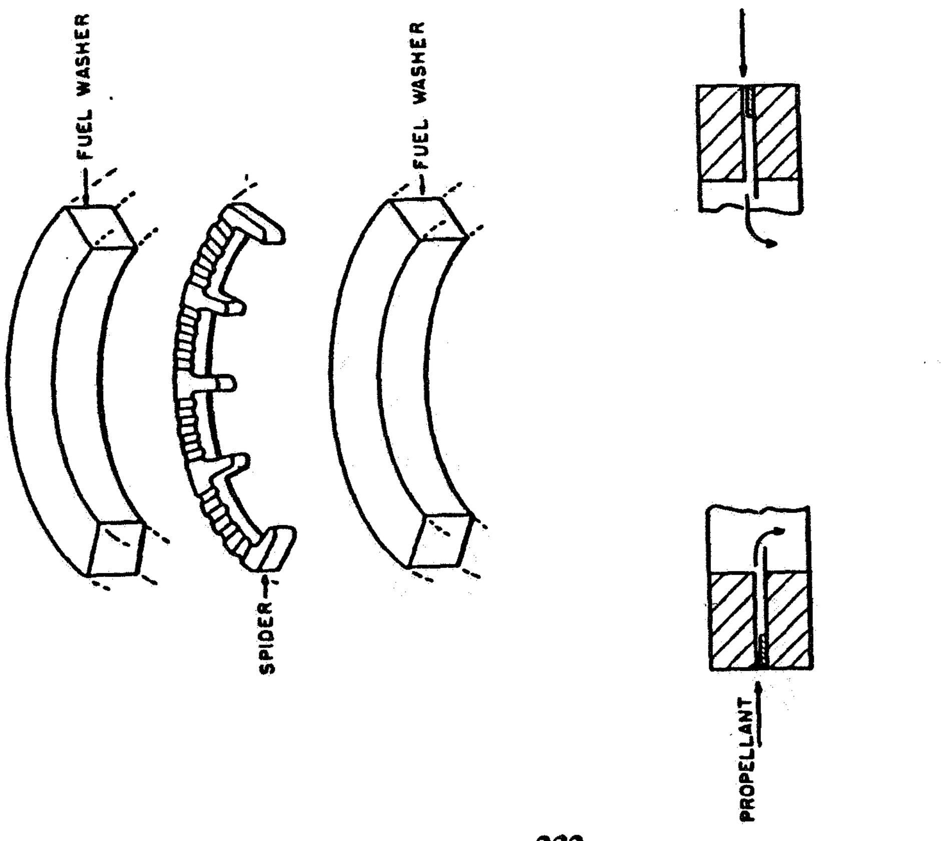
295

Figure 2



Nomenclature of Components of Dumbo-A

Figure 3



Proposed Dumbo-A Fuel Elements

Figure 4

DUMBO

KEY CHARACTERISTICS

- FOLDED FLOW
- FUEL WASHERS
- LARGE FLOW AREA
- LARGE FUEL SURFACE AREA
- SMALL FUEL VOLUME
- HYDRIDE MODERATOR
- CERMET FUEL (1959)
- UC•ZrC FUEL (1990)

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Figure 5

DUMBO

General Status

After reviewing the current status of the Dumbo (Molybdenum) reactor, the decision has been made to cancel the project. The two major factors which prompted this decision were (a) a recognition that the Dumbo design as it now exists does not offer any appreciable advantages in terms of performance or light weight over KiWi designs, and (B) the complexity of the design and the need for many precision parts, coupled with the difficulty of making meaningful component tests, leads to the conclusion that the development of a successful reactor would be a long and arduous task. It should be noted that the foregoing remarks refer in particular to a Mo-UO_2 fuel system. There is a possibility that with tungsten in place of molybdenum higher temperatures may be achieved than in graphite reactors. Consequently, an evaluation of tungsten-based fuel elements, in terms of their high-temperature capability, will be continued.

CHAPTER 4

QUARTERLY STATUS REPORT OF LASL ROVER PROGRAM FOR PERIOD ENDING SEPTEMBER 20, 1959

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Figure 6

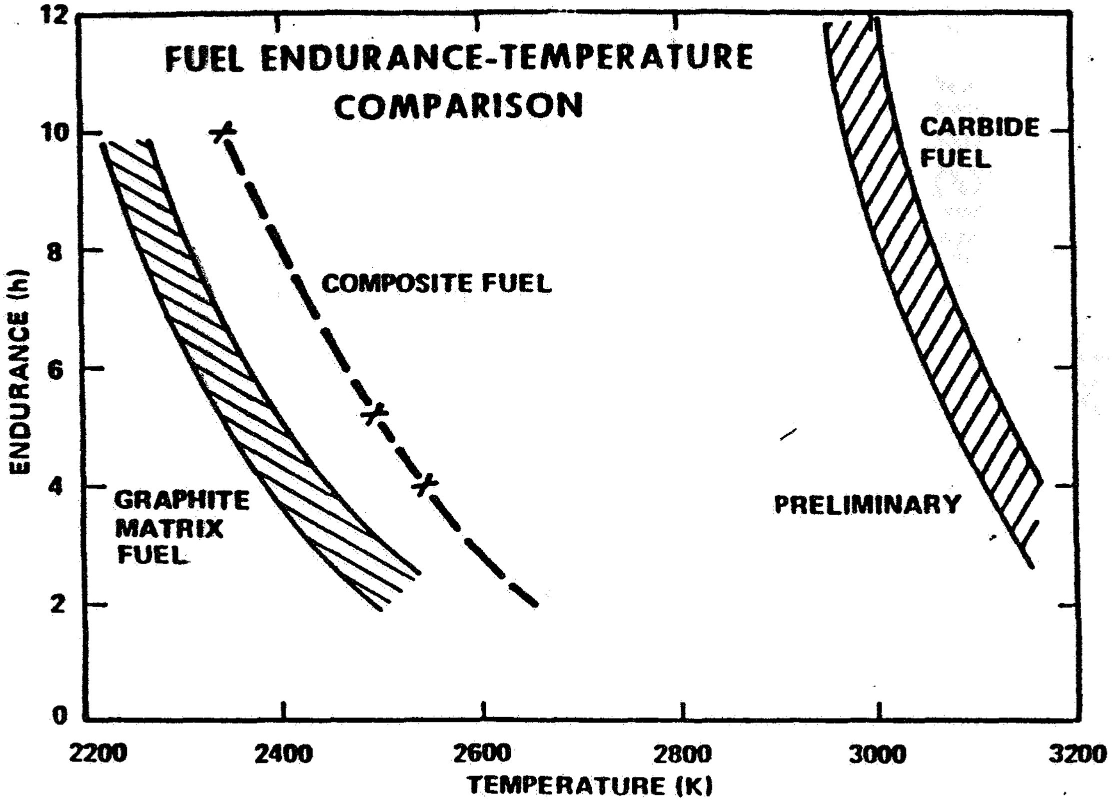
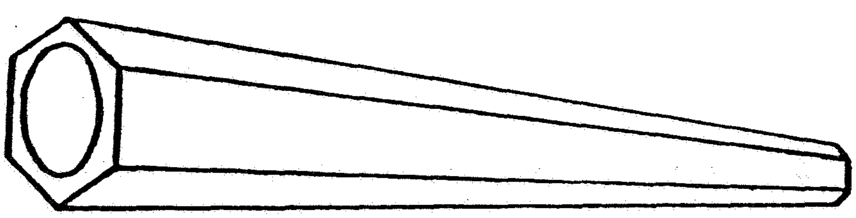


Figure 7



Full-length view of (U,Zr)C (carbide) fuel element. Length, 0.64 m; across flats, 5.5 mm; coolant-channel diameter, 3.2 mm; and carbon/metal atom ratio, 0.8 to 0.92.

Figure 8

ADVANCED DUMBO

POSSIBLE FUEL CONFIGURATIONS

1. FLAT FUELED WASHERS ALTERNATING WITH GROOVED UNLOADED WASHERS.
2. GROOVED FLAT FUEL WASHERS
3. RADIALLY SPLIT FLAT WASHERS

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Figure 9

ESTIMATED FUEL PARAMETERS FOR A 1500 MW REACTOR

	ROVER FUEL*	ADVANCED** DUMBO	PARTICLE +
FUEL VOLUME, m^3	0.6	0.2	0.1
FUEL SURFACE AREA, cm^2	2.9×10^6	5.2×10^6	8.4×10^6
FUEL FLOW AREA, cm^2	1.4×10^3	4×10^4	1.6×10^4
FUEL SURFACE/ VOLUME RATIO, m^{-1}	480	2600	8400

*Based on 19-hole element with 1.1 MW/element (2500 MW/m^3)

**Assumes 30-mil fuel and 10-mil flow passage, element $\bar{D} \cong 3$ in., fuel thickness $\cong 0.5$ in. 7500 MW/m^3

+Assumes 500 μ particles, 15,000 MW/m^3, 70% packing

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Figure 10

ADVANCED DUMBO

- OFFERS AN ALTERNATIVE FUEL GEOMETRY FOR UC-ZrC FUEL
- REDUCES THERMAL STRESS
- EASES FUEL FABRICATION?
- DEFINED FUEL FLOW PASSAGE
- RADIALLY SELF-SUPPORTING FUEL

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Figure 11

ADVANCED DUMBO

FUNDAMENTAL DESIGN ISSUES

- FLOW BALANCING
- ENGINEERING COMPLEXITY
- FUEL FABRICABILITY
- FUEL THERMAL STRESS
- AXIAL SUPPORT

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Figure 12

N92-11101

PELLET BED REACTOR FOR NUCLEAR PROPELLED VEHICLES: I. REACTOR TECHNOLOGY

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Good afternoon. I am going to talk today about the pellet bed reactor concept. First, I would like to acknowledge my coauthors. Nick Morley is a graduate student now in the process of deciding whether or not to pursue his dissertation in nuclear propulsion. I would like also to acknowledge Bill Haloulakos from McDonnell Douglas. He kindly volunteered to do mission analysis associated with the pellet bed reactor, and he is going to present that analysis today.

Historically the pellet bed reactor concept was developed as part of the Multi-Megawatt Program (MMW). It was a joint project between SAIC and the University of New Mexico. The principle investigators on the project were David Buden and myself; one of the people who did the technical development is here also, Jim Mims from S-Cubed.

Figure 1 is a simple outline of the integration of the pellet bed in the rocket platform, and you can see the reactor, (a hot shield is inside the reactor), a shadow shield, and a bank of propellant tanks. Also, we have the Mars transfer vehicle and the crew compartment; the shield would be optimized between the shadow shield, the hot shield, and also the biological shield.

In this vehicle design we tried to satisfy the five REM per year reactor radiation requirement. This is what the reactor might look like (Figure 2). In the integrated nozzle, the coolant comes in and cools the structure. It also cools the reflector and the structure and then goes to the hot shield, flows down, cools the axial reflector, then flow in the annulus outside the core, flows radially through the core, and then axially down the center. The core is just one annulus. The dimensions for the core give you an idea of size, the diameter is about 70 centimeters and the height is about 1.3 meters. The advantage of this concept is that this kind of reactor is not neutronics limited, so you can increase the height-to-diameter ratio without really causing problems with neutronics.

The critical issue here is the thermal hydraulics. So by reducing the path of the flow we reduce pressure losses. By using pellets, which are about 1 cm diameter, we also increase the surface to diameter ratio.

To give you a point of reference, we can operate at about 3 megawatts per liter, compared to NERVA's 1.8 megawatts per liter. This is less than 4 to 4.3 megawatt per liter at which the particle bed would operate.

The fuel pellets (Figure 3) consist of a graphite matrix, with microspheres dispersed through the matrix. You can adjust the ratio of the fuel to the graphite in your design optimization of the neutronics. We use zirconium carbide coating here to reduce the diffusion of the graphite and interaction with the hydrogen. This is a major problem as some of you are aware. Also there is a problem with the losses of graphite from zirconium carbide. The zirconium carbide here doesn't provide any structural strength, just better compatibility with the hydrogen.

These are just our thoughts about how important the compatibility problem is (Figure 4). Most of the concepts we listened to this morning use hydrogen. There is a similar problem here with hydrogen. It's having graphite and hydrogen for a really long period of time. We didn't have any data that showed they would be compatible for a year or more.

We're using zirconium carbides because it could use a eutectic that would reduce the melting temperature of zirconium carbide by about maybe 20, 30 percent or something like 200-300 degrees. However, it's a good choice compared to niobium carbide because zirconium carbide doesn't lose graphite as fast or doesn't lose as much graphite in contact with hydrogen as niobium carbide.

By chemical vapor deposition, you can apply zirconium carbide close to the operating temperature. During operation you will not have any stress in zirconium carbide. However, the particle will be under compression at startup.

To show you some of the comparison, Figure 5 is the zirconium carbide and niobium carbide in a hydrogen atmosphere at constant temperature, and this graph shows you how much graphite you lose. These data were published by the Soviets at a meeting in May, and show that in the 3,000 to 6,000 second range, you can lose a lot of carbon from niobium carbide compared to zirconium carbide. But as to the effect of these carbon losses on zirconium carbide strength, I haven't seen anything to quantify that, but it remains an issue.

The microsphere is a trisosphere. Because of the fact that with nuclear thermal propulsion, we only operate at very high temperature, then we cannot use uranium-zirconium carbide as was proposed in the original particle bed. What I am proposing here (Figure 6) is using uranium carbide-tantallum carbide, (although I don't like it because of the neutronics, the high absorption concept here for tantallum), or uranium carbide-niobium carbide.

To my knowledge this technology needs to be developed; we know very little about it and it's just the typical try to design to have pyrolytic graphite. The graphite here reacts with niobium carbide, with uranium carbide-niobium carbide, also with uranium carbide and tantallum carbide, and for eutectic. The reduction in temperature here is about 200 degrees in each case. We can still operate at about 3,000 K, which is not the case with uranium-zirconium carbide.

The thickness of the pyrolitic carbon here is about 15 - 20 microns to absorb the damage that will be caused by the fission fragments, It is then surrounded by high density graphite, and also it has niobium carbide or tantallum carbide outer coating; this is really the pressure vessel for the microsphere. The idea here is to retain all the fission gases inside the sphere. The porosity in the fuel as well as in the pyrolitic carbon will provide the means to accommodate these fission gases without much increase in pressure. Of course, the design has yet to be done and optimization for the thickness of different layers have to be done.

An important issue will be how to coat these microspheres (Figure 7). As I said before, it has to be designed to accommodate the stresses due to the buildup of the fission fragments, particularly since you are talking now about five to ten atom percent burnup, which is a high burnup for this kind of microspheres.

Another option or alternative that we will be proposing today is to consider refueling in orbit; we believe that this concept provides the means to refuel in orbit. So you will have to make trade studies such as, designing the reactor to operate to a 5 atom percent burnup and refueling it versus designing the fuel for 10 atom percent burnup and not refueling it. I cannot tell you more about this because it is now in the process of getting patented.

You have seen this graph before (Figure 8), and we think that the operational condition would be in this range shown. And as I said, the zirconium carbide, uranium carbide-zirconium carbide seems out of question for nuclear thermal propulsion because you will not be able to get 3,000 degree Kelvin with it. It might be good for nuclear electric propulsion, but not here. So the only alternative you have is the niobium carbide and tantallum carbide; the temperature here is for the single phase. For the eutectic, just reduce that by roughly about 200 degrees Kelvin; so we are talking about, in this range, maybe 3,500 to 3,700 degrees Kelvin. So if you operate at an exit temperature of about 3,000 degrees Kelvin, the maximum fuel temperature would be 3,100, giving a margin of about 400 to 600 degree Kelvin below the melting temperature.

Figure 9 is just additional information about the different carbides or coatings that you can use to replace the niobium carbide. As I said, we know nothing about niobium carbide, but we do know about silicon carbide. Above 1,800 degree Kelvin you have this amoeba effect where the uranium will diffuse out of the kernel through the silicon carbide; silicon carbide is really out of question above 1,800 K (Figure 10 & 11).

At about 2,000 degrees, you have the same problem with zirconium carbide, so zirconium carbide should not be used above about 2,000 degrees Kelvin. This puts a lot of limitation on whether zirconium carbide would be the choice. And we don't know, with a similar scenario, what will happen with niobium carbide.

In my opinion, at the core of reactor design for nuclear thermal propulsion is the fuel material development. Without the fuel, we cannot build the system. There are a lot of issues dealing with that development that need to be investigated, ranging from compatibility to fabrication, to dealing with new material, with which we have not dealt before.

I will show you some of the results that General Atomic has published as part of their high temperature gas cooled reactors (Figure 12). In this case, horizontally you have uranium carbide in contact with uranium-zirconium carbide. Vertically is uranium content in weight percent. This is the interface, and as you see here, after operating for about 50 hours at about 2,100 Kelvin, the uranium diffuses up to about 45 microns into the zirconium carbide.

At the interface, the content of the uranium is close to 28 weight percent. This is a lot of uranium, because you will have fission, and also you will damage the zirconium carbide. This becomes worse if you operate either for a longer period of time or at a higher temperature.

Here it goes up to 70 percent if you increase the temperature by 200 degrees, so 70 weight percent will be uranium at the interface, and then it will penetrate up to about 1500 microns. If this is not a problem, I don't know what else would be a problem. So this is one issue.

The second issue is in the stress analysis (Figure 13). Recently, we did some work on the thermal stress analysis of the particle bed. In the beginning of the work we had to find out how much we know about the failure pressure of zirconium carbide. The scattering of the data, varies between 300 to 1,000 megaPascal. So to design this kind of microspheres, we really have to get better data on the structure and strength of these materials.

Now, going back to the pellet bed reactor, these are the parameters (Figure 14) that we used in our mission analysis today. The nominal power is 1,500 megawatts thermal. The dimensions for the core are shown. The power density is about 3 megawatts per liter. The diameter of the central channel is about 20 centimeters using hydrogen as coolant. The maximum fuel temperature is 3,100 degrees Kelvin, the maximum core exit temperature 3,000 degrees Kelvin, and the core inlet temperature is 120 degrees Kelvin. The inlet temperature to the reflector is about 70 to 80 degree Kelvin.

The coolant flow rate is 32 kilograms per second. This compares to NERVA's rate of about 24 kilograms per seconds, which makes for the difference in the specific power here. The specific mass for the reactor is 1 kilogram per kilowatt, excluding the shields, which is one ton. There are the two kinds of fuel proposed, pending an investigation, uranium-tantallum carbide -- uranium carbide-tantallum carbide, uranium carbide-niobium carbide. I couldn't find anything that would be better than these materials for these temperatures.

Why should we consider pellet bed reactor (Figure 15)? It is modular. You can build the reactor smaller or bigger. You can have more than one unit. The particle is self-supporting. I consider that an advantage, because it will enable refueling in orbit. We can get high thrust because of high specific power and also high specific impulse because we will be operating at about 3,000 degrees K. Then, it makes full use of the available technology for the fabrication of the particle, again pending knowing more about the fabrication and the high temperature material properties. But in the German-AVR Program we are building similar pellets. The only difference here is that the pellets are optimized for 1 centimeter in diameter, the pellets for the AVR were about 6 centimeters in diameter.

As I said, it provides the possibility for refueling in orbit, which would be a great advantage. I am not proposing a dual mode here, but if the option is to go with nuclear electric propulsion, you can use the same reactor design for that or, if the option is to go to nuclear thermal propulsion, the reactor design could also be used for that.

It is designed so that in a case of loss of flow, the conductive/radiative passive decay heat would be sufficient to cool the system, because of the high thermal conductivity of the graphite.

It has been designed for pulsed and continuous modes of operation. It also has a redundant mechanism for the control. The concept has two independent control mechanism, each of which would be sufficient to operate the system. We have the typical control drums on the periphery of the core and also we have safety rods. We think that it has a relatively low development cost. However, we have to quantify that.

As to the safety features (Figure 16), it satisfies being subcritical during water immersion, assuming that the water fills all the holes inside the core. It has two independent safety systems, 24 control drums and five safety rods, located about 19 centimeters from the center of the core. It could be refueled in orbit. It has passive decay heat removal. The design of the pellet, given that we must further investigate the material and properties, provides a safe containment of the fission fragments. It has a high height-to-diameter ratio, which provides a small cone angle for the shield; this is very important when you look to this to optimize the shield mass.

How long will development take (Figure 17)? My wild guess, is that it will take about 10 to 16 years to flight qualification, at the cost of about $3.1 billion. From what I have seen today, this doesn't look bad at all.

Well, I am running out of time, but you can read Figure 19. I think I covered all of these key issues. This is what I think of the status of technology (Figure 20), except for the fact that we know how to build these reactors. We have been doing that for so many years, as well as we know -- the best choice for shielding. The rest of the technology in between 1 and 3.

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Mohamed El-Genk
Pellet Bed Reactor Concept

Acknowledgments

The development of the PeBR for electric and thermal nuclear propulsion missions has been performed by the University of New Mexico's Institute for Space Nuclear Power Studies. The System Trades and Performance Studies were performed at McDonnell Douglas Space Systems Company, Huntington Beach, CA; the contributions of T.M. Miller and J.F. LaBar are hereby acknowledged.

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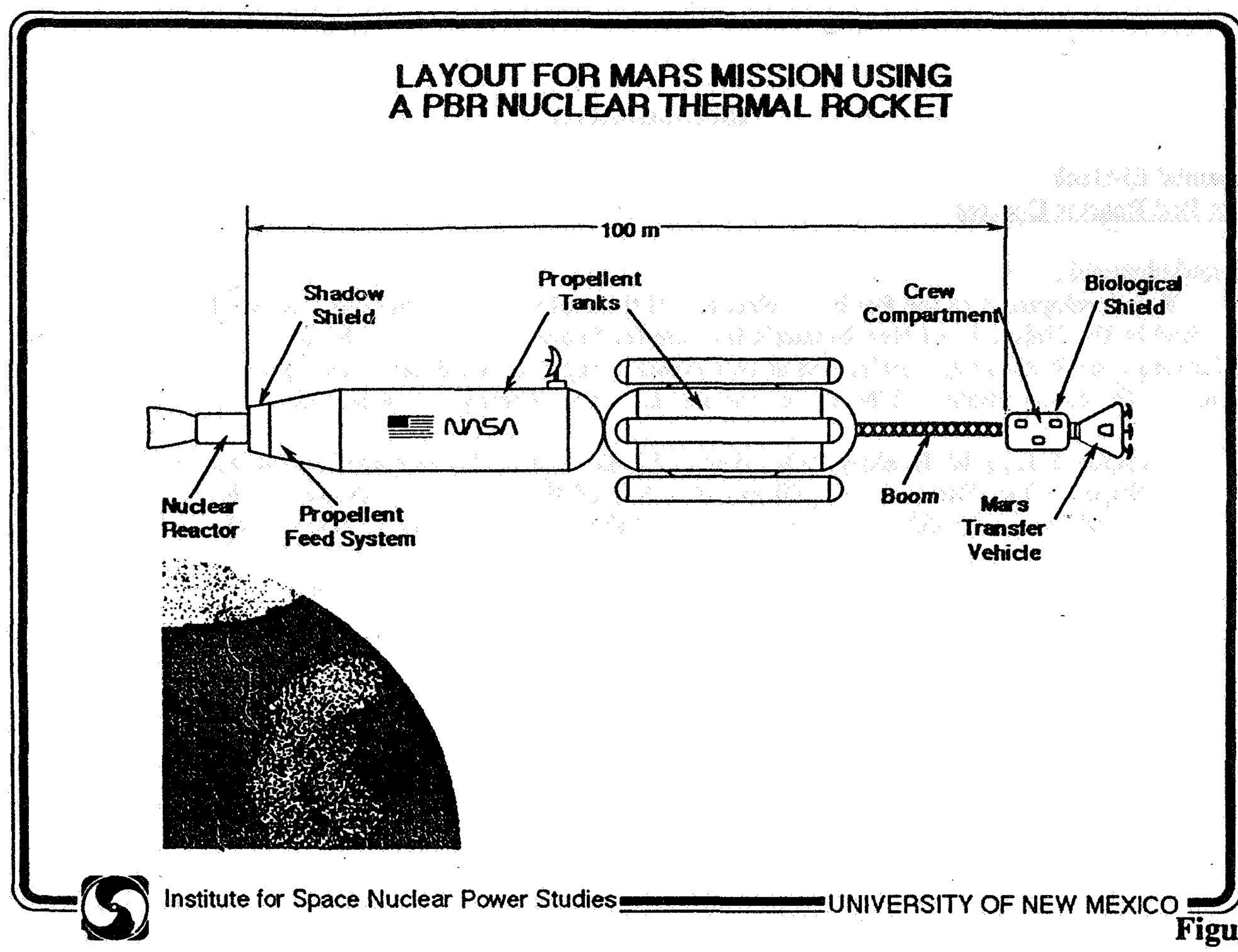


Figure 1

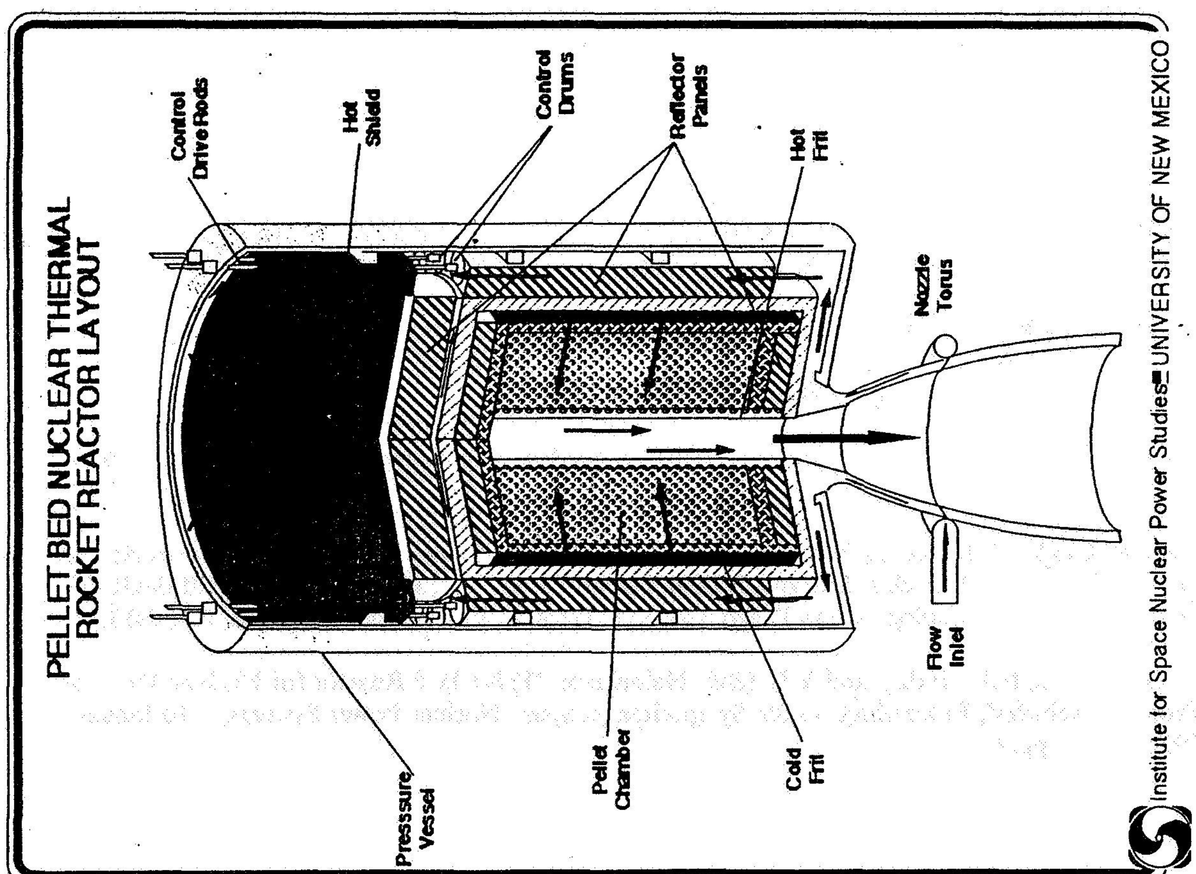


Figure 2

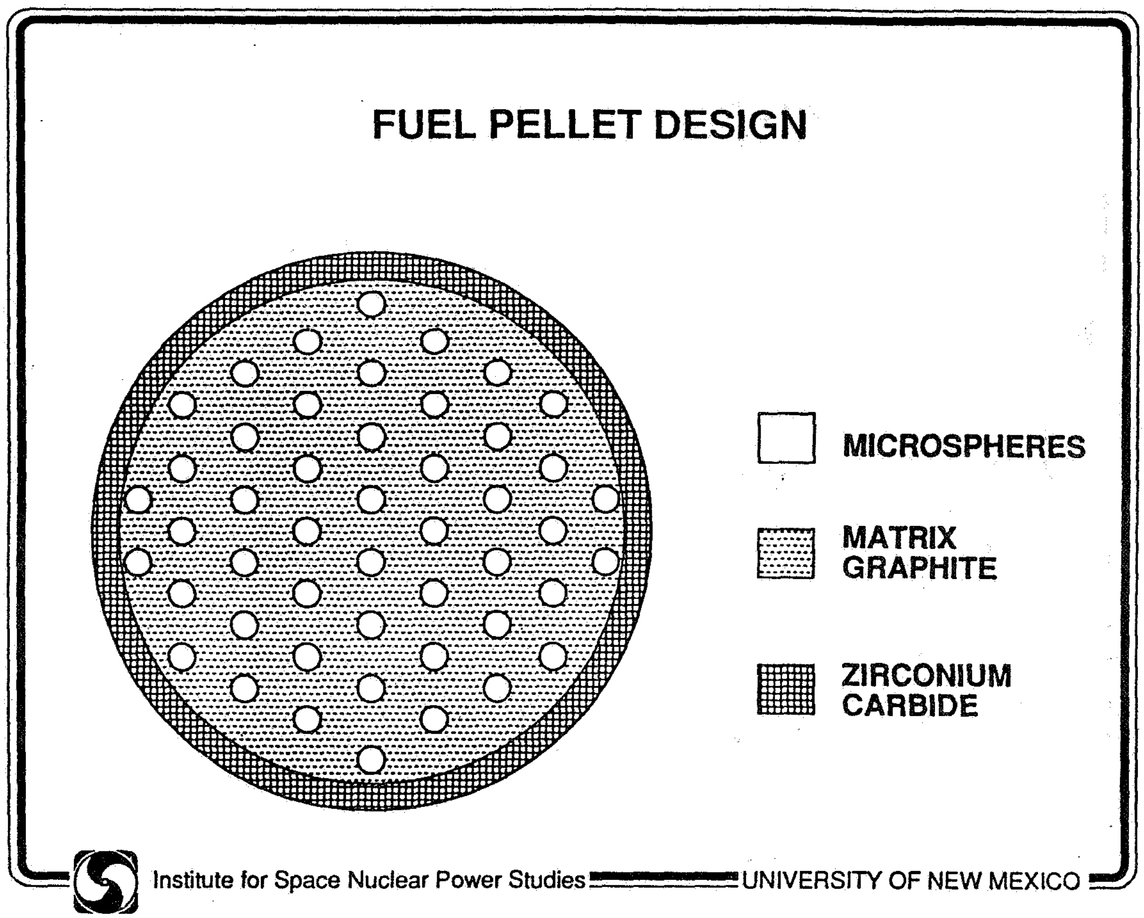


Figure 3

MATERIAL COMPATIBILITY

HYDROGEN-GRAPHITE REACTION

- o HYDROGEN COOLANT REACTS WITH GRAPHITE AT HIGH TEMPERATURES TO FORM:
  - -- METHANE
  - -- OTHER HYDROCARBONS
- o THESE REACTIONS CAUSE CONTINUOUS LOSS OF GRAPHITE FROM THE FUEL PELLETS.
- o A SOLUTION TO SLOWDOWN THE GRAPHITE LOSSES IS TO COAT THE FUEL ELEMENTS WITH A THIN LAYER (FEW MICRONS THICK) OF ZrC.
- o ZrC COATING IS A GOOD CHOICE BECAUSE:
  - -- HYDRIDING OF ZrC IS NEGLIGIBLE AT INTERMEDIATE -TO- HIGH TEMPERATURES (<2500K), BUT INCREASES WITH TEMPERATURE.
  - -- DIFFUSION COEFFICIENT OF GRAPHITE IN ZrC IS SMALL, HENCE SLOWING DOWN GRAPHITE LOSSES.
  - -- ZrC CAN BE APPLIED AT OR NEAR OPERATING TEMEPRATURE OF THE FUEL ELEMENTS, THUS ELIMINATING THERMAL STRESSES IN THE COATING DURING REACTOR OPERATION.
  - -- ZrC IS A BETTER CHOICE OVER NbC BECAUSE OF ITS EXCELLENT ADHESION PROPERTIES TO GRAPHITE; LOWER NEUTRON ABSORPTION; AND LOWER CARBON LOSSES AT HIGH TEMPERATURES IN HYDROGEN.

Institute for Space Nuclear Power Studies UNIVERSITY OF NEW MEXICO

Figure 4

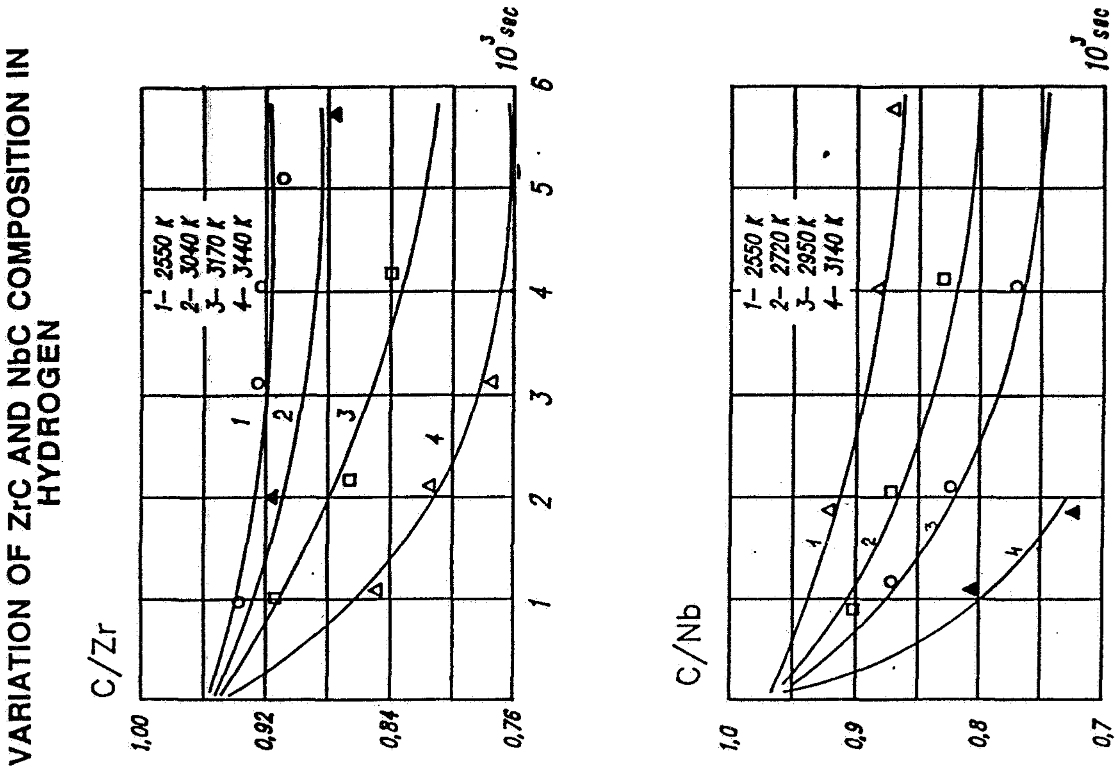


Figure 5

FUEL MICROSPHERE FOR PELLET BED NUCLEAR THERMAL ROCKET

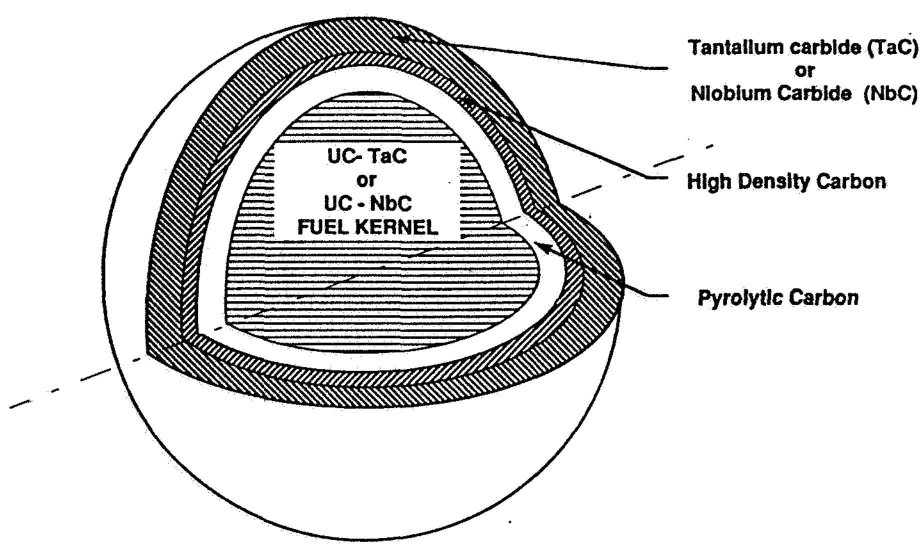


Figure 6

COATING AND FUEL MATERIALS IN MICROSPHERES

COATING DESIGN

IN DESIGNING A FUEL MICROSPHERE, IT IS IMPORTANT TO CHOOSE A COATING THAT HAS:

- COMPARABLE THERMAL EXPANSION COEFFICIENT TO THAT OF THE FUEL
- A THICKNESS GREATER THAN THE FISSION PRODUCT RECOIL RANGE
- STRONG ENOUGH TO ACCOMODATE STRESS DUE TO FISSION PRODUCT BUILDUP
- HAS HIGH THERMAL CONDUCTIVITY FOR REMOVING HEAT FROM THE FUEL MICROSPHERE

Figure 7

MELTING POINTS IN QUASIBINARY SYSTEMS UC-ZrC, UC-NbC, AND UC-TaC

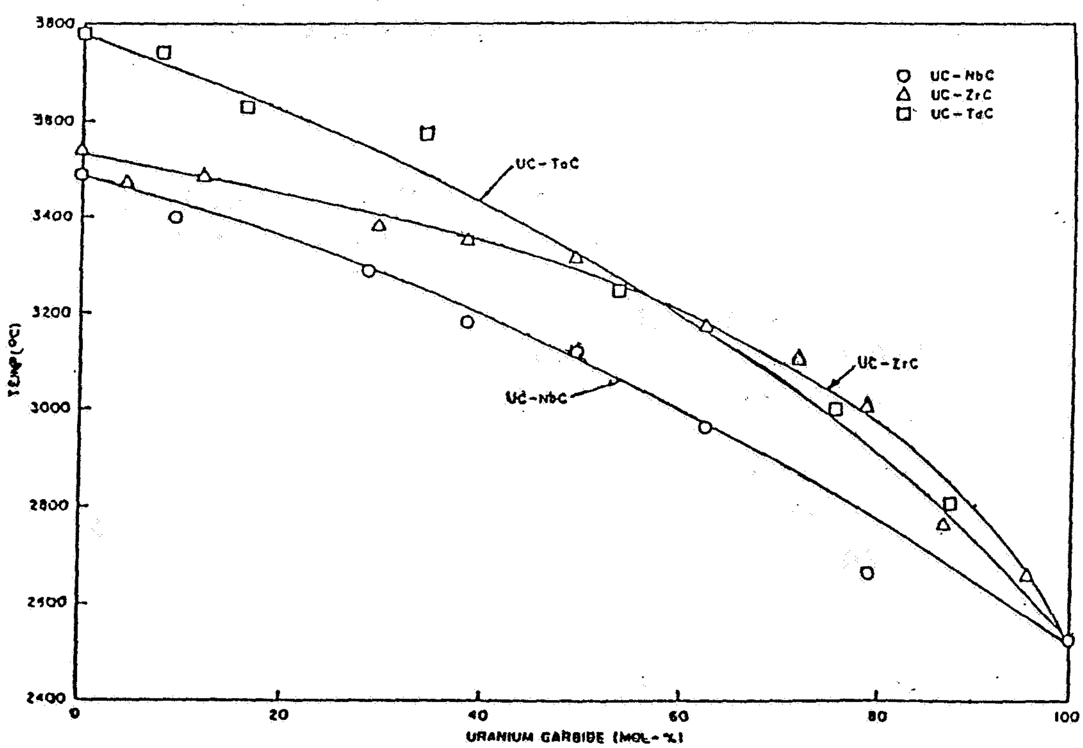


Figure 8

COATING PROPERTIES

COATING COMPOUND	DENSITY (g/cm^3)	THERMAL CONDUCTIVITY (W/cm K) @ 1600 K	FISSION FRAGMENTS RANGE (μ m)	THERMAL EXPANSION COEFFICIENT x 10^6 (K^{-1})
C	3.01	.357	10	3.0 - 5.07
SIC	3.21	.30	11	10.2
ZrC	6.40	.38	9	6.3 - 8.5
NbC	7.32	.721	7	7.1 - 9.0

Figure 9

MATERIALS COMPATIBILITY (CONTINUED)

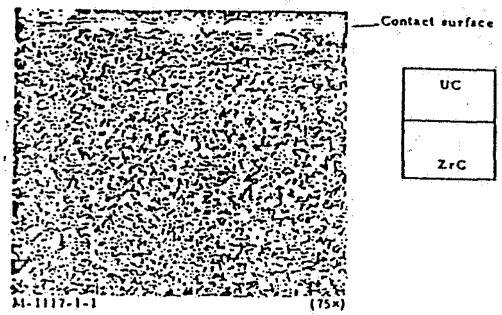
(2) DIFFUSION OF U THROUGH ZrC COATING

- THERE HAS BEEN EVIDENCE TO SHOW THAT THE UC FUEL, WHEN HEATED ABOVE 2073 K URANIUM WILL MIGRATE THROUGH THE KERNEL AND INTO THE ZrC LAYER . THIS MIGRATION WILL CAUSE FISSIONING IN THE ZrC LAYER LEADING TO ITS DESTRUCTION AND FAILURE OF FUEL MICROSPHERES.

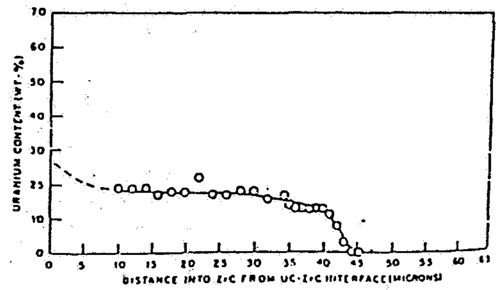
- THE RATE OF URANIUM MIGRATION AND ITS PENETRATION DISTANCE INTO THE ZrC COATING IS A STRONG FUNCTION OF TEMPERATURE AND THE TIME-AT-TEMPERATURE.

Figure 10

MIGRATION OF U FROM UC FUEL INTO 45 MICRONS OF ZrC WHEN IN CONTACT FOR 50 HOURS AT 2073 K



ZrC contacted with UC at 1800° ±20°C for 50 hr. Note that there is no apparent change in microstructure.



Electron-microprobe trace of uranium penetration into ZrC

Figure 11

MIGRATION OF U FROM U FUEL IN 1300 μm LAYER OF ZrC WHEN IN CONTACT FOR 30 HOURS AT 2273 ± 20 K (CONTINUED)

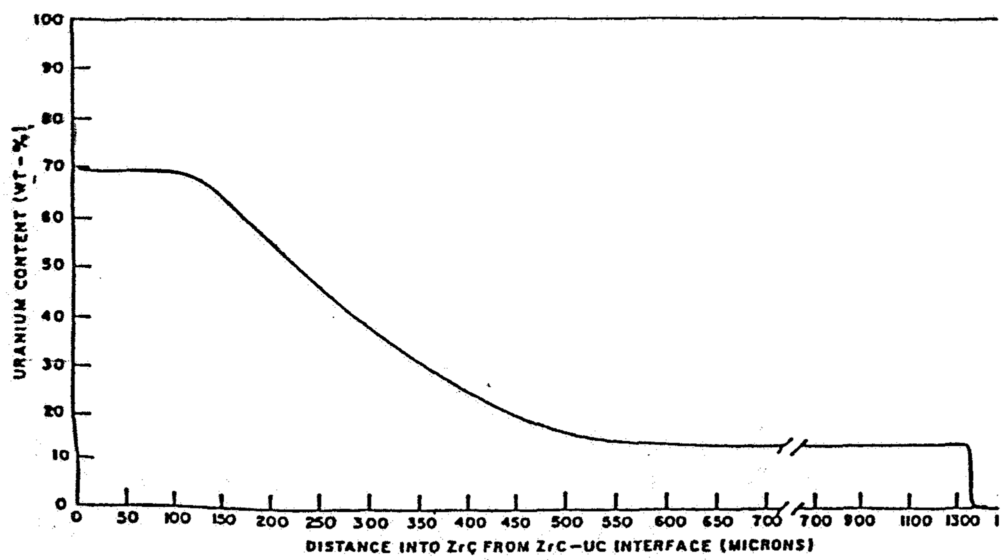


Figure 12

STRESS ANALYSIS OF MICROSPHERES

- TOTAL STRESS INDUCED IN MICROSPHERES = STRESS DUE TO BUILDUP OF FISSION GASSES, VOLATILES, AND SOLID FISSION PRODUCTS +THERMAL STRESS

- STRESS DUE TO FISSION PRODUCTS BUILDUP:
  - ACCOMODATION OF SOLID FISSION PRODUCTS REDUCES POROSITY OF THE FUEL MATRIX
  - FUEL CONSUMPTION BY FISSION INCREASES THE POROSITY OF THE FUEL MATRIX
  - THE NET POROSITY IN BOTH THE FUEL AND LOW DENSITY GRAPHITE COATING DETERMINE THE PRESSURE BUILDUP IN THE FUEL MICROSPHERES AS A FUNCTION OF BURNUP AND OPERATING TEMPERATURE
  - PRESSURE BUILDUP IS PROPORTIONAL TO THE STRESS INDUCED ON THE COATING

- THERMAL STRESSES ARE SMALL SINCE THE COATING PROCESS OF THE MICROSPHERES WILL BE PERFORMED AT ALMOST THE SAME TEMPERATURE AS THE FUEL OPERATING TEMPERATURE

Figure 13

PELLET BED REACTOR CONCEPT/NTP

CORE PARAMETERS

Rated Reactor Power (MWt)	1,000
Core Diameter (m)	0.8
Core Height (m)	1.3
Reactor Core Power Density (kW/cm3)	3.0
Diameter of Central Coolant Channel (m)	0.2
Coolant	Hydrogen
Maximum Fuel Temperature (k)	3,100
Maximum Coolant Exit Temperature (k)	3,000
Core Inlet Temperature (k)	120
Reflector Inlet Temperature (k)	70-80
Coolant Flow rate (kg/s)	32
Specific mass of reactor (excluding shield)* (kg/kWt)	1.0
Fuel Type [m·pt·(k)]	UC-TaC (2,800-3,670**±50) UC-NbC (2,800-3,570**±50)

* Compared to about 3.0 kg/kWt in Rover Program.

** M·pt in equilibrium with carbon; single phase UC-TaC has a higher melting point.

Figure 14

WHY PELLET BED REACTOR CONCEPT?

- Modular
- High Specific Power
- High Thrust and Specific Impulse
- Make full use of available technology base (NERVA-German AVR).
- Long lifetime; provide for possibility of in-orbit refueling.
- Could be coupled to any dynamic conversion system of choice (Stirling, K-Rankine, Direct Brayton Cycle) for NEP option.
- High heat capacity and passive decay heat removal.
- Could be used both for pulsed and continuous modes of operation.
- Redudancy in the reactor control mechanism.
- Low Development Cost

Figure 15

PELLET BED REACTOR CONCEPT/NTP

SAFETY FEATURES

- Subcritical water immersion and compaction.
- Two independent control systems, each capable of maintaining safe operation of the Reactor.

  --24 Control drums (only 20 are sufficient for reactor shutdown).
  --5 Safety Rod Drivers in the core.
- Reactor could be fueled in low earth orbit .
- Passive decay heat removal by radial conduction/radiation to space.
- Fuel pellet design provides for safe containment of fisson products, low thermal gradient (<100 K/cm), and structural integrity.
- High height/diameter ratio of the core provide for small cone angle, thus lower shield mass.
- Orbit Refueling.

Figure 16

PELLET BED REACTOR CONCEPT/NTP

KEY FIRST YEAR ACTIVITIES

ESTABLISH ENGINE SYSTEM MODEL

- Operation parameters (flow rate, temperature and pressure)
- Weight
- Critical Mass
- Thrust and Isp
- System redundancy and modularity

SHIELDING AND SAFETY ANALYSIS

- Optimization of biological shield mass
- Criticality Calculations
- Orbit refueling options
- Post-accident heat removal
- Fuel design

ASSESSMENT OF MATERIAL AND FUEL FABRICATION CAPABILITIES

- Fuel Fabricability
- Compatibility with structure materials and hydrogen
- Development of irradiation testing program for fuel burnup up to 10 at.% @ ≥3000 K.

INTEGRATION STUDIES

- Spacecraft-reactor system integration
- Single versus multiple engine option.
- Gas effluent interaction with vehicle structure.

Figure 18

PELLET BED REACTOR CONCEPT/NTP

COST ESTIMATE TO BRING TECHNOLOGY TO READINESS BY 2006 TECHNOLOGY FREEZE

PHASE	MINIMUM TIME (YRS)	COST ($M)
Conceptual Design and Technology Issue Resolution	3-5	100
Preliminary design and Component Development	5-7	800
System Ground Demonstration	1-2	1,000
Flight Qualification	1	1,200
TOTAL	10-16	2,100

Figure 17

PELLET BED REACTOR CONCEPT/NTP

KEY ISSUES

- Development of high temperature fuel and demonstrate its reliability at high burnup (up to 10 at .%) and high temperature >3000 K.
- High temperature materials compatible with hydrogen.
- High temperature instrumentation.
- Development and demonstration of MMW thrusters with lifetime up to 2 years at up to 3500 K.
- Environmental issues (e.g. CO_2 Mars atmosphere, creep strength at high temperature).
- Autonomous operation and fault detection technology.

Figure 19

PELLET BED REACTOR CONCEPT/NTP

PRESENT TECHNOLOGY LEVEL

TECHNOLOGY	READINESS LEVEL
Fuel	2
Reactor Design	4-5
Core & Structure	3
MW Thruster	1
Reactor Control	3-4
Instrumentation	2-3
Autonomy and Fault Detection	1-2
Shielding	5-6

Figure 20

N92-11102

PELLET BED REACTOR FOR NUCLEAR PROPELLED VEHICLES: II. MISSIONS AND VEHICLE INTEGRATION TRADES

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As Mohamed said, I will be discussing the mission and vehicle integration trades and so I am not going to say anything about reactors, neutronics or anything else. The issue here is that you can make a reactor or an engine, but unless you can hang it into a vehicle it won't go anywhere. So I would like to address some of these issues.

You have to go through all of these factors (Figure 1) before you know if the vehicle can fly. You have to look at the whole vehicle. You can have all kinds of efficiencies you want in the reactor, but if it doesn't fly, it won't go anywhere.

Here are some of the trades done back then during the NERVA program (Figure 2). What shape is your tank and where do you put your rocket engine and your reactor? You go in with some distance to avoid the radiation (this will cause feed system problems), then you begin to play with geometry; the optimum that came out is a 15 degree cone angle.

Figure 3 shows the mass and radiation breakdown for the shielding from the previous chart that I showed you. The 15-degree cone angle gives you the lowest radiation for a given shield mass. So, based on this chart it was decided that we would pick the 15-degree cone angle as the bottom of the tank.

There were many other trades that were done. Here is what the problem looked like; you are not going to Mars and get rid of the reactor, you are going to fire it, shut it down, and then you have to cool it. When you use propellant as coolant, you lose specific impulse. The trades done back then show what happens to your specific impulse as you cool the reactor down (Figure 4). So you have to go through these trades as well.

As to radiation maps (Figure 5), I am not a radiation expert, but these were done back for the NERVA engine. You have neutron flux, you have gamma radiation, a reference point up there and we are talking about a 1575 megawatt reactors operating for 53 minutes and so on. So all these factors have to be addressed.

Then as to what happens after shut down (Figure 6), you have a decay which goes as shown, and here is the radiation versus distance, which continues on, and so on.

In our present studies (Figures 7-9) we are moving from the 1960's to the 1980's and 1990's via computer programs. We had a very good correlation between the calculations from the old NERVA data that we got out of the design handbooks. The same thing was found for a small engine that was supposed to operate an ROTV out of the space shuttle, (if you can believe that) (Figure 7).

For a pellet bed reactor mission to Mars, just the other day one of our guys gave me these numbers (Figure 10). If you fly on May 11, 2018, taking 250 days for the total trip, with 30 days stay, these are your Delta-V breakdowns. So on the basis of this, we can take a thrust, an engine, and hang it on the vehicle and start calculating some system masses and see what happens.

This is what happens when you plot Delta velocity versus mass (Figure 11). The way we break things down is shown in Figure 12. We have a Delta velocity and a specific impulse of 1,000 seconds when we calculated with our program. We come up with a payload of 36 metric tons, the thrust is 315 kilo-Newtons. That's about 70,000 pounds or so, including the mass of the shield. This is the output. I must say this mass ratio is not payload fraction. Payload fraction is shown in Figure 13. This is for the top curve, the heaviest vehicle that we got and that's almost a half a million kilograms there. Pretty big stuff!

Looking at it parametrically in terms of payload fraction, we show that, as you demand more and more velocity out of a fixed performance, your vehicle becomes almost like the chemicals we have today, which have something like three to four percent payload fraction. This says that what you want to do is increase the specific impulse. And by the way, if you go to a single stage Delta V, which is like nine to ten kilometers per second with a nuclear vehicle, you begin to approach 25 percent of payload fraction.

I was talking to airplane people who design airplanes being flown for money and they say that of their takeoff weight, fuel is something like 40 percent. What we would like to do is drive the space vehicles in that direction.

We didn't do anything on cost for this workshop, but we did a lot of work on cost back in the 1970's. There is a whole bunch of reports that I sent NASA, and one written on February 1973 cost data, 1973 dollars. Oh, do they look good. I suggest that you take that to Congress when you go and talk to them.

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Bill Haloulakos
PeBR for Electric and Thermal Nuclear Propulsion

Acknowledgments

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DESIGN TRADE STUDIES

- Propellant Tank Geometries
  - Weight
  - Volumetric Efficiency
  - Radiation Considerations
- Skirts and Interfaces
- Handling, Transportation and Launching Factors
- Reusable vs Expendable
- Refueling, Refurbishing
- Start, Shutdown, Restart Factors
  - Fluid Transients
  - Heat Soak Back
  - Post Shutdown Cooling
    - — Performance Loss/Recovery

Figure 1

CONVENTIONAL TANK CONFIGURATION

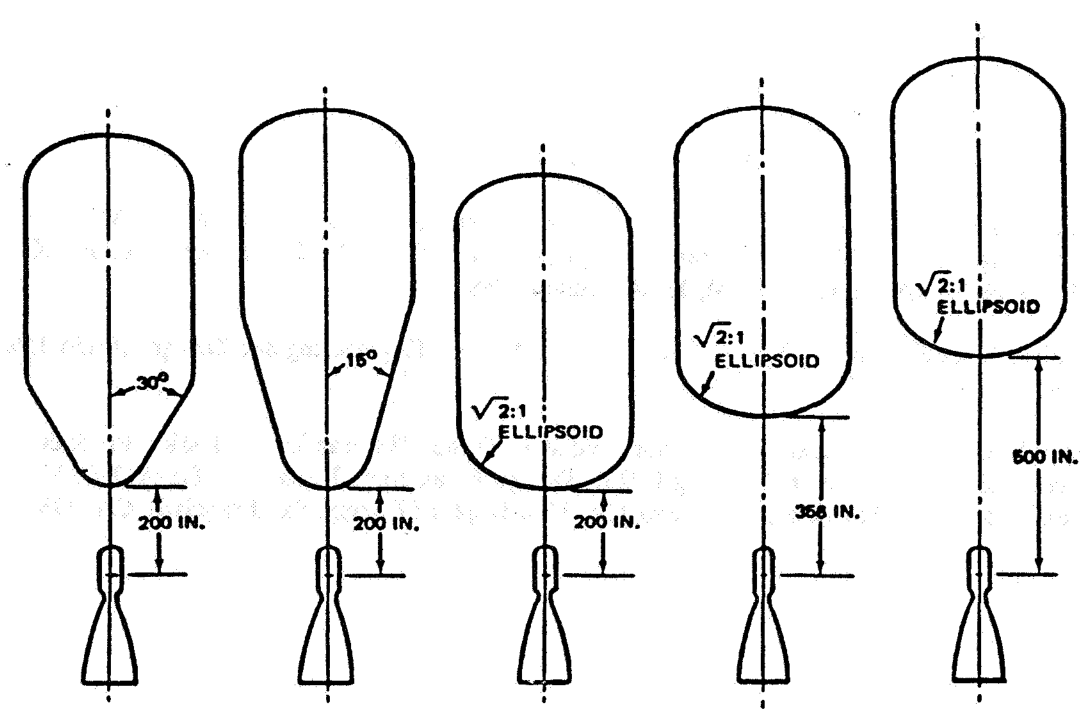
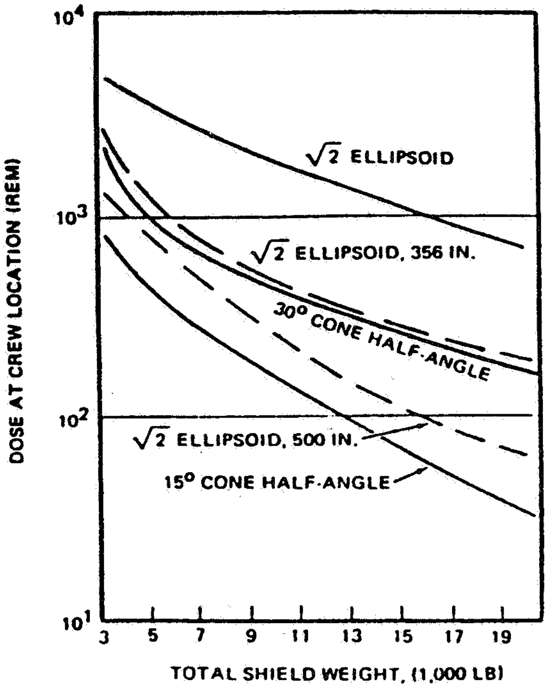


Figure 2

SHIELD WEIGHT REQUIREMENTS FOR CONVENTIONAL TANK CONFIGURATIONS



300,000 LH_2 CAPACITY
3,300–LB INTERNAL SHIELD
7,500–LB LH_2 RESIDUAL
3–ZONE DISK SHIELD
ALL SOURCES IN MAY 1969 CRAM RPT
200–IN. ENGINE TANK SEPARATION
(UNLESS OTHERWISE NOTED)

Figure 3

EFFECTIVE SPECIFIC IMPULSE

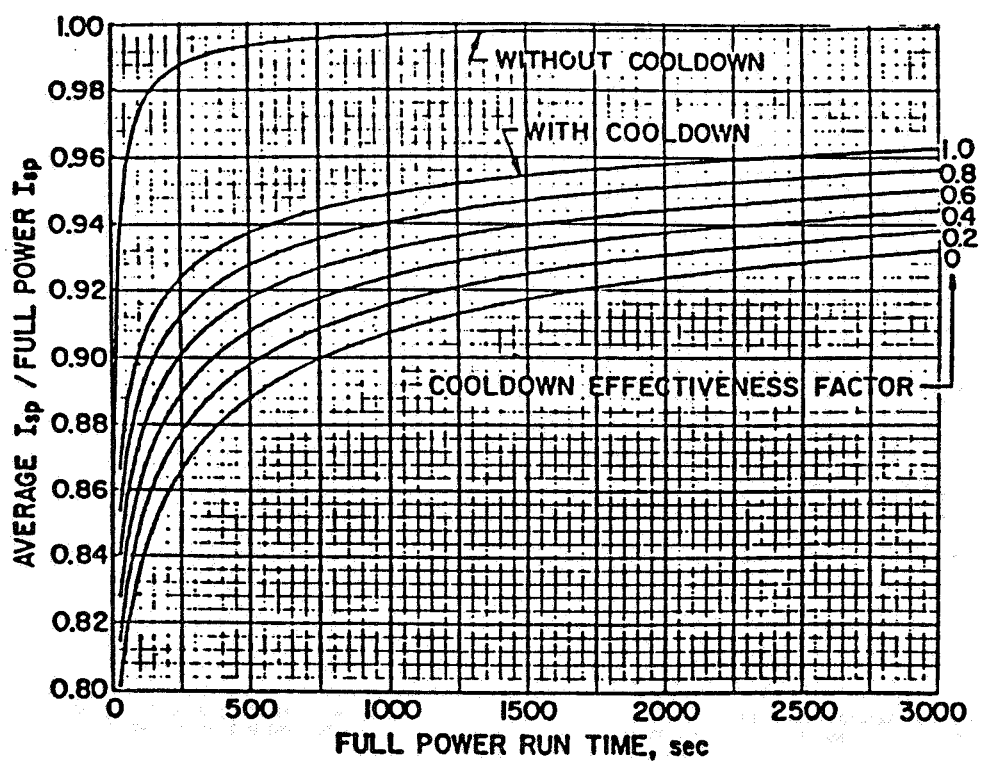


Figure 4

RADIATION MAP

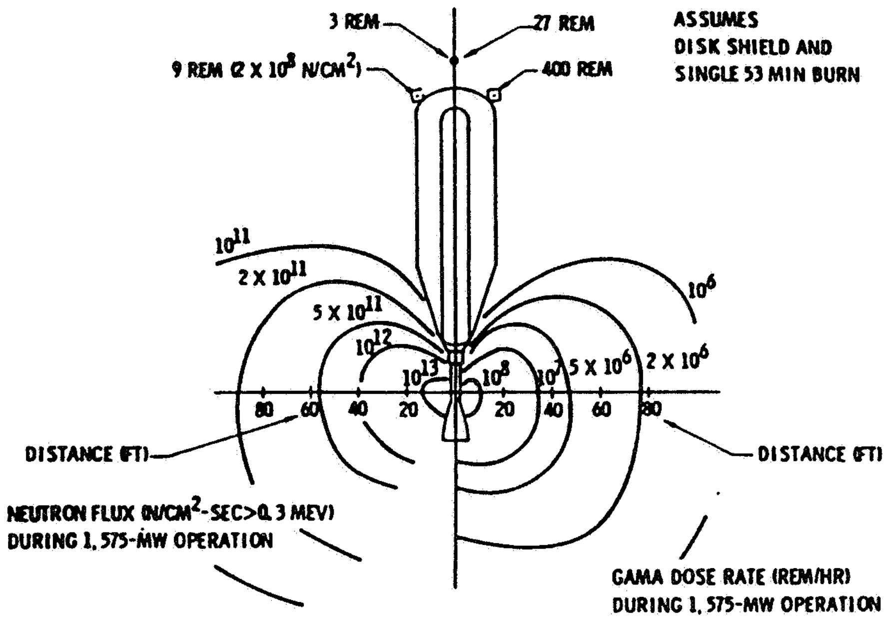
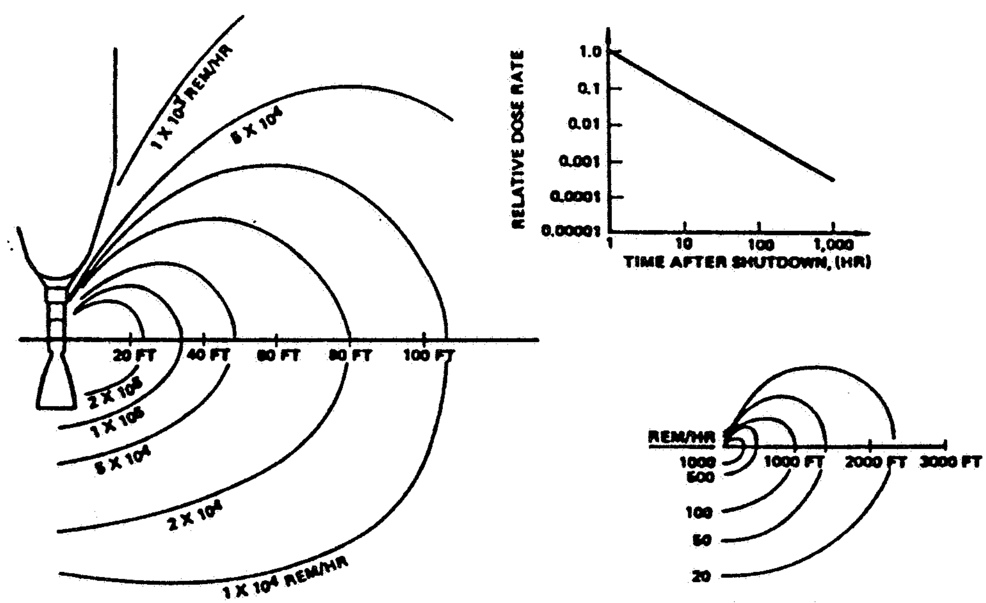


Figure 5

DOSE RATE AFTER SHUTDOWN



FISSION PRODUCT DOSE RATE
1 HR AFTER 0.5-HR OPERATION AT 1,575-MW

FISSION PRODUCT DOSE RATE 1 HR
AFTER 0.5-HR, 1,575-MW OPERATION

Figure 6

CLASS 1 SINGLE-MODULE HYBRID RNS

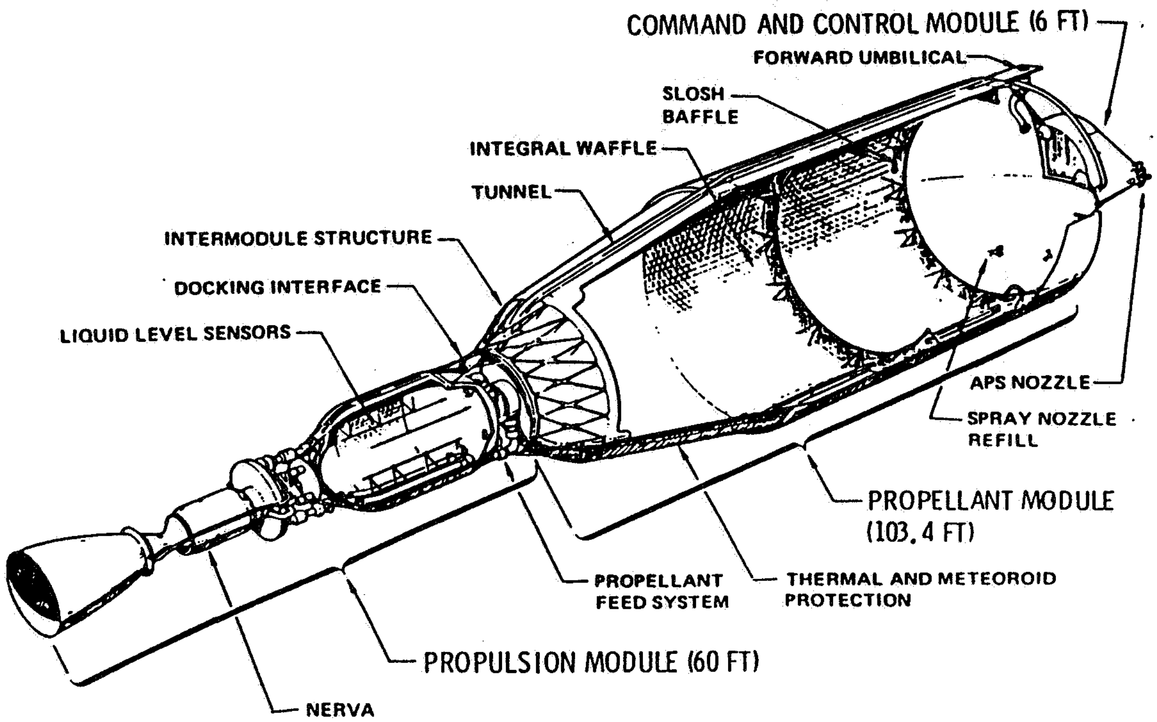


Figure 7

SPACE TRANSFER VEHICLE DESIGN DATA

LEO to GEO With Return Mission Data : Payload Mass 36,000 kg; Velocity Increment 9,000 m/s

	CRYOGENIC (O2/H2)		NUCLEAR FISSION	FUSION	
ROCKET ENGINE	6 RL - 10's	J - 2	LANL ALPHA 2	NEW	NE
Thrust (kN)	400.3	902	71.7	100	1
Specific Impulse (s)	450	429	860	3,500	1
Burn Time (s)	3,675	1,850	12,835	5,766	1
MASS BREAKDOWN					
PROPELLANTS (kg)	333,291	396,423	109,120	16,792	99,
Fuel (LH2)	51,275	60,988			
Oxidizer (LOX)	282,015	335,435	-	-	
PROPELLANT TANK(S)					
Total Volume (m3)	970	1,154	1,748	269	1,6
Mass (kg)	8,156	9,701	11,746	1,808	10,7
PRESSURIZATION (He system) (kg)	1,374	1,635	1,979	305	1,8
ENGINE (kg)	792	1,579	2,567	17,168	80,9
MISCELLANEOUS (kg)	3,411	4,058	4,913	756	4,4
TOTAL VEHICLE MASS (kg)	383,024	449,394	166,326	72,828	233

Figure 8

SPACE TRANSFER VEHICLE DESIGN DATA

LEO - GEO - LEO Mission ; Mpl = 36,000 kg ; Del V = 9 km/s ; Burn Time = 3675 s

ROCKET ENGINE		CRYOGENIC , 6 RL-10's	NUCLEAR , 4 ALPHA 2's	FUSION , Me=12 (Isp)
Thrust	(kN)	400	278	208
Specific Impulse	(s)	450	860	2500
MASS BREAKDOWN				
PROPELLANTS	(kg)	333,291	134,548	34,685
Fuel	(LH2)	51,275		
Oxidizer	(LOX)	282,015	-	
PROPELLANT TANK(S)				
Total Volume	(m^3)	970	1,937	499
Mass	(kg)	8,156	10,849	2,797
PRESSURIZATION				
He System	(kg)	1,374	1,828	471
ENGINE(S)	(kg)	792	10,270	30,000
MISCELLANEOUS	(kg)	3,411	4,538	1,170
TOTAL VEHICLE MASS		383,024	198,033	105,123

Figure 9

MARS MISSION ΔV SUMMARY

- TRANS-MARS INJECTION : 4.71 km/s
- MARS CAPTURE : 9.03 "
- TRANS-EARTH INJECTION : 9.93 "
- EARTH CAPTURE : 7.20 "

TOTAL : 30.87 km/s

- LAUNCH DATE : 11 MAY 2018
- TOTAL TRAVEL TIME: 250 DAYS
- MARS STAY TIME: 30 "

Figure 10

TOTAL OTV MASS vs. VELOCITY INCREMENT

PELLET BED REACTOR NTR

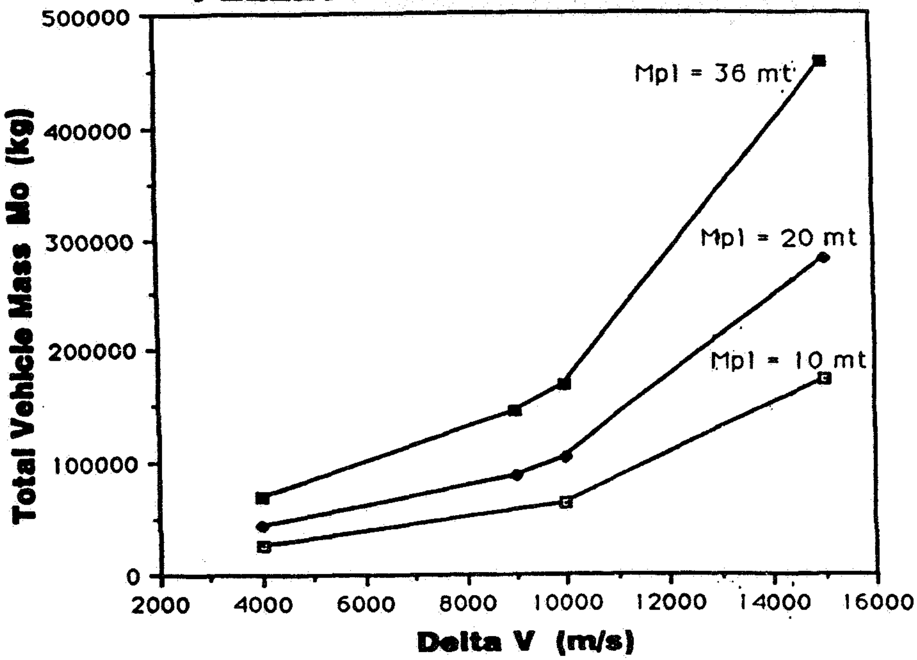


Figure 11

SAMPLE PELLET BED NUCLEAR OTV MASS BREAKDOWN

Input Parameters		
Delta V (ΔV)	15,000	m/s
Specific Impulse	1000	s
Payload Mass	36,000	kg
Thrust	315	kN
Engine Mass	1,875	kg
Shield Mass	4,000	kg
Calculated Parameters		
Mass Ratio R	4.611	
Propellant Fraction (Mp/Mo)	0.857	
Payload Fraction (Mpl/Mo)	0.086	
Tank Volume	5,249	m^3
Burn Time	170	min
Component Mass Breakdown		
Propellant (H_2)	364,568	kg
Propellant Tank	34,703	kg
Thrust Structure	649	kg
Pressurization System (He)	4,365	kg
Meteoroid/Thermal	9,164	kg
Total Vehicle Mass	455,324	kg

Figure 12

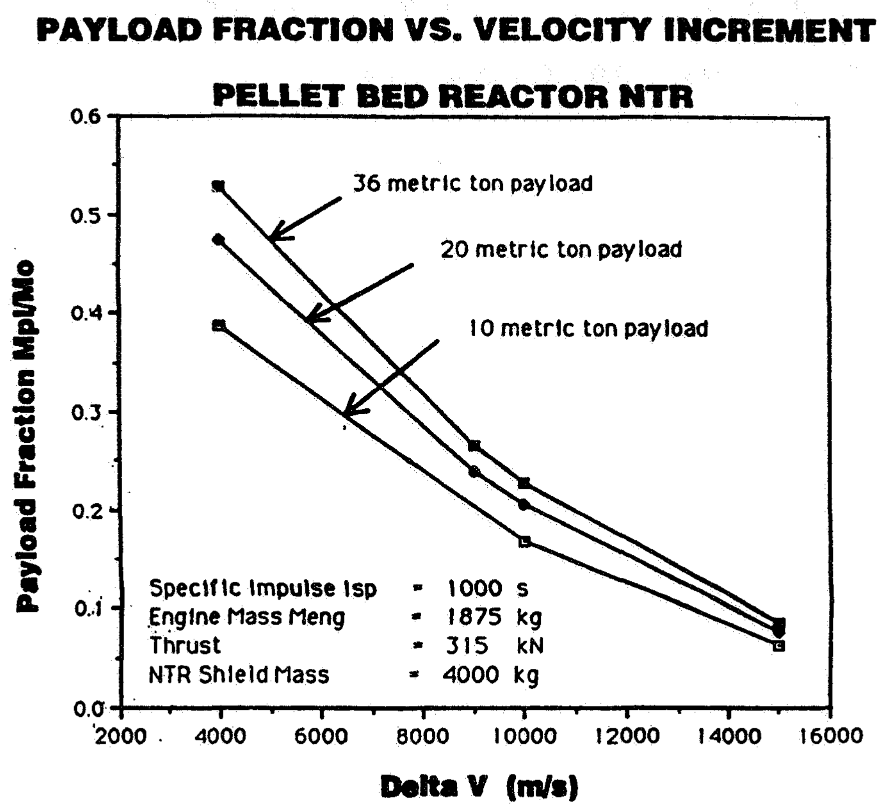

PAYLOAD FRACTION VS. VELOCITY INCREMENT
PELLET BED REACTOR NTR
36 metric ton payload
20 metric ton payload
10 metric ton payload
Specific Impulse Isp = 1000 s
Engine Mass Meng = 1875 kg
Thrust = 315 kN
NTR Shield Mass = 4000 kg
Payload Fraction Mpl/Mo
0.0
0.1
0.2
0.3
0.4
0.5
0.6
2000
4000
6000
8000
10000
12000
14000
16000
Delta V (m/s)

Figure 13

N92-11103

FISSION FRAGMENT ASSISTED REACTOR CONCEPT FOR SPACE PROPULSION-- FOIL REACTOR

Steven A. Wright
Sandia National Labs

Well, I am not the salesman that Mr. Zubrin is, nor the poet that Mr. Kirk is, but I think we have a reactor concept that will be intellectually stimulating and fun. It is called the foil reactor in the agenda, but I will be referring to it as a fission fragment assisted reactor concept for space propulsion. And as Mr. Kirk said, the idea is not new, it is just a collection or combination of ideas that have been around for quite sometime.

What we want to do (Figure 1) is to fabricate a reactor using thin films or foils of uranium, uranium oxide and coat them on to substrates. We would make these coatings so thin as to allow the escaping fission fragments to directly heat a hydrogen propellant. This idea is not new. In 1958, Bussard and Delauer mentioned a concept of similar nature in their book; however, they didn't investigate it very much in depth.

At Sandia we have been studying this idea of direct gas heating and direct gas pumping in a nuclear pumped laser program. In this program we are actually using fission fragments to pump lasers. And to show you that I am stealing ideas, I actually have one of their vugraphs that fits very nicely in this talk (Figure 2).

In this concept two substrates are placed opposite each other. The internal faces are coated with thin foil of uranium oxide. The foils are so thin that a large fraction of the fission fragments escape into the gas. The gas is chosen so it will be excited by escaping fission and emit light to provide light amplification. This method of pumping a laser does indeed work.

We have taken another idea for our concept from the particle bed reactor. In the particle bed reactor porous frits are used to control the flow to the fuel element. For the foil reactor, we will also use substrates that are porous. However, our substrates will be coated with thin films of uranium oxide. The gas flows to the substrate into this folded flow reactor, and it comes down and flows through, and heats up through this substrate, which will pick up approximately 2,000 or 2,300 degrees Kelvin. Then, in the exit plenum between the foils, a large fraction of energy is being directly deposited, and will heat the gas another thousand degrees. So our gas temperatures are much, much hotter than our substrate temperatures and we would do the same thing on the other sides. The one thing we have to optimize is the spacing between the plates so you don't get a lot of heat transfer back to the substrates.

We selected a hydrogen propellant pressure of 1000 psia. To stop the fission fragments that travel through the plenum between the foils, you need approximately two centimeters of hydrogen at this pressure. However, we are proposing a system which uses five centimeters. This spacing was selected to minimize the heat conduction or heat transfer back to the substrates. There exists a large technology base (Figure 3) that supports this concept of direct gas heating, and most of it comes from the nuclear pumped laser program called FALCON, which stands for Fission Activated Laser CONcepts. These experiments are being performed at Sandia, and in conjunction with experiments at INEL.

We already have experimental verification for the amount of energy and the number of fission fragments that escape foils, as a function of foil thickness. I will show you the vugraph supporting that in a minute. Since we are doing experiments, we have to develop technology to coat UO_2 on a variety of substrates (Figure 4), including stainless steel, aluminum, alumina, and beryllia. The technology to make coatings is available, but we do need to advance the technology, especially to place them on porous substrates.

Figure 4 shows a scanning electron micrograph of a uranium oxide coating placed on an alumina substrate. We have made these types of coatings on both aluminum oxide and beryllium oxide ceramics.

In our experiments (which are transient experiments), we have verified that one can heat gases at least 1,000-1,500 degrees above the substrate temperature. In these experiments the power densities are approximately 17 kilowatts per square centimeter of foil surface area. This is 17 times higher than the power densities that we are proposing for the nuclear propulsion concept described here.

Let me show you that we really do know how much energy is getting out of these foils as a function of foil thickness (Figure 5). This figure shows the energy escape fraction as a function of foil thickness. The diamond marks are actual measurements. With a three micron foil, you can get about 20-21 percent energy release fraction. We are proposing, in this concept, to work between the one and two micron foil thickness; thus we would expect to see fission fragment escape fractions (in terms of energy) of, say 24 to 30 percent. The squares on the figure show you the actual particle escape fraction, and that's important because it tells the number of the fission fragments that are lost out the exhaust of the reactor.

If you make a reactor out of a coated porous substrate and assemble these fuel elements to make a nuclear driven rocket engine out of this type reactor geometry, what does it get you (Figure 6)? We feel like this gives us enabling technology that is well beyond what is feasible with current designs. The major advantage of this approach is that the propellant gas is much hotter than the structure, approximately a thousand degrees hotter. As a consequence we also get very respectable Isps; 800 to 1,000 seconds for very low substrate temperatures. Here is an example. A 2,000 degrees Kelvin substrate temperature allows one to obtain a gas temperature of 2,700 degrees and an Isp of 836 seconds. I believe we did this calculation for one and a quarter micron foil thickness.

This reactor is very big, it's very dilute, so it can be run at very high power levels to obtain tremendous thrust; 600 thousand pounds or more. It's a lot of thrust.

How would you make a reactor out of this? What we proposed is to place the foil-coated substrates into an annular geometry as shown in Figure 7. The gas flows down in the narrow gap between these plates. There is a three millimeter gap between the plates. Cold, dense hydrogen gas flows down, turns the corner in both directions and flows through the beryllia substrates, which we assume to be porous and have a one to two micron coating of uranium oxide. The gas flowing though the substrates heats up 2,000 degrees. Once the gas reaches the exhaust channels the escaping fission fragments heat the hydrogen up another thousand degrees.

Figure 8 shows a cross section of one fuel element module. Each module a is approximately 36 centimeters in diameter and 4 m long. The module is a self-contained pressure vessel that uses carbon-carbon for the containment boundary (Figure 9). One would assemble these modules in a hexagonal or a square lattice to form a reactor. Each module uses the beryllium oxide as a neutron moderator and as the porous substrate upon which the uranium oxide is coated. At the exit end of the module the pressure vessel is shaped into a nozzle which could, if needed, be transpirtation cooled. The weights (engine masses) that I will show include the fuel and all the structure, including the nozzle at the bottom.

About a hundred of these modules are required for the reactor to have sufficient criticality. It is a big system (Figure 10); about four meters tall and four meters in diameter. Figure 10 shows fewer modules than a hundred, but this is just a schematic to illustrate the concept.

Because the fuel is so dilute, a substantial reflector is required (Figure 11). The reflector should be somewhere between 75 centimeters and a meter thick. A wide choice of reflector materials can be used. You can use heavy water, but that is heavy. You can use beryllium, which works quite well, but also it is about as heavy as heavy water. A nearly ideal material to use would be liquid deuterium, but we feel the power required to keep the deuterium liquid would be too high. So we are proposing a new material; deuterated methane. With fairly low pressures and pumping powers you can compress it and keep it liquid. For a fuel module that uses a two micron foil thickness, you need only three-quarters of a meter of deuterated methane to reflect enough neutrons back into the reactor to have sufficient criticality margins.

The next two figures show schematics of the reactor (Figures 11 & 12). In our design the reflector covers the circumference, and the top of the reactor. No reflector is used on the bottom or exit end of the engine. Since the reflector is so thick, an external shield is not required. This 0.75 m reflector can reduce the gamma radiation dose rates by about four orders of magnitude. Consequently, all the weights that I will show you include our reflector/shield.

Let me summarize the key features of this concept (Figure 13). I have already talked a little bit about the size; a hundred modules, four meters in diameter by four meters tall. We are assuming a two micron foil thickness, which gives us an efficiency of 24 percent for the energy going directly into the gas. We need 30 kilograms of uranium oxide fuel to go critical. If you sum up all the weights, including some seven tons put in for pumps and control, you end up with 42 tons. This is big, but you also have a lot of thrust.

The power densities are low; about 300 watts per cubic centimeters. This is equivalent to a surface flux of a thousand watts per square centimeter. For reference purposes this power density is a fourth of what NERVA had. Total power is 13 gigawatts. Two percent of this power is deposited in the reflectors. This presents a problem. We have to cool that reflector, and so we are going to take some penalty for providing a cooling system. I will talk a little bit more about that in a minute.

In spite of the large reflector, the thrust-to-weight ratio is still quite respectable. It is six and a half, even for a huge reactor.

Continuing to examine Figure 13 and the key features, one sees we are limiting the maximum surface temperature to 2,700 degrees Kelvin. This is a good hundred degrees below the melt temperature of beryllia, and 400 degrees below the melt temperature of uranium oxide. Our gas temperatures are 3,400 degrees Kelvin and this and gives us an Isp of 940 seconds. For the design we proposed, we do not have a large expansion ratio nozzle. This is because we are limiting the diameter of the nozzle to the diameter of the module. One can conceive of grouping modules to increase the expansion ratio to a 100 to 1 or 200 to 1.

We have done some scoping calculations to estimate the dose rates (Figure 14). Because we have so much hydrogen propellant between the reactor and the crew habitat, which is placed at a hundred meters away from the reactor, we don't expect significant dose rates until the last burn, when the last 30 meters of hydrogen above the reactor are expended. Even though the average dose rate is high, we have so much thrust that our burn times are short. Because of the tremendous thrust, the burn time is only 11 minutes for the Mars to Earth Burn, and a short 3 minutes for the Mars to Earth burn. The cumulative dose is 4.5 Rads.

We thought a little bit about what some of the safety features of this reactor concept or rocket concept are. Figures 15 lists both advantages and disadvantages. The major advantage is that the structure is much cooler than the propellant; about a thousand degrees cooler. Additionally, the hot surfaces are limited to very, very small surfaces on the substrates. Only the outer 20 microns are hot. The rest of the materials are cool because they are bathed in cold hydrogen.

Another advantage is that the fissile inventory is low, 18 to 30 kilograms. We have redundancy, because of the large number of self-contained pressure vessels in each module. We have very short burn times, three to ten minutes for each one of the burns; as a consequence, we have total burn times of 22 minutes. So we are running at low temperatures and not running very long.

I don't know if you want to include this as an advantage or disadvantage, but it is such a large dilute reactor that it would more than likely break up on re-entry or impact. In case of impact, criticality is not a problem, if it's an impact into water. It is difficult to make this reactor go critical, so immersion in water has a negative K-effective affect. Just about anything you do to this reactor is going to make it go subcritical.

The hydrogen worth itself is negative. The hydrogen has a negative $4 worth for the whole reactor core. Over a single module it is about 4 cents, so loss of hydrogen from a single module results in 4 cents positive reactivity. This will result in a rapid power transient. You could easily deal with the resulting power increases. We also think that you could provide enough fuel modules in the reactor design so that if you lost all the hydrogen and the fuel from the fuel modules you could still go critical.

An additional safety feature is the low power densities. If power to flow mismatches did occur, the heat-up rates would be relatively slow. And in addition, since it is difficult to find sources of large positive reactivities, large energetic accidents should not occur. Thus the core design naturally provides slow accident progressions.

I think you can summarize all of these advantages into three major titles:

(1) We have increased reliability because of the lower temperatures and modularity.

(2) It is tolerant to power-to-flow mismatches. A significant power-to-flow mismatch, would vaporize the uranium oxide surface and blow that out the back end; however, you could still be critical; and

(3) The design inherently leads to graceful failure modes. You shouldn't be able to destroy the reactor through energetic reactor reactivity-induced accidents.

The major disadvantage is a perceived disadvantage. We are throwing a lot of fission fragments out the back end of the reactor in the exhaust plume. Another disadvantage is that the reactor design has a low structural mass and is quite large. It may be difficult to withstand the required loads.

A significant effort is required to learn how one might design a reactor or rocket of this concept. An additional penalty or disadvantage is that a significant amount of equipment is required to cool the reflectors.

In some aspects, losing fission fragments out the exhaust has a positive effect. About half of our fission fragments are gone. That's why I pointed out the particle escape fraction earlier. As far as the crew is concerned, having lower fission product inventory is a benefit.

What are some of the key technology issues (Figure 17)? You have to remember we have taken this idea from the nuclear laser program, and there we are trying to get all the energy we put in to the gas back out as light. If we get light out of this excited hydrogen, it is going to heat up our substrates and the concept isn't going to work; so we need to make sure that we test the concept of directly heating hydrogen with fission fragments. We have to try hydrogen in the SNL laser experiments to find out if we get significant quantities of light out. We think the answer is no, because hydrogen is a symmetric molecule. If you want to make a laser, you use $C0$ or $C0_2$, which is an asymmetric molecule. Additionally, our experience indicates that most of the excitation energy will end up as thermal energy if we have high gas pressures and high temperatures, which we do.

We think the physics is in our favor here, but we don't know. We have to test it. Also we need to study dilute system critically. Nobody has spent much time on this or reported on it, although we scoped it out a bit. We also need to study reactor structural designs for large dilute systems. Again, this hasn't been done. And finally, we need to learn how to fabricate porous frits and ceramics. They could be made from the beryllioxide as I mentioned, but there is no reason why we couldn't use carbon porous frits with zirconium carbide or uranium carbide overcoatings. These materials would increase our temperature capabilities.

We have investigated techniques to coat solid substrates, but we haven't coated porous materials. Once you can do these things, we need to study its integrity. How much of the hydrogen erosion would occur on the fuel and substrate? What kind of maximum thermal gradient can be tolerated before we start popping off or flaking off fuel. And we need to take a really good look at the reflector cooling, at how much it weighs and how one would go about cooling the reflector. We don't think you can push cool hydrogen down into the liquid or the deuterated methane to cool it, because hydrogen is poison to this reactor. So you have to pump methane out of the reactor to some sort of heat exchanger up above the reflector.

The critical tests to verify such a proposed concept are closely related to the key issues (Figure 18). We need physics experiments. This should require a couple years of work, which have to be performed in-pile, so it's fairly expensive, $5 million. We need scoping studies for dilute system criticality, reflector cooling, and structural design. Again, I estimate it will take a team of people about two years and $5 million.

Additionally, we need technology development. We need to learn how to build porous substrates either out of beryllioxides or carbides. We need to learn how to make coatings, again, with oxides or carbides. And we need to study and test the integrity of these uranium and zirconium carbide coatings.

We need component testing. Ideally these should be channel-type tests, i.e. tests where you would have one of these coated substrates assembled to mock-up a fuel module. You would like to test them at prototypic power, temperature and flow rates. Unfortunately there aren't any reactors around that can meet the desired flux levels that you need. Two candidates would be HFIR reactor and Advanced Test Reactor. I am not sure of the accuracy of these numbers, but it is in this range. I believe we can only get about 50 to 100 watts per square centimeter power density on the surface of such a reactor. There is another reactor being proposed for the nuclear pumped laser program and this reactor might be available in 1995. If this reactor is built, you might be able to get up about 400 watts per square centimeter. If this test reactor is built, you might be able to get up about 400 watts per square centimeters. If this test reactor existed, one would need about $20 million in two years worth of module testing experimentation.

Then finally you need systems integration, site preparation, engineering fabrication, and facility operation. My total numbers here are in the same range as everybody else's, 1.2 to $2.4 billion. The cost depends on whether you want to go first class, or do it a little cheaper, or on how many people are involved.

How would you ground test such a thing (Figure 19)? Shooting fission fragments out the back end would not be acceptable. What we are proposing is that one could overcoat the UO_2 films with sufficient amounts of zirconium carbide or another material to stop the fission fragments so they don't get out, and to do this to all the modules except one. Then for the coated modules we would propose a closed 13 gigawatt loop heat exchanger. It's no small item, but probably is within reason, because you have 33 gigawatt nuclear power plants. Then, in that one module, you could run it as an open loop at about 130 megawatts. You would have to vent the exhaust through a scrubber. So this one scheme could be used for testing.

Now, I am a nuclear engineer, not a rocket scientist, and I feel rather uncomfortable putting up Figures 20-23. We have tried to make an estimate of what the IMLEO would be as a function of thrust-to-weight, and I believe we are roughly in the category shown. We are expecting Isps of about 900 or 950 seconds, so we are predicting an IMLEO of about 450 metric tons including shields. We think this compares favorably with the NERVA baseline.

What are the mission options (Figure 23)? I think we have a variety of them. Because we have such high thrust you can carry more propellant, and you can make much shorter trip times if you can get the propellant up there. You can take more cargo as another option, but again, you have to take more propellant. You could also carry extra modules or extra equipment to add redundancy.

We think this concept might be ideal for a freighter because if has so much thrust. In fact, it has so much thrust it might be a problem to humans on board. Coming back from Mars, you have several G's of acceleration. You might be able to use it for earth-moon freighting, perhaps distant planetary exploration or cargo ships to Mars.

As to the burnup, we think this thing might even be reusable, because it has such low temperatures and it would be limited only by burnup.

Let me conclude. I've listed a few of the advantages (Figure 24) of this technology. In general, however, we feel that if you look at all solid-core nuclear thermal rockets or nuclear thermal propulsion methods you are going to find they all look pretty much the same. They look good compared to the chemical approach, but within themselves they vary 10, 20, 30 percent; small percentages. So we think you are going to have to make your decision based on something else. We feel that something else could be, and should be, safety or reliability. We feel that this reactor has higher potential reliability. It has low structural operating temperatures, very short burn times, we think there are graceful failure modes, and it has reduced potential for energetic accidents. If you do have a failure on the ground or anywhere else, you are not likely to kill people or damage equipment through energetic accidents or energetic explosions, and we could increase the redundancy through modularity.

In conclusion, going to a design like this would take the NTP community part way to some of the very advanced engines designs, such as the gas core reactor, but with reduced risk because of much lower temperatures.

Fission Fragment Direct Heating Concept

Fabricate a Reactor from thin Foils of Uranium coated on substrates to allow escaping fission fragments to directly heat H_2 propellant

Bussard and Delauer (1958)
Nuclear Pumped Laser Tests (FALCON)

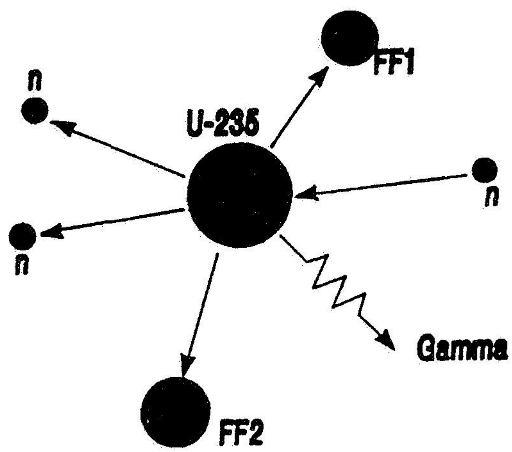
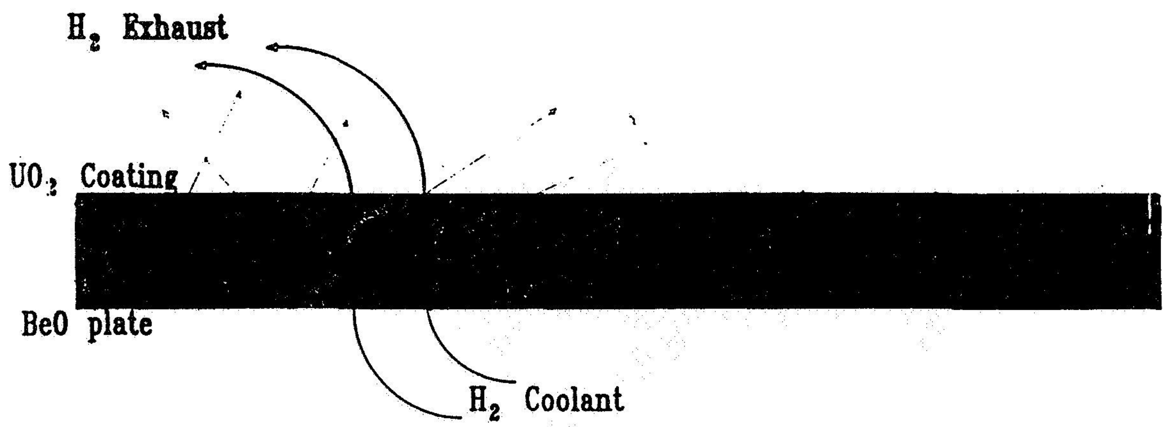


Figure 1

FISSION FRAGMENT DIRECT GAS HEATING



SECTIONED FOIL SHOWING COOLANT FLOW AND FISSION FRAGMENT HEATING OF EXHAUST

Figure 2

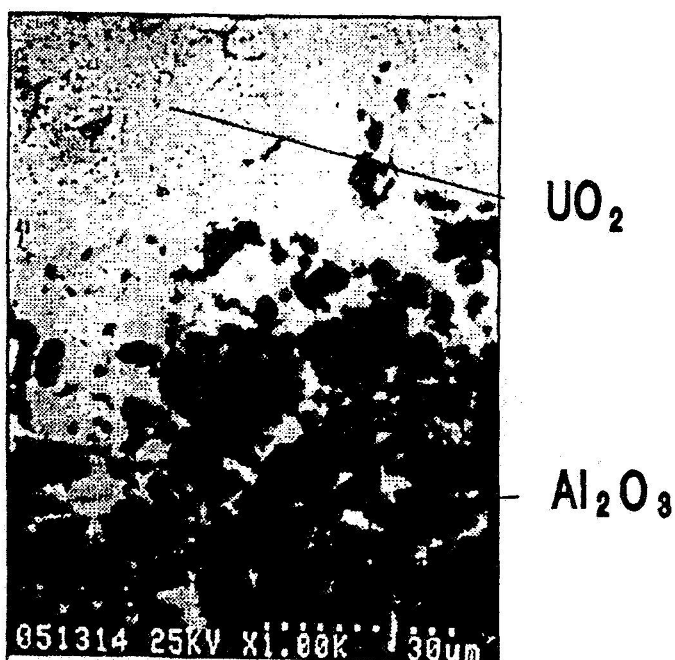
Existing Technology Base

Nuclear Pumped Laser Experimental Program
(FALCON at SNL & INEL)

- Experimental verification of fission fragment energy escape fraction versus UO_2 foil thickness
- Coating technology of UO_2 films on metalic and ceramic substrates exists, and is being advanced
- Experimental verification of direct gas heating well above substrate temperatures (> 1500 K)

Figure 3

Coating Technology



UO_2 Film on Al_2O_3

Figure 4

Comparison of Measured and Calculated FF Energy Release Fraction Versus UO_2 Foil Thickness

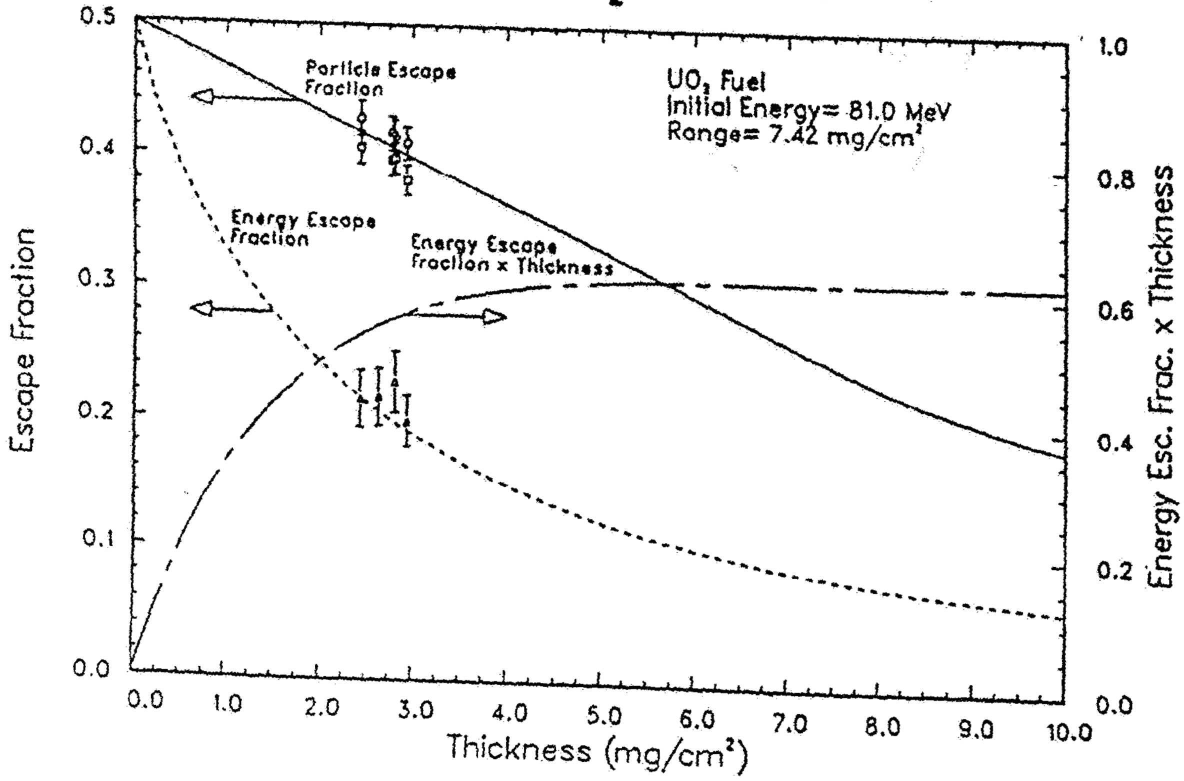


Figure 5

Advantages of Direct Gas Heating "Enabling Technology"

- Cool Structure ,Hot Gas $T_{gas} - T_{substrate} = \approx 1000$ K
- Operating Conditions provide good ISP and High Thrust

$T_{substrate}$	T_{gas}	ISP	Thrust
2000 K	2700 K	836 sec	686,000 lbf
2300 K	3100 K	898	633,000
2500 K	3370 K	937	604,000
2700 K	3630 K	975	578,000
3000 K	4040 K	1030	545,000

Figure 6

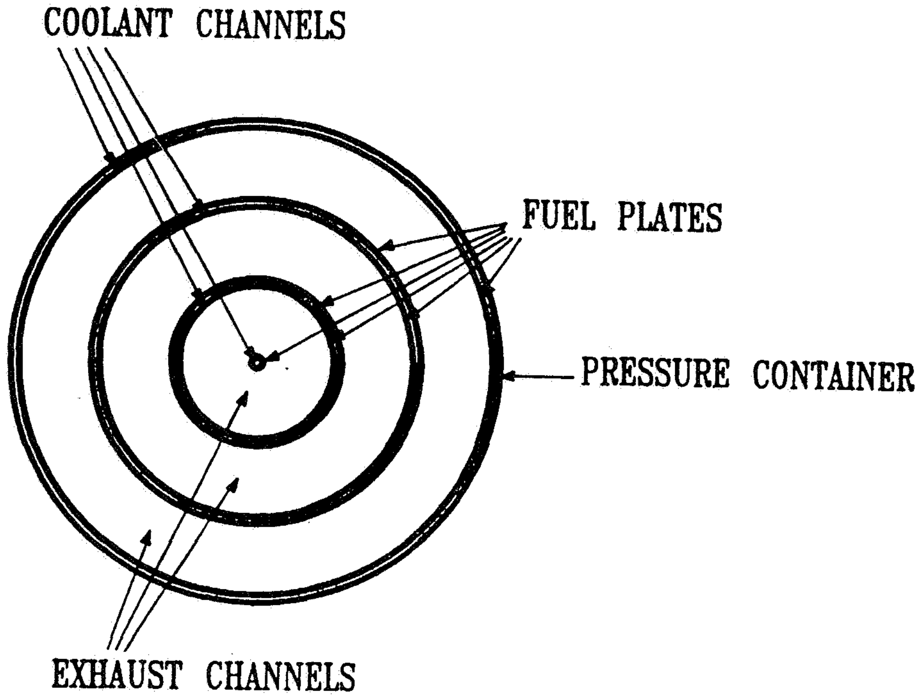
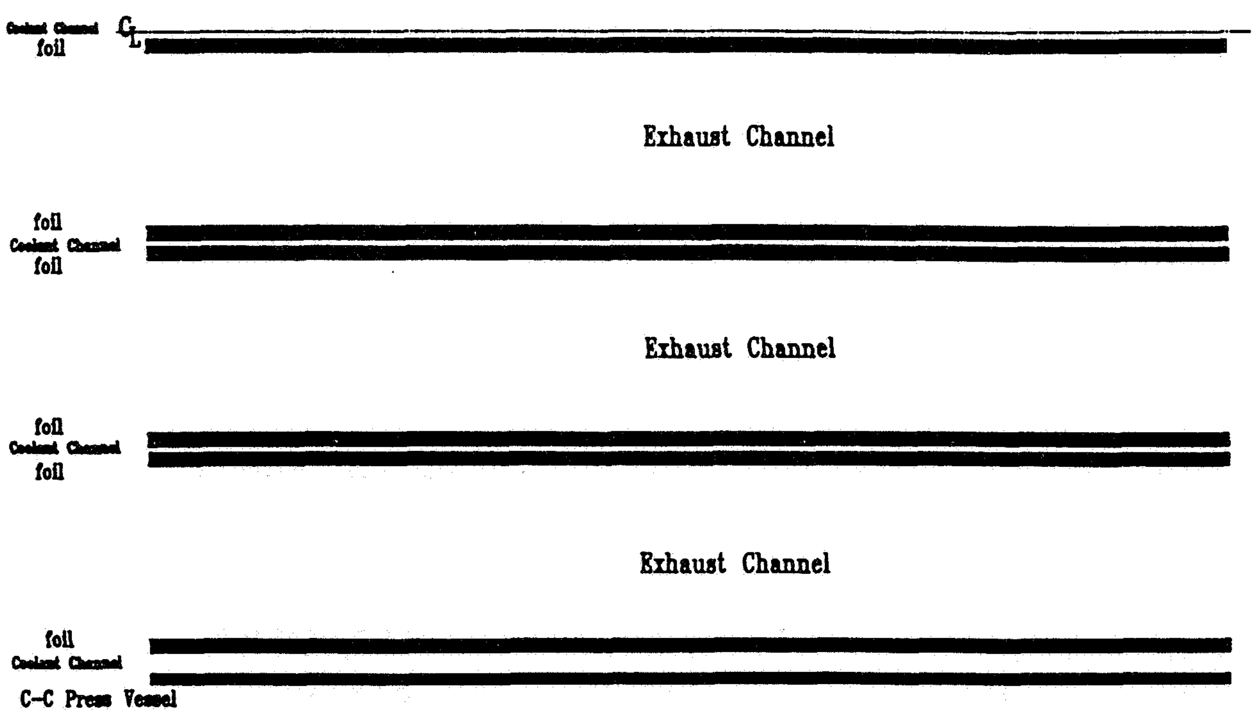
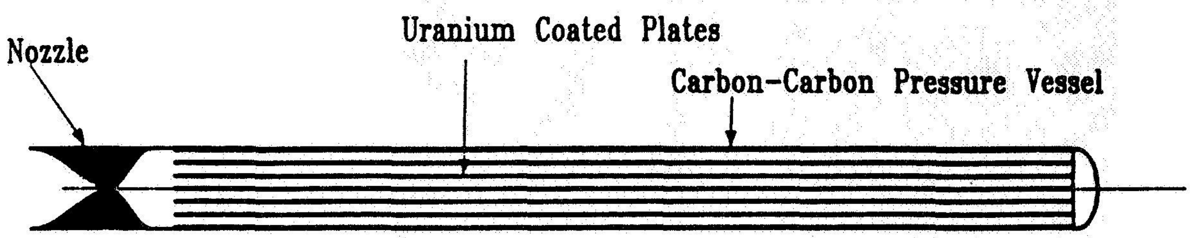


Figure 7



CROSS SECTION OF FOIL REACTOR MODUAL SHOWING
PRESSURE VESSEL, FOIL, AND CHANNEL ARRANGEMENT

Figure 8



CROSS SECTION OF REACTOR MODULE SHOWING FUEL, PRESSURE VESSEL, AND NOZZLE ARRANGEMENT

Figure 9

Schematic of Direct Heating NTR

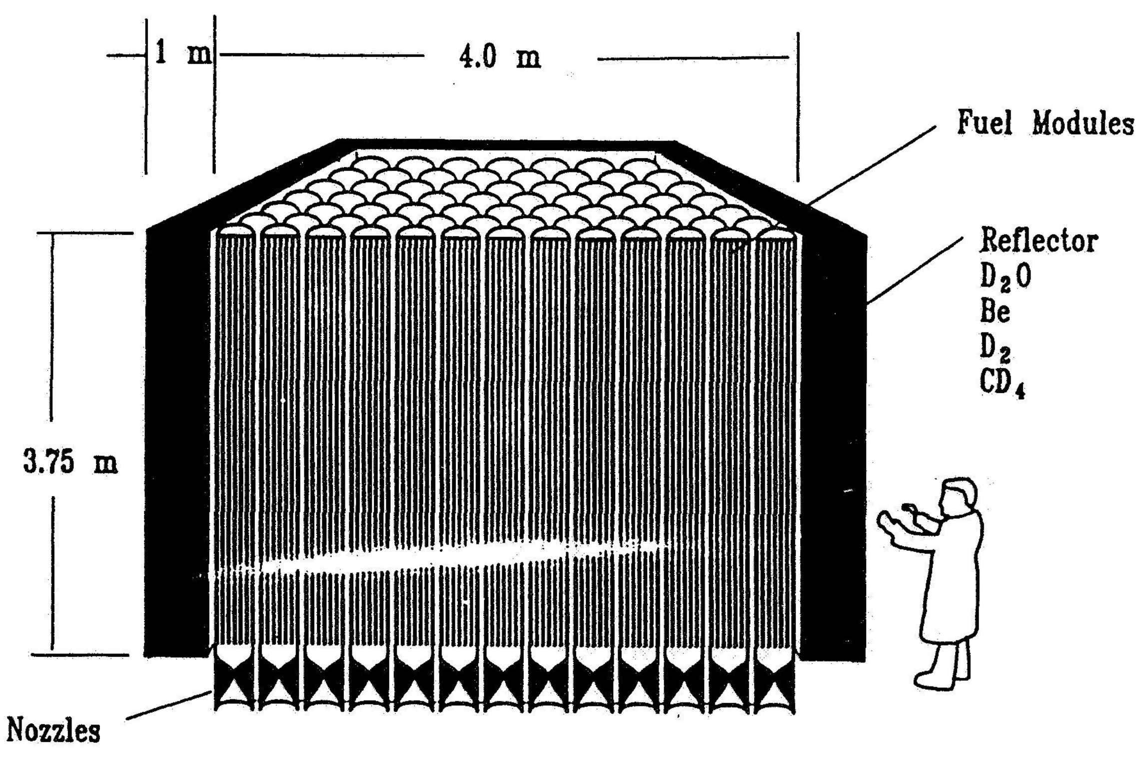


Figure 10

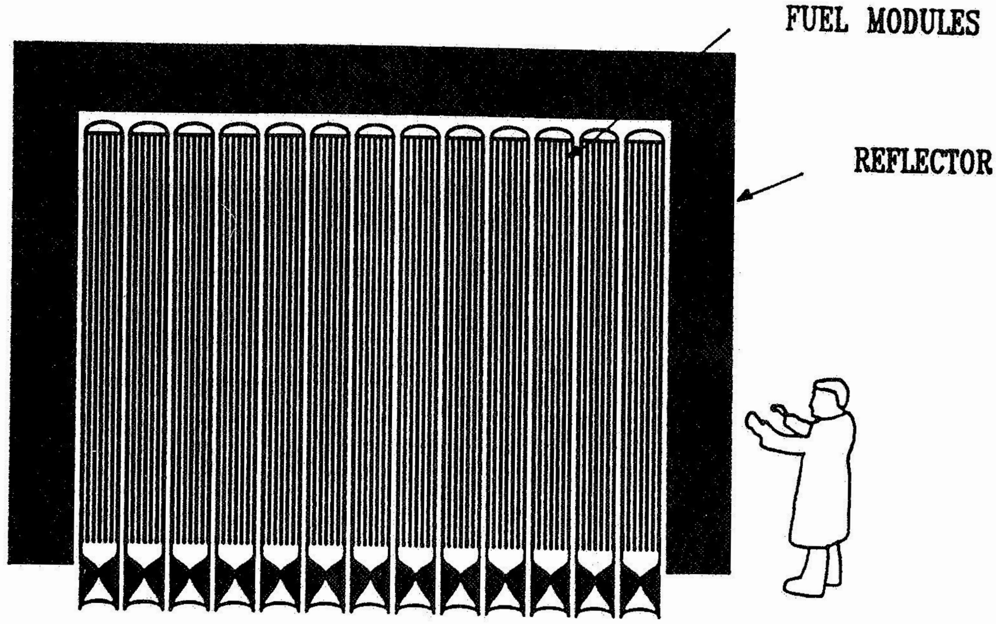


Figure 11

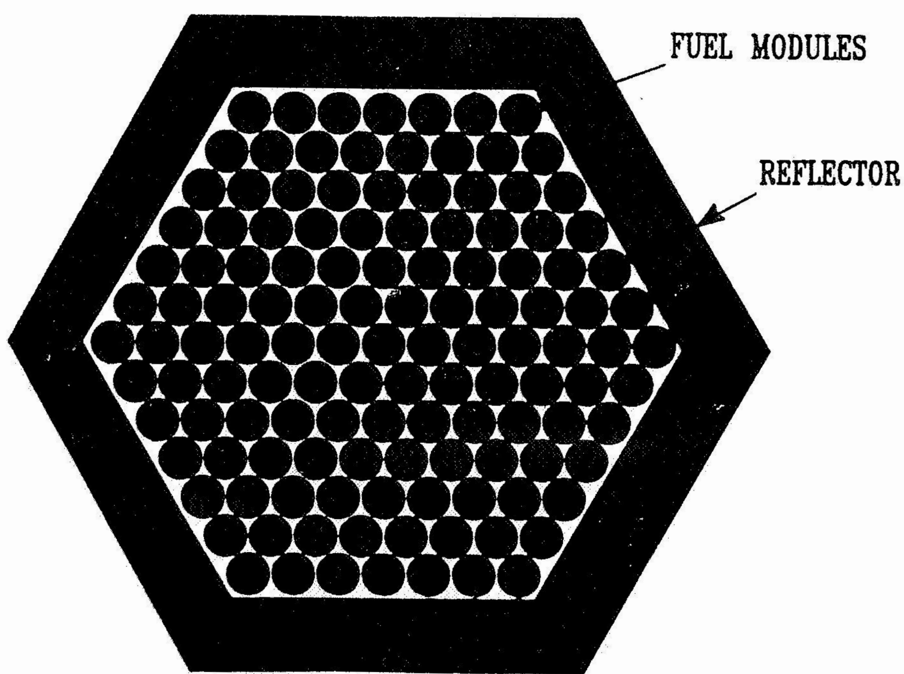


Figure 12

Key Features of Direct Heated NTR

Diameter	3.75 m	Power Density	310 W/cc
Height	4.0 m		1000 W/cm^2
Reflector Thickness	.7 m		
No. Fuel Modules	100	Power	13.3 GW
Module Dia.	.358 m		
UO_2 Mass	18-30 Kg	Reflector Power	2%
Mass			
Moderator (D_2O)	5 T	Thrust	600,000 lbf
Nozzle Refl (Be)	3 T		
Tube Wall (C-C)	2 T	Thrust/Weight	6.5
Substrate (BeO)	7 T		
Reflector (CD_4)	18 T		
Pumps & Control	7 T		
Shield (Not needed)			
Total	42 T		
$T_{foil-max}$	2700 K	(4% Heat Transfer losses)	
T_{gas}	3400 K	(Dissociation not included)	
ISP	990 sec	Gas Exit Velocity	70 m/s
Nozzle Expansion Ratio	43.1	Gas Pressure	1000 psia
Foil Thickness	2 μm	Foil Efficiency	24%

Figure 13

Radiation Dose Rates and Shielding

Assumes No External Shield and only 1m of D_2 Reflector
13.6 GW power level and crew habitat at 100 m

Burn Number	Burn Time (sec)	Dose Rate (R/hr)	Dose(R)
1 **Earth to Mars** (60 m H_2)	690	0	0
2 **Mars Breaking** (30 m H_2)	420	0	0
3 **Mars to Earth** Shielding removed during burn (30 m -> 0 m)	190	86	4.5

Figure 14

Safety Features of Fission Fragment Direct Heating Concept

Advantages

- Structure much cooler than Propellant
- Hot surfaces limited to a very small volume
- Low Fissile Inventory (18 kg)
- Redundancy through self contained modular fuel elements
- Short Burn Times 3 - 10 minutes (22 minutes total)
- Almost certain breakup upon reentry or impact
- Subcritical up water emersion (k_{eff}=0.1)
- H_2 worth in module is negative (4 ¢)
- Loss of H_2 and fuel in a few modules; Still Critical
- Low Power Densities (300 w/cm^3)
- No energetic accidents are likely
- Slow progression during accidents

- Increased Reliability
- Tolerant to Power/Flow Mismatch
- Graceful Failure Modes

Disadvantages

- Fission Fragment escape in Exhaust Plume
- Low Structural Mass and Large Size
- Reflector Cooling Mass Penalty

Figure 15

Key Features of Concept

- Gas is Directly Heated by Fission Fragments
  - Cool Structure Relative to Gas/Propellant Temperature
  - Increases Reliability
- Large Dilute Reactor System (requires unique design)
- Moderator Flexibility (D_2O, Be, D_2 liquid or gas, CD_4)
- High Power and Thrust
  13 GW 600,000 lbf
- Fission Fragments Discharged to Space

Figure 16

Key Technology Issues

- H_2 Excitation Physics
- Dilute System Criticality
- Reactor Structural Design (large dilute system)
- Frit/Porous Ceramic Design and Fabrication
- Coating Technology
- Fuel Integrity
  - H_2 Erosion
  - Thermal Gradient
- Reflector Cooling

Figure 17

Critical Tests to Verify Technology

Category	Description	Time	Cost
Physics	H_2 Excitation Radiation	2 yr	5 M$
Scoping Studies	Dilute Systems Criticality Reflector Cooling Structural Design	2 yr	5 M$
Technology Development	Substrate (BeO, Carbides) Coatings (UO_2, {U,Zr}C) Integrity (H_2, Temperature)	5 yr	60 M$
Component Testing	Channel Tests -Prototypic Power,Temp,Flow		
	HFIR, ATR 50-100 W/cm^2	2 yr	20 M$
	FALCON (FTR) 400 W/cm^2	2 yr	20 M$
System Integration Tests	Site Preparation	5 yr	.2 - .5 B$
	Engineering and fabrication	15 yr	.8 - 1.5 B$
	Facility Operation	5 yr	.1 - .3 B$
		Total	1.2- 2.4 B$

Figure 18

Ground Testing

- Overcoat UO_2 films to prevent escape of fission fragments on all modules except one
- 13 GW closed loop with heat exchanger
- 130 MW Open Loop for one Module with Scrubber

Figure 19

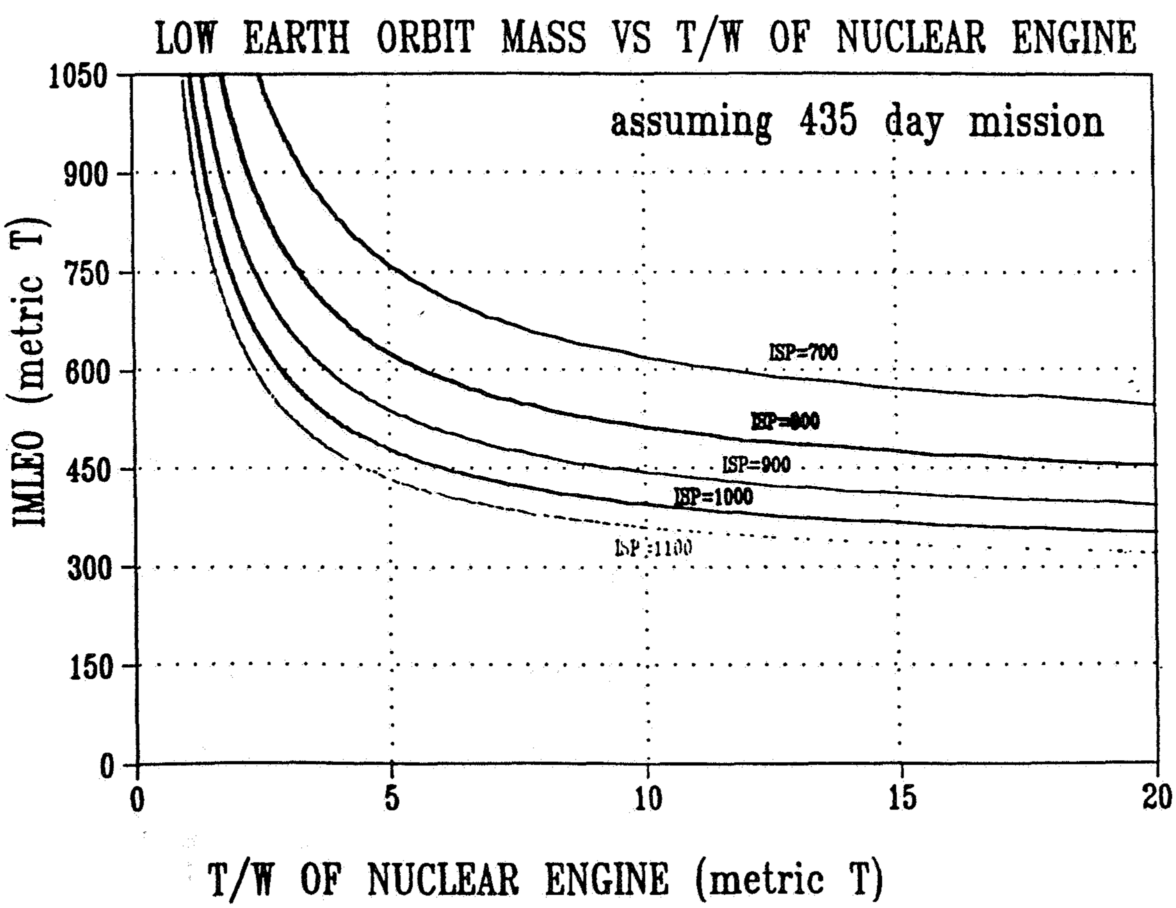


Figure 20

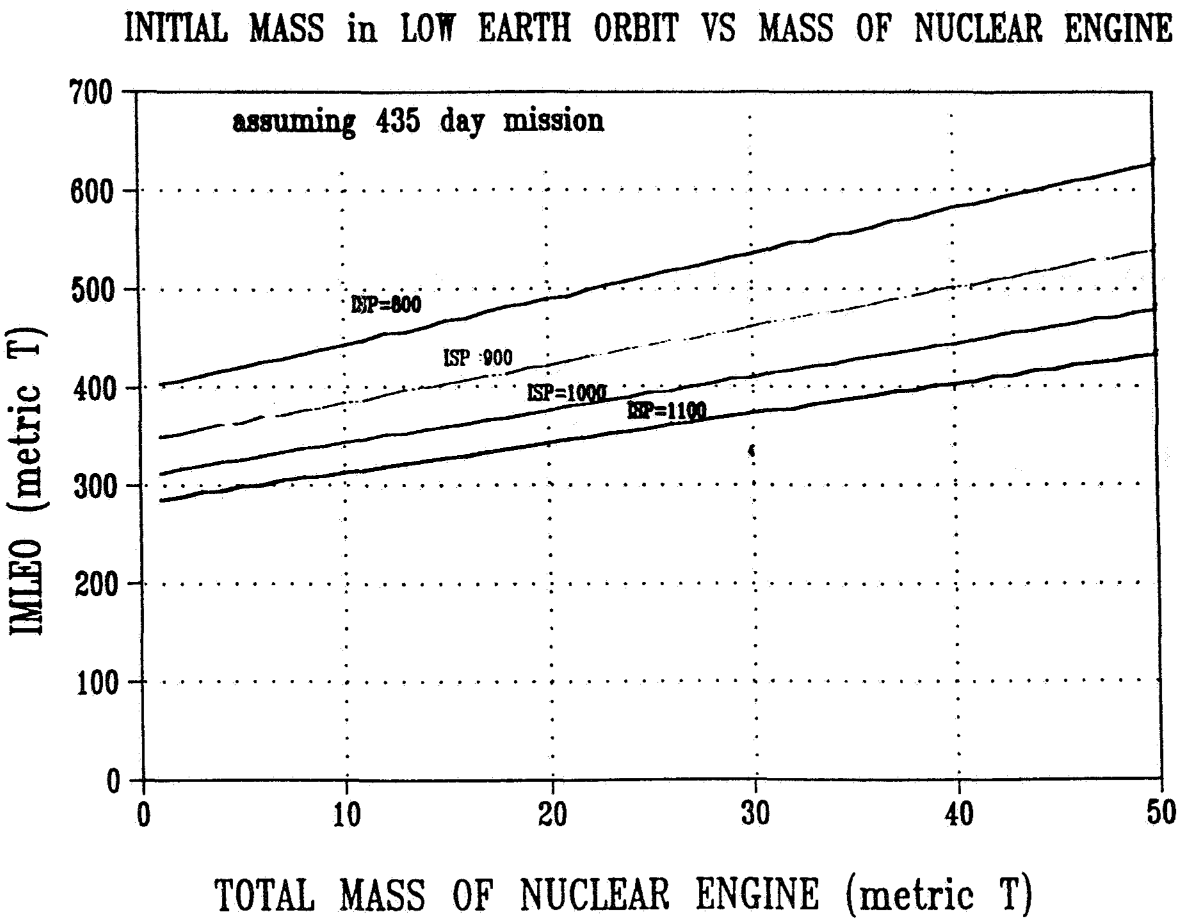


Figure 21

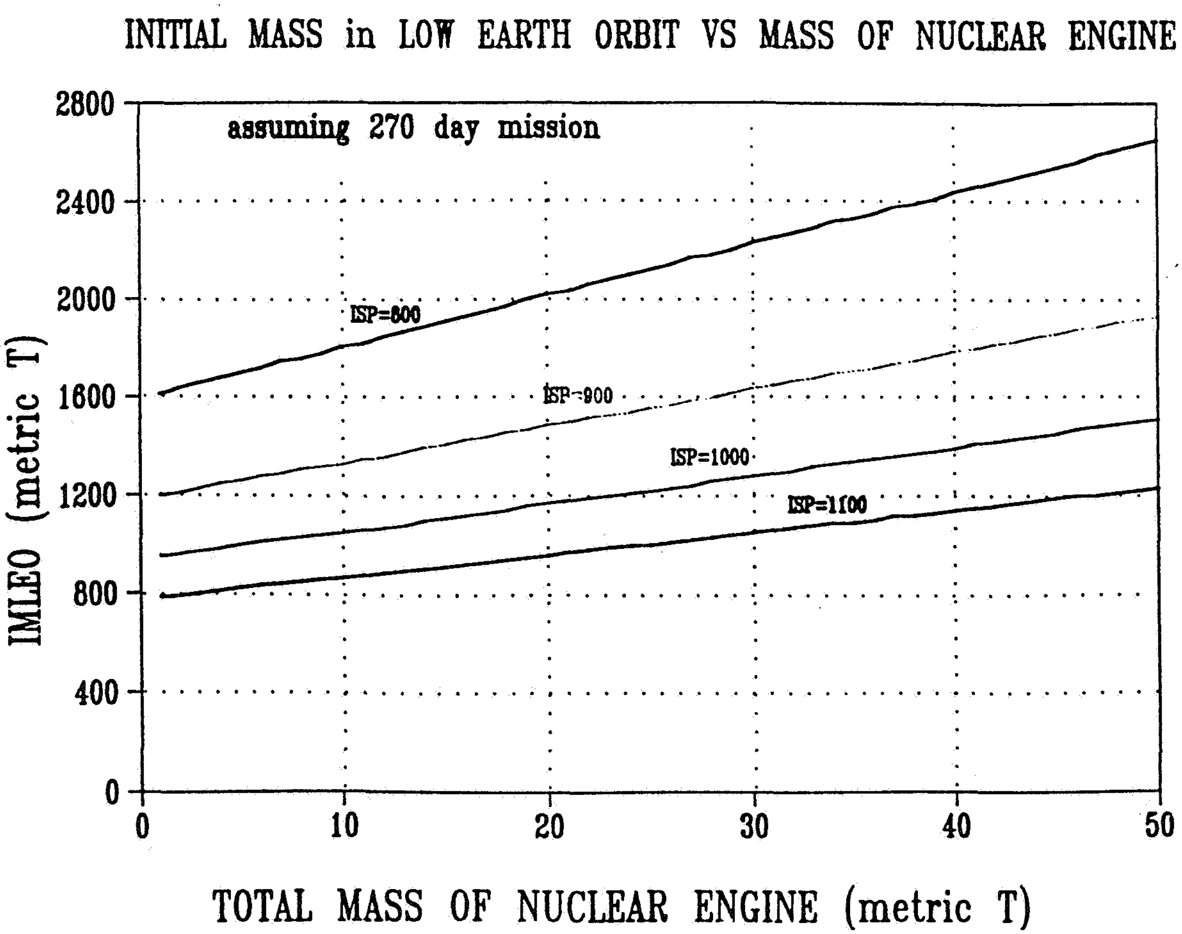


Figure 22

Mission Options

- High Thrust -> More propellant for shorter trip times
  -> Carry more cargo
  -> Carry extra modules equipment for redundancy

- Ideal for a freighter
  Earth Moon
  Planet Robotic Exploration
  Cargo Ship to Mars

- Reusable -> Limited by burnup only

Figure 23

Advantages Direct Heating NTR over Baseline

"Conclusions and Summary"

- Compares favorably to baseline NTR for 435 day mission
  - 10% advantage for short 270 day mission

- Higher potential reliability
  - Lower Structure Operating Temperatures
  - Shorter Burn times (22 min.)
  - Graceful Failure Modes
  - Reduced Potential for High Energetic Accidents
  - Redundancy through modularity

- Part way to very advanced engines, but with reduced risk

Figure 24

N92-11104

LIQUID ANNULUS

Hans Ludewig
Brookhaven National Laboratory

As seen in Figure 1, the specific impulse varies as the square root of the temperature and inversely with the square root of the molecular weight of the propellant. Typical values for specific impulse corresponding to various rocket concepts are shown.

The Liquid Annulus core concept consists of a fuel elements which will be arranged in a moderator block. What is shown in Figure 2 is still a single element. The element rotates about its axis (Figures 2, 3, and 4), with the inner surface molten.

Inlet hydrogen gas enters one end, which would the left side on Figure 2 (may or may not be seeded), flows down the channel picking up heat and exits the other end at 5,000 - 6,000 K. The moderator in this case is beryllium. The other elements would be arranged in a hexagonal pattern; all of them rotating.

The overall coolant path is down through the beryllium, cooling it and then back up through an annulus surrounding the element and then through the hot section. This concept is based on an experiment carried out by Grosse in 1963. He carried out an experiment in which he used liquid alumina and achieved a temperature of over 3,500 K in the central cavity. Figure 4 shows a detail of the Grosse experiment and we see a sectional view of the element. The inner layer would be liquid and the second would be a solid layer backed up by the structural components.

These are the advantages we see for the system (see Figure 5): high specific impulse; structural material will all run at low temperature; lower fission product inventory because of evaporation.

Size estimates were carried out on the concept (see Figure 6-7). Heat radiates from the surface and depends on the temperature. The power is dependent on the emissivities and view factors. Estimates of these figures were obtained from NASA publications. We use emissiivity of 0.4 - 0.8 for our reactor designs.

Using these heat fluxes one can layout a reactor design. We picked seven elements, and basically these are the parameters: 200 megawatts; 7 elements; and a length to radius ratio of 24 (see Figure 7) per element. Once one falls below 5,000 K the reactor gets very massive and the concept loses its appeal. We really have to operate above a temperature of 5,000 K.

Figure 8 shows the reactor parameters for emissivites of f= 0.4 and 0.8; 200 megawatts; 6,000 degrees; ten atmospheres pressure; and seven elements. The fuel element radius (see Figure 9) varies depending on the thickness of the fuel bed. The pitch in both cases is 30 centimeters, the diameter of the total reactor is 110 cm.

We have carried out some first order analysis of heat transfer. However, there is still a lot of work to be done in the analysis of a rotating dissociating gas.

New technologies in fuel development will require investigation of the binary or ternary alloys. i.e., (U,Zr)C and (U,Zr,N6)C. Finally enhanced light weight structures are important. Platelet technology would be useful in a nozzle.

As far as technology issues are concerned (see Figure 10), we need to understand the fluid dynamics and heat transfer of the rotating fuel element. We also need to know about the mass transfer from the surface. Depending on how fast one rotates it, it could act as a centrifuge, but this still has to be studied.

There has to be a mechanism for rotating the elements. A gas bearing at the top to act as a thrust bearing is needed. Nuclear data needs to be acquired at these elevated temperatures.

The values in Figure 11 are the mission parameters for this concept. We would be looking at specific impulses in the range of 1,600-2,000 seconds. The only reason there is a range here is due to the uncertainty of how much uranium will evaporate and end up in the outlet stream. If it is an extreme amount it might be more in the 1,600 range, in which case the concept will lose some of its appeal. About 2,000 seconds is probably the desirable level.

For this particular engine we are talking about 200 megawatts. The approximate thrust level is 20000 N, and the engine mass is about 3,000 kilograms. The other engine would be a little lower depending on the value for emissivity that we use. The thrust to weight level is approximately unity. Without a shield it will be twice that, maybe three times.

Multiple starts and stops in this particular concept are not a problem. In this case one does not require the gas to pass through the liquid, which could freeze. It goes along the surface, so we should be able to start this without any trouble.

Depending on the operating time, one might want to take uranium along and add it to the fuel region, or at least to the seed material. It can be added to the reactor as it is running. It depends on how low the thrust is and how long the operating time is. This is really a mission dependent requirement.

There are five schedule and cost areas that will have to be worked: design; technology; element test reactor; and engine development for the ground; and space qualification. We estimate a cost of around about 1 to $2 billion (see Figure 12).

The really important activity would be the fuel element test. If this is not successful then there is no point in going further. Even at this level one could probably discern whether it is a worthwhile technology or not.

The first year we would develop a plan to test the fluid dynamics and heat transfer in that the rotating element. In phase one we would continue the design work, and we would demonstrate the heat and mass transfer of the prototypic fuel element (see Figure 13).

We would certainly want to carry out a critical experiment. It is not quite clear how one would do that for a system such as this because there would also be a requirement to verify the nuclear properties at the elevated temperatures. Some work will have to be carried out on the nozzle. It is not clear how to design such a nozzle; maybe platelet technology with transpiration cooling.

For phase two and phase three (see Figure 14), we would have to select the test site. This experiment would be very stringent because of the guarantee of losing fission products. An efficient scrubber system would be required.

Critical experiments can be carried out, however, I would like to point out that the machines will have to be modified quite dramatically to do a critical on this reactor. See Figure 15 for the rest of the facility requirements.

As a conclusion, we feel that this concept is worth at least a first look because of the promise of very high specific impulse. Because of the low thrust one would probably need a cluster of engines. This is not necessarily bad because there would be some redundancy, but because of the low thrust one might have to refuel while running. Again, depending on the fuel vaporization, material can be included in the uranium that is injected as one is running along.

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LARS Based Concept

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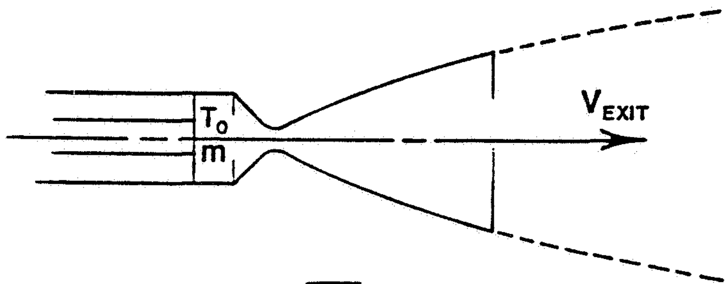
WHY LIQUID CORE ?

HIGHER I_{SP} ;

T NOT LIMITED BY T_{MELT} OF FUEL



$$I_{SP} \propto V_{EXIT} \propto \sqrt{\frac{T_0}{m}}$$

ROCKET TYPE	I_{SP}, sec
CHEMICAL	150–450
SOLID CORE NUCLEAR	800–1200
LIQUID CORE NUCLEAR	1500–2000

Figure 1

LARS CONCEPT

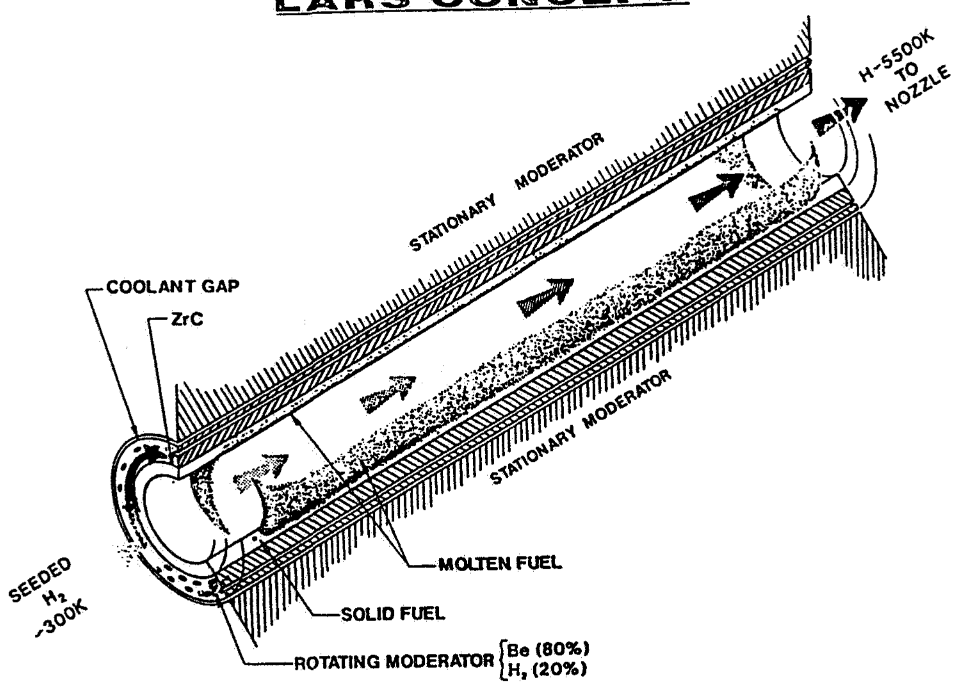


Figure 2

LARS CONCEPT (CONT'D)

KEY FEATURES:

1. MOLTEN FUEL CONTAINED IN ITS OWN MATERIAL.
2. LAYERS STABILIZED BY CENTRIPETAL FORCE.
3. HYDROGEN IS DISSOCIATED AT HIGH T LEADING TO HIGH I_{SP}.

NOTE:

ROTATIONAL CONTAINMENT OF LIQUID REFRACTORIES BY COOLED SOLID OUTER LAYER HAS BEEN DEMONSTRATED BY A.V. GROSSE (Science, 1963).

Figure 3

LARS ROTATING FUEL ELEMENT

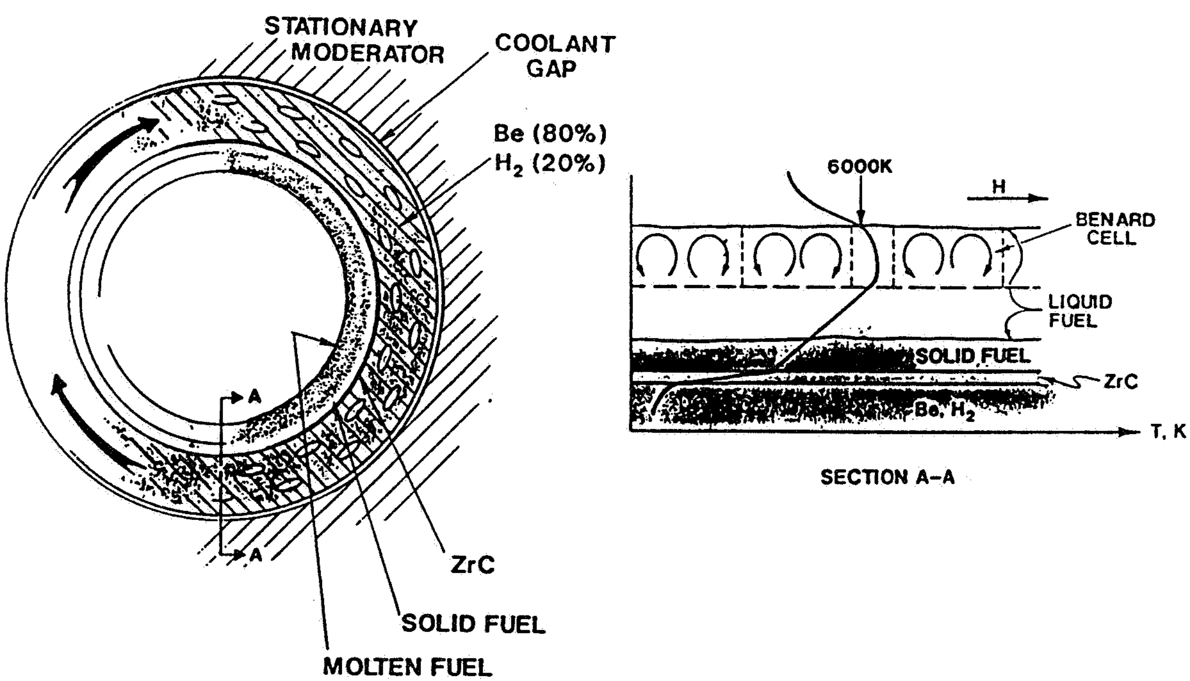


Figure 4

- LARS ADVANTAGES -

- HIGH SPECIFIC IMPULSE
- NO STRUCTURAL ELEMENTS OPERATE AT ELEVATED TEMPERATURES
- POTENTIALLY A SELF-CLEANING REACTOR SYSTEM - FISSION PRODUCTS EVAPORATED INTO DIRECTED EXHAUST STREAM, REDUCING FISSION PRODUCT INVENTORY
- LOWER FISSION PRODUCT INVENTORY REDUCES AFTER HEAT AND COOLANT REQUIRED TO REMOVE AFTER HEAT

Figure 5

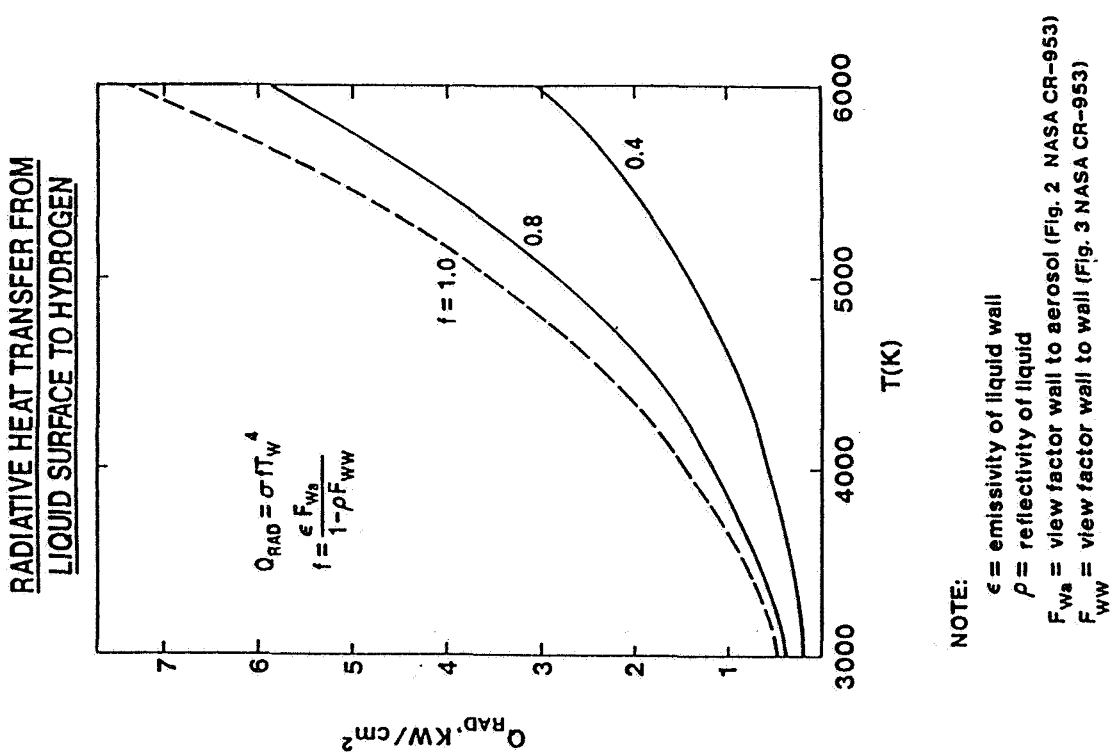


Figure 6

FLOW CHANNEL RADIUS vs T

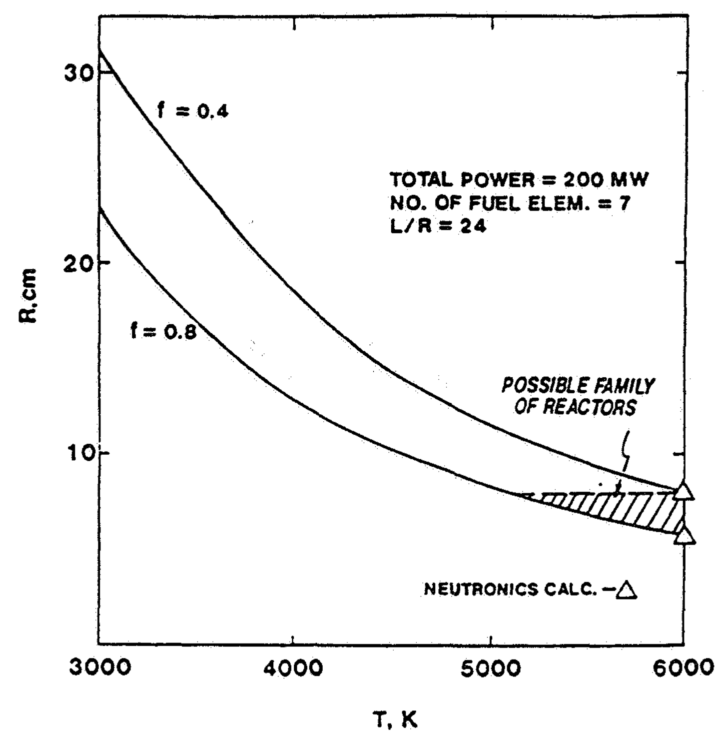


Figure 7

REACTOR PARAMETERS

	f = 0.4	f = 0.8
TOTAL POWER (MW)	200.	200.
OUTLET TEMPERATURE (K)	6000.	6000.
OUTLET PRESSURE (ATM)	10.	10.
NO. OF ELEMENTS	7	7
OUTLET DUCT RADIUS (CM)	8.0	5.6
FUEL BED RADIUS (CM)	9.4	8.1
ROTATING DRUM RADIUS (CM)	12.4	11.1
ELEMENT PITCH (CM)	30.0	30.0
REACTOR O.D. (CM)	110.0	110.0
REACTOR HEIGHT (CM)	192.7	135.5
TOP REFLECTOR (CM)	3.0	3.0
BOTTOM REFLECTOR (CM)	5.0	5.0
RADIAL REFLECTOR (CM)	10.0	10.0
MASS OF UC_2 (KG)	30.0	30.0
MULTIPLICATION FACTOR, K_{eff}	1.08	1.10

Figure 8

LARS CROSS SECTION

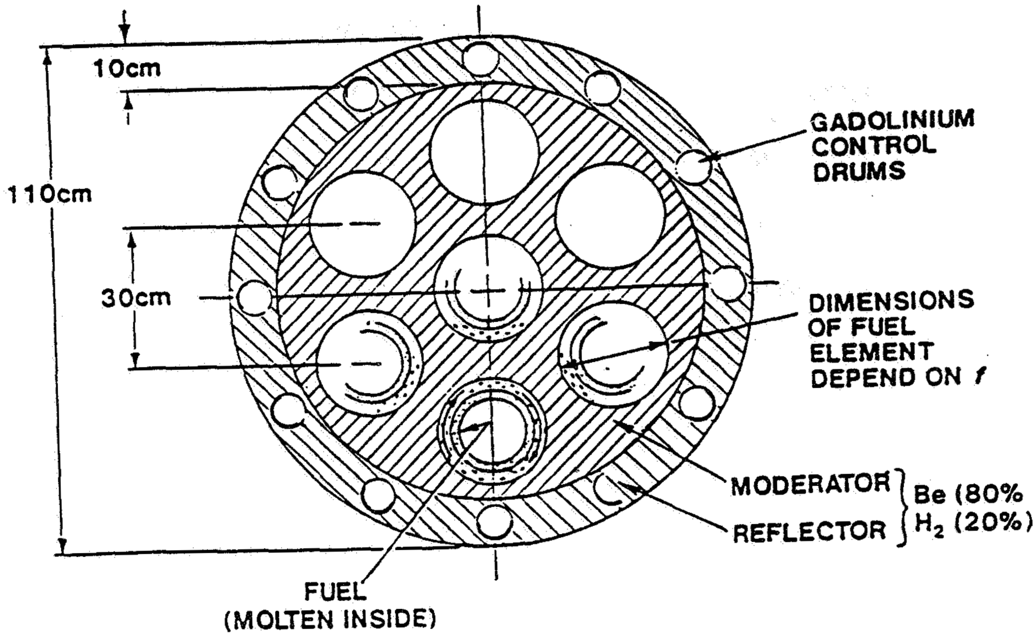


Figure 9

KEY TECHNOLOGY ISSUES

- ANALYTIC AND EXPERIMENTAL UNDERSTANDING OF COOLANT BEHAVIOR IN OUTLET DUCT - BOTH FLUID DYNAMICS AND HEAT TRANSFER

- ANALYTIC AND EXPERIMENTAL UNDERSTANDING OF MASS TRANSFER FROM MOLTEN FUEL SURFACE TO PROPELLANT STREAM

- DEVELOPMENT OF ROTATING MECHANISM AND GAS THRUST BEARING

- NUCLEAR DATA AT ELEVATED TEMPERATURES

Figure 10

KEY MISSION PARAMETERS

SPECIFIC IMPULSE(S)	1600 - 2000
THRUST (N)	2.0 (4)
ENGINE MASS (kg)	3000
REACTOR POWER (MW)	200
THRUST/WEIGHT	1.0

- MULTIPLE STARTS AND STOPS SHOULD NOT POSE ANY PROBLEM SINCE PROPELLANT PASSES OVER MOLTEN SURFACES RATHER THAN THROUGH THE MOLTEN LAYER

- STORAGE OF URANIUM BEARING POWDER MAY BE REQUIRED - DEPENDING ON RUN TIME

Figure 11

SCHEDULE AND COSTS

PHASE: I, II, III

FY: 90 92 94 96 98 00 02 04 06 08 10 12 14 16 18 20

ACTIVITY		COST ($M)
1. DESIGN AND ANALYSIS	1ST YEAR; CDR (ETR, GTE); CDR (FTE)	30
2. TECHNOLOGY DEVELOPMENT	COMPLETED TESTS	50
3. ELEMENT TEST REACTOR	CDR; SITE PREP.; TESTS COMPLETED	350
4. ENGINE DEVELOPMENT AND GTA	SITE PREP. TEST; GTE MANUFACTURE; GTE TEST COMPLETED	570
5. SPACE QUALIFICATION		600
		1600

Figure 12

CRITICAL TEST/ACTIVITIES

- FIRST YEAR
  - DEVELOP ENGINE DESIGN COMPATIBLE WITH MISSION ANALYSIS
  - DEVELOP A PLAN FOR PROOF OF PRINCIPLE AND PROTOTYPIC EXPERIMENTS
  - START EXPERIMENTAL WORK
- CRITICAL TESTS - PHASE I
  - CONTINUE ENGINE DESIGN AND DEVELOPMENT
  - DEMONSTRATE HEAT AND MASS TRANSFER IN A PROTOTYPIC FUEL ELEMENT BOTH ELECTRICALLY AND NUCLEAR HEATED EXPERIMENTS.
  - CARRY OUT CRITICAL EXPERIMENT - VERIFY NUCLEAR DATA AT OPERATING TEMPERATURES
  - VERIFY NOZZLE DESIGN
  - DESIGN ETR

Figure 13

CRITICAL TEST/ACTIVITIES (cont'd)

- CRITICAL TESTS - PHASES II AND III
  - SELECT SITE FOR ETR AND GTE AND SATISFY REGULATORY AND SAFETY REQUIREMENTS
  - PREPARE SITE
  - DESIGN AND CONSTRUCT ETR AND GTE
  - CARRY OUT ETR AND GTE TEST PROGRAM

Figure 14

FACILITY REQUIREMENTS

- CRITICAL EXPERIMENT FACILITY (LANL, ANL (WEST AND EAST))
- FLUID DYNAMICS FLOW FACILITY (NASA LABS)
- SITE FOR ETR - NEW, WILL REQUIRE ALLOWANCE FOR FISSION PRODUCTS IN EXHAUST
- ETR - NEW, MAY BE CONCEPT SPECIFIC
- SITE FOR GTE
- GTE - CONCEPT SPECIFIC
- GTE - ALTITUDE CHAMBER TO TEST START UP

Figure 15

N92-11105

DROPLET CORE NUCLEAR ROCKET (DCNR)

by
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The most basic design feature of the droplet core nuclear reactor (Figure 1 & 2) is to spray liquid uranium into the core in the form of droplets on the order of five to ten microns in size, to bring the reactor to critical conditions. The liquid uranium fuel ejector is driven by hydrogen, and more hydrogen is injected from the side of the reactor to about one and a half meters from the top. High temperature hydrogen is expanded through a nozzle to produce thrust.

The hydrogen pressure in the system can be somewhere between 50 and 500 atmospheres; the higher pressure is more desirable. In this system, uranium droplets are intimately mixed with hydrogen. The fission energy transferred to the gas is 30-40% direct. In the uranium (Figure 2), the mean free path of neutrons is very short; most of the fission occurs close to the surface and from 20-40% of the fission fragments is directly stopped in hydrogen. Heat is also transferred from droplets to the hydrogen directly by conduction and also by radiation. In about one and a half meters from the top, the uranium droplets and hydrogen temperature reach close to 4,000 degrees K. From our own calculations, it is evident that uranium impingement on the wall is a function of droplet size and flow conditions in the core. That's a function of the size of the droplet in the boundary layer. If uranium droplets that are larger than about 30 microns enter the boundary layer, they have a tendency to go toward the wall. However, the hydrogen inflow brings back smaller droplets to the center of the reactor.

In the lower core region (about one and a half meters from the top), hydrogen is tangentially injected to serve two purposes: one, to provide a swirling flow to protect the wall from impingement of hot uranium droplets; two, to generate a vortex flow that can be used for fuel separation.

Tangential injection driven vortex flow is ceased after about one meter, where liquid lithium is injected downward along the wall. After tangential hydrogen is stopped, droplets escape in the direction of their tangential velocities, and land on the lithium-6 film on the wall. Liquid uranium is cooled down on the lithium-6 and flows along the wall to a separator where lithium and hydrogen are separated and the uranium is recirculated to the system.

The hydrogen can reach temperatures of 5,000 to 7,000 degrees, depending on the pressure at which the nuclear engine operates. If pressure is 500 atmospheres, which is what we have used for our base-line analysis, the boiling point of uranium is 9,500

Kelvin. The system can be operated to heat-up the hydrogen to 6,000 degrees or so. That's the basis for our calculations and the conceptual design analysis of DCNR.

As is shown in Figure 2, for the first one and a half meters, uranium and hydrogen are intimately mixed. Because of tangential flow and vortex, the droplets are contained and prevented from impacting the wall. Once the tangential injection in the lower region is removed, heavier fuel droplets move in the direction of their tangential velocities and follow a diverging helical trajectory to the wall. That mechanism provides for fuel separation right at the end of the reactor. The nuclear engine is also designed such that about 70 to 80 percent of the power is generated in the upper part of the reactor. Neutron flux peaks at the upper part of the reactor and is highly depressed at the last half meter of the reactor.

The proposed nuclear reactor is about one meter in diameter and three meters in length, which provides for the type of energy release and power distribution needed to operate at very high temperatures.

Design of this core concept has evolved from the colloid core reactor concept that was proposed in the 1960's (Figure 3 & 4). The colloid core concept utilizes fine particles of uranium-zirconium carbide and vortex flow to confine the fuel particles in the reactor. A very important result of the colloid core study was that Anderson and his colleagues (Figure 3) have demonstrated the vortex flow confinement of the particles. They performed an experiment using tungsten particles and also talcum powder to show that particles indeed can be confined in the core. In the liquid uranium droplet concept, we are not trying to confine droplets in the reactor; they can leave the reactor, and be recirculated. However, in the colloide core concept, complete confinement of the fuel was desired. The effectiveness of the vortex confinement process is rather limited. As a result, the uranium loss might be very significant (six kilograms for six minutes or so), and not acceptable for long missions.

The liquid annulus concept (based on what I have read in the open literature) utilizes solid and liquid uranium compound fuel. Hydrogen is forced to bubble through the liquid fuel to reach 5,000K at the core exit. The reactor core has to be rotated at the rate of 7,000 RPM to contain liquid fuel on the wall. Since hydrogen is bubbled though the liquid uranium at high velocities, the fuel loss due to forced vaporization and entrainment can be very high, and beyond the level acceptable to any mission.

In 1987-88 we developed the droplet core reactor concept, primarily for a multi megawatt space power reactor system (Figure 4). For the past few years, we have studied the properties of droplet fuel transport, heat transfer, thermal hydraulics, neutronics, and material aspects of this concept.

The droplet core concept is different from the rotating liquid core concept mainly due to the fact that uranium is not confined in the core but is actually recirculated. Based on our studies we have concluded that due to the axial velocity of uranium, it is very difficult to achieve effective core confinement. In the droplet core reactor concept, we try to redirect and bring the droplets close to the wall and then separate and re-circulate them.

Uranium has a very broad and stable liquid phase (Figure 5). At 500 atmospheres, uranium melts at 1,400 K and boils at about 9,000 K.

To spray liquid uranium in very small sizes, hydrogen should be injected at velocities ranging from 500 to 2,000 meters per second. This is in the nozzle spray system; to obtain smaller droplet sizes (<5microns), one has to blow hydrogen at higher velocities (Figure 6). However, once the gas comes into the reactor, the average velocity drops to tens of meters/sec. At the top region of the reactor, the average velocity is about four to five meters per second, and near the core exit it is about 30 to 40 meters per second. At these moderate velocities, the uranium droplets and the hydrogen gas do not have a significant relative velocity, which minimizes the forced evaporation of uranium.

In mid-core region, the established method of vortex flow is used to keep droplets having very high temperature away from the wall. The reverse process is used in lower core region to separate the droplets and bring them toward the wall where it is injected by lithium-6. Since lithium, as you know, has enormously high latent heat of vaporization (21MJ/kg), it provides a lot of heat sink capacity to cool uranium droplets from about 6,000 or 6,500 degrees to about 2,000 degrees, which can then be handled in the fuel storage and recirculation system.

As for the hydrogen transport, this system relaxes two major design restrictions. First, the rocket engine is not thrust limited because the hydrogen flow rates can be very high. The liquid uranium volume in the core is about two liters, so loading in a core of this size is about 20 kilograms. The total volume of the core is about 2,5000 liters, so the void fraction for this system is about 99.9. Therefore, hydrogen is practically free flowing, and the mass flow rate of hydrogen is unlimited by core losses and so is the thrust. The heat transfer area is not limited; about 40 percent of the fission fragments energy is directly deposited into the hydrogen propellant. Furthermore, the heat transfer area for fuel droplets is very large. It is about four orders of magnitude larger than any other non-colloid fuel reactor concept.

Another important feature of this nuclear propulsion concept is that it can augment the Isp beyond the temperature limits by radiation-induced dissociation and subsequent recombination of the hydrogen. Since the reactor is operated at a high temperature, even at 500 atmospheres pressure, thermal dissociation of hydrogen is significant (Figure 7). At this pressure, in addition to 20 percent dissociation at 6,000 degrees, there is nuclear enhanced ionization of the hydrogen.

Primary and secondary electronics that are generated by fission fragments increase the dissociation of hydrogen. The dissociation energy of these hydrogen molecules is fully recovered after expansion through the nozzle because of their non-equilibrium conditions. This is on the top of the thermal dissociation that can further enhance the Isp for the system. For the baseline design, a total of 20% dissociation and recombination is assumed. This leads to 2,000 seconds of Isp.

Materials play a very significant role when you are talking about such extreme temperatures (Figure 8). Tungsten and tantalum are the only two refractory metals that are fully compatible with uranium. Uranium neither dissolves nor forms any kind of metallic or chemical bond with these metals. However, uranium at high temperature attacks both tungsten and tantalum by diffusion through the grain boundary.

But if we use single crystal tungsten or tantalum, the granular attack by uranium can be mitigated. There is existing technology for growing single crystal tungsten, but for tantalum it is still under investigation.

Many other high temperature materials that have become available in recent years can help with the development of high Isp rocket technology. For example, tungsten-rhenium-hafnium carbide alloys (Figure 9) have outstanding mechanical properties at temperatures above 3,000 K, even up to 3,400 K. These alloys have been demonstrated to have acceptable mechanical properties that can be used as structural materials for large reactor vessels.

Let's summarize the basic design features of (Figure 10) uranium fuel droplets and hydrogen propellant when they are intimately mixed. The energy transfer, in addition to direct deposition of fission fragments, is through the high surface area of droplets. In this system, the fuel surface area density (m^2/cubic meters of fuel) is about four orders of magnitude larger than solid core reactor concepts.

Very high propellant temperature can be reached in this rocket engine (3,000 to 7,000K). For the baseline analysis we have used 6,000 degrees. The hydrogen flow rate in this systems is not restricted by fuel heat transfer area.

If pumping power is available, hydrogen can be pumped though the system even at 1,000 kilograms per second. There is no limiting factor for hydrogen flow, although a very high rate of hydrogen flow for this mission is not needed. For the desired thrust for this concept, 17 kilograms per second of hydrogen should be actually pumped though the reactor.

For the baseline design, at 6,000 K propellant temperatures and 20 percent dissociation and recombination, an Isp of 2,000 sec. is calculated.

There is also a very important safety feature for this system. The reactor can be loaded in orbit. Uranium powder can be used for initial start-up. Therefore, the reactor does not need to be launched with the fuel in the core.

Another feature that could be a liability, (or also could be a benefit of this concept), is that most of the fission fragments escape from the core. It reduces the radioactive loading of the reactor. This is the good aspect of the design if radioactive material release is acceptable.

From my standpoint, the fission fragment release is a benefit of this design. This is a good safety feature of this system, because the reactor is drained of radioactive materials and removes the shielding requirement for non-prompt radiation. This allows for reactor repair after initial start-up.

Low uranium loading is needed for this concept. About 20 kilograms of 95% enriched uranium is needed to reach critical conditions. If the system is optimized, it is expected to reduce the core loading to about ten kilograms. This core loading is defined based on minimum uranium-235 concentrated in the core. In my calculation, it is 20 kilograms of fully enriched uranium. The total inventory of about 100 kilograms of uranium is circulated in the system.

The reactor is designed to maximize the energy generation in the upper region of the core. This is where we have a thick reflector (Figure 11). The core is three meters in length. Figure 14 shows the thermal flux and the fast flux in the core and the reflector. As you can see, the power decreases in the lower core region where lithium-6 is injected. In this region the flux goes down by four to five orders of magnitude. At the end of the reactor, the power generation is minimized.

Again, to summarize (Figure 12), the system can result in an Isp of 2,000 seconds, and a thrust-to-weight ratio of 1.6 for the shielded reactor. The nuclear engine system can reduce the Mars mission duration to less than 200 days. It can reduce the hydrogen consumption by a factor of 2 to 3, which reduces the hydrogen load by about 130 to 50 metric tons.

The engine dimensions are as follows: the inner diameter is basically one meter and the core is three meters in length. The total length of the engine and the reactor is about 13 meters and the thrust to weight ratio with the shield is about 1.6.

The hydrogen flow rate and the reactor power can be scaled up without changing the core dimensions. The same reactor can be made critical and can be operated at different power levels and thrusts. For the baseline, the hydrogen flow rate is 17 kilograms per second and it can go to 25 or maybe even to 150 kilograms per second without changing the dimensions of the reactor.

The only change that can be made by increasing the hydrogen flow rate is the uranium injection rate into the core. This is also regulated by hydrogen flow that drives the liquid uranium spray system. The maximum level is ultimately limited by the nozzle flow capability. This is obviously a major problem. To expand molecular and atomic hydrogen at 6,000 K through the nozzle, the heat flux would be extremely high and beyond the current technological capabilities. It may force a reduction in the maximum core outlet temperature. However, this is a generic problem for all advanced concepts.

Critical technical issues that need to be addressed are listed in Figure 11. I do not have time to go though all of them. One is modeling of the uranium droplet transport in the hydrogen. This needs to be tested, and the energy transfer process must be analyzed. Droplet fuel separation and uranium loss must be accurately analyzed, especially uranium loss. If uranium loss due to evaporization is high, we have to seed hydrogen with depleted uranium hydride. Uranium hydride at 500 atmospheres dissociates at 1,200 K; it becomes uranium vapor and hydrogen. This can suppress the enriched uranium evaporation and loss. The loss of depleted uranium from the nozzle results in a penalty of 5-10% loss in Isp.

The hydrogen driven uranium spray nozzle design needs to be investigated. For mercury and helium, the weight ratio is not as high as uranium hydrogen.

After separation from lithium and hydrogen, liquid uranium fuel is pumped and recirculated. Pumping of liquid uranium at temperatures and flow rates of interest to this concept has not been done yet. Although the technology of pumping uranium compounds in molten salt reactors is well developed, the forced recirculation and pumping of liquid uranium have to be investigated.

The materials compatibility and fabrication technology for refractory alloys are issues that must be investigated. Last but not least, the rocket nozzle design for operation in a molecular and atomic hydrogen environment at about 6,000 degrees is another key technical issue that must be investigated. And here basically I would like to stop.

A VOICE: You have the gas being rotated in the lower part of the vessel in order to send the uranium to the wall and be collected, whereas in the upper part you want the uranium to be in a colloid. Yet if the rotational motion of the gas in the lower part will also cause the gas in the upper part to rotate to some degree, that will send your colloid to the wall as a liquid layer.

How can you maintain a colloid under those conditions?

MR. ANGHAIE: You mean droplet liquid?

A VOICE: It seems to me that the gas rotation in the lower part of the vessel will translate into some gas rotation in the upper part. It will take this mixture of droplets and cause them to precipitate against the walls of the machine and, thus, instead of getting this very large heat transfer area with all these droplets and suspension, you will just get a simple liquid layer along the perimeter of the vessel.

MR ANGHAIE: For separation of uranium you do not need very high rotational velocity. Then the forces acting on droplets are (1) the drag force (2) the thermoforetic force that is due to the temperature difference and that pushes the uranium droplets toward the wall; and (3) the dynamic force due to the pressure difference between the walls and the core centerline, which is moving at maximum velocity. This force tends to bring droplets toward the center.

In the upper regions of the core, the pressure difference due to lower velocity at the wall and higher velocity at the center keeps droplets away from the wall. With the type of rotational velocity and because of uranium density that is 300,000 times larger than the density of hydrogen, we don't believe that very high rotational velocities are needed. Therefore, the rotational momentum added in the mid-core regions cannot diffuse to the upper core region. Furthermore, even if droplets would go to the wall, the temperature of uranium droplets is not more than 3,000 to 4,000 degrees in that region, and, again, hydrogen is being injected in the upper core region so it doesn't seem to be a problem in this regard. However, optimization has to be done regarding the balance of the vortex flow containment and separation of liquid uranium.

A VOICE: Well, the rotational lower part has to be substantial or else the uranium will be convected out the bottom of the machine. It won't have time to fall against your collection device unless it is a high G force.

MR. ANGHAIE: It really doesn't need high G force. Once you remove the force, uranium droplets just escape in the direction of their tangential velocities. The axial velocity in this system is about 20 to 30 meters in the upper part. What you need is a velocity of a few meters per second -- because the drag force by axially flowing hydrogen is not that large. Calculation has shown that vortex flow separation is a serious problem; however, this problem has to be further investigated.

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"Numerical Study of High Temperature Supersonic Flow Under Influence of Magnetic Field and Strong Internal Heat Source", J.G. Zhang and S. Anghaie, submitted for presentation at the <u>AIAA 22nd Fluid Dynamics, Plasma Dynamics and Lasers Conference</u>, Honolulu, Hawaii, June 24-26, 1991.

"Numerical Simulation of Supersonic MHD Flow", J. Zhang and S. Anghaie, submitted for publication at the <u>Fourth Symposium on Transport Phenomena</u> (ISTP-4) - Heat and Mass Transfer, Sydney, Australia, July 14-18, 1991.

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"Stability of MHD Plasmas Ionized by Nuclear Fragments", Y. Watanabe, J. Appelbaum, I. Maya, *abstract* submitted to the <u>26th Intersociety Energy Conversion Engineering Conference</u>, Boston, Massachusetts, August 3-9, 1991.

"Gas Core Reactor Concepts and Technology", N.J. Diaz, S. Anghaie, E.T. Dugan and I. Maya, *abstract* submitted to the <u>AIAA/NASA/OAI Conference on Advanced SEI Technologies</u>, Cleveland, OH, September 4-6, 1991.

"Heat Transfer Analysis of Fuel Assemblies in Heterogenous Gas Core Nuclear Rocket", Y. Watanabe, J. Appelbaum, N.J. Diaz, I. Maya, *abstract* submitted to the <u>AIAA/NASA/OAI Conference on Advanced SEI Technologies</u>, Cleveland, Ohio, September 4-6, 1991.

"A Nuclear Driven Metallic Vapor MHD Coupled with MPD Thruster", S. Anghaie, *abstract* submitted to the <u>AIAA/NASA/OAI Conference on Advanced SEI Technologies</u>, Cleveland, Ohio, September 4-6, 1991.

"Ultrahigh Temperature Vapor Core Reactor-MHD System for Space Nuclear Electric Power", I. Maya, S. Anghaie, N.J. Diaz and E.T. Dugan, *abstract* submitted to the <u>AIAA/NASA/OAI Conference on Advanced SEI Technologies</u>, Cleveland, Ohio, September 4-6, 1991.

"Reduced Gravity Forced Convective Boiling", Masaki Misawa and Samim Anghaie, paper submitted for presentation at the "ICONE-1"; the Japan Society of Mechanical Engineers and the American Society of Mechanical Engineers, <u>(JSME-ASME), First Joint International Conference on Nuclear Engineering</u>, Tokyo, Japan, November 4-7, 1991.

"Electrical Conductivity of UF_4 - K/KF Gas Partially Ionized by Fission Fragments", Y. Watanabe, J. Appelbaum and I. Maya, paper submitted for publication in "Nuclear Science and Engineering Journal", 1991.

"Corrosion Mechanism of Yttria by UF_4 and its Morphological Aspects", Z. Erkmen, S. Anghaie and S.C.P. Wang, paper submitted for publication in "Nuclear Technology", February, 1992.

DROPLET CORE NUCLEAR ROCKET (DCNR)

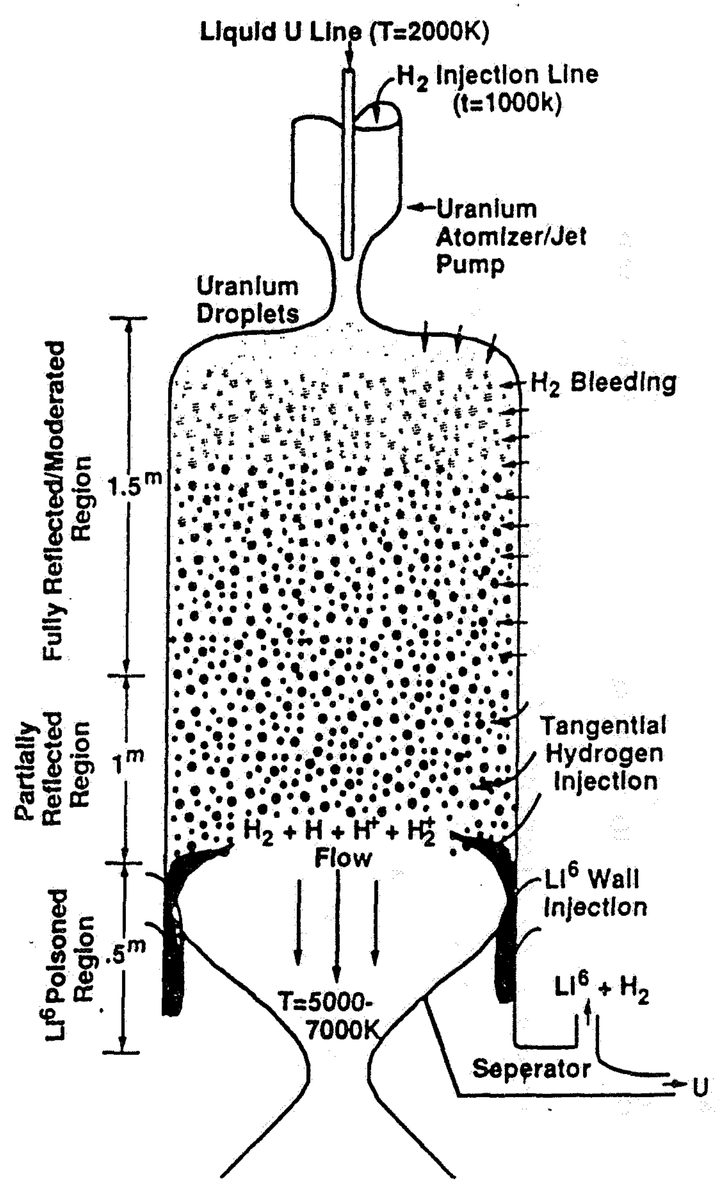

Figure 1

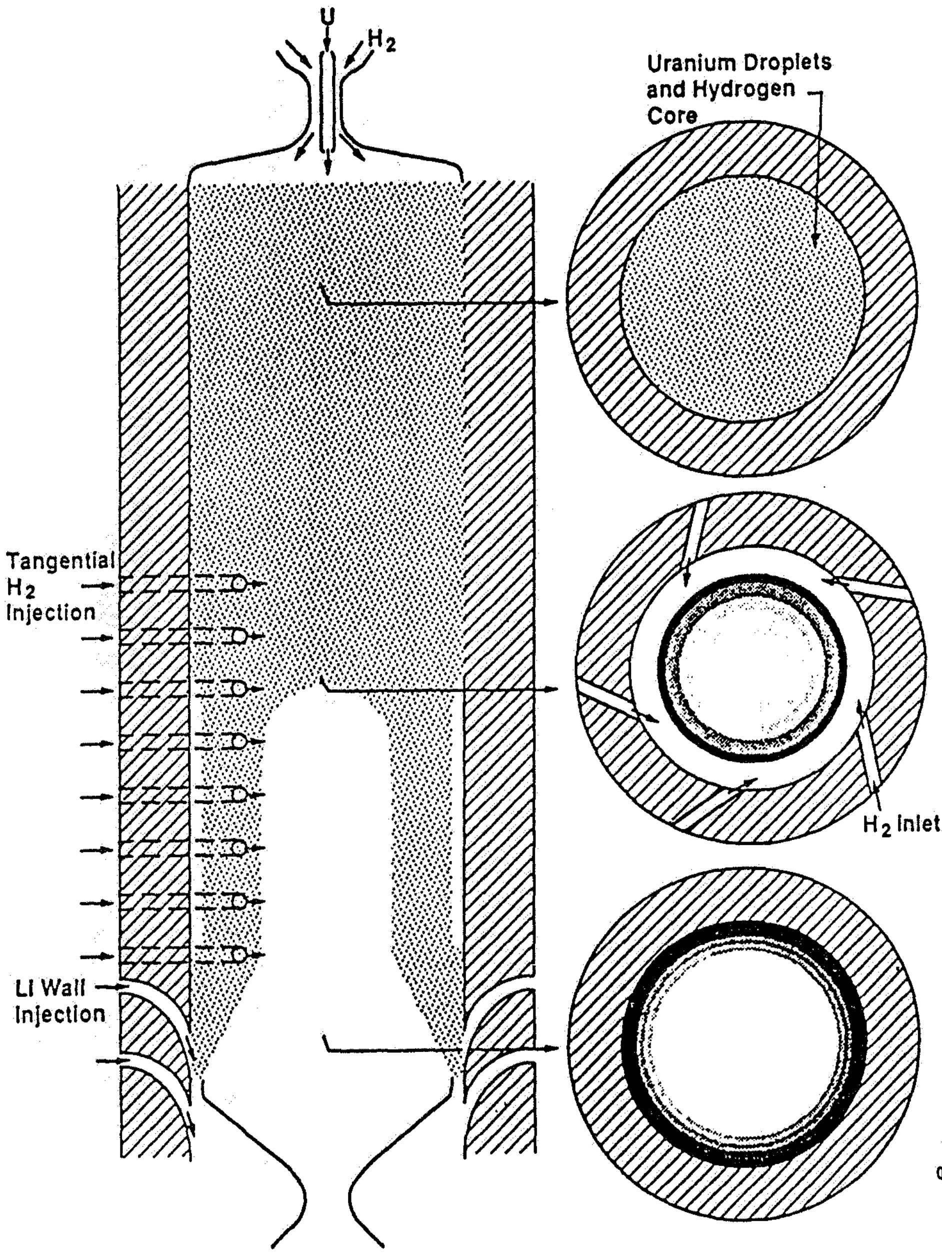

Illustration of Liquid Uranium Droplet Fuel Distribution in DCNR (out of scale).

Figure 2

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EVOLUTION OF DROPLET CORE REACTOR DESIGN

1. COLLOID CORE CONCEPTS (Y.S. TANG ET AL. 1970)

   * U-C-ZR FINE PARTICLES CONFINED IN A VORTEX FLOW CAVITY
   * COMPACT CORE, T=3700K, ISP=1100S, T=20,000LB
   * VORTEX PROPERTIES OF COLLOID CORE REACTOR WERE DEMONSTRATED (L.A. ANDERSON ET AL., 1972)
   * VERY HIGH RATE OF U LOSS (100 G/S)

2. LIQUID CORE CONCEPT (J.P. MCGUIRK, 1972)

   * CORE CONTAINMENT USING CENTRIFUGAL FORCE (ROTATING AT 7000 RPM)
   * FORCING HYDROGEN TO BUBBLE THROUGH UC-ZRC LIQUID FUEL
   * T=4800K, ISP=1500S, T=9000LB
   * HIGH RATE OF URANIUM LOSS, LACK OF A RELIABLE MECHANISM FOR ROTATION AT 7000 RPM

Figure 3

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EVOLUTION OF DROPLET CORE REACTOR DESIGN

3. DROPLET CORE REACTOR (S. ANGHAIE 1988)

   * RECIRCULATION OF URANIUM INSTEAD OF CONFINEMENT
   * UTILIZATION OF VERY STABLE URANIUM LIQUID PHASE (@ 500 ATM T_{MELT} = 1400K, T_{BOIL} = 9500K)
   * FULL ENTRAINMENT OF DROPLETS SIGNIFICANTLY REDUCES THE FORCED EVAPORATION AND MINIMIZES THE URANIUM LOSS (LESS THAN 50 KG/MISSION)
   * ESTABLISHED METHOD OF TANGENTIAL INJECTION INDUCED VORTEX FLOW IS USED FOR WALL PROTECTION AGAINST URANIUM DROPLETS AND SUBSEQUENT SEPARATION.
   * MAXIMIZES HYDROGEN FLOW AREA AND RELAXES THRUST LIMITATIONS (2500<T<400,000LB)
   * NUCLEAR ENHANCED DISSOCIATION OF HYDROGEN INCREASES ISP (1500<ISP<3000S)

Figure 4

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VAPOR PRESSURE OF URANIUM AS A FUNCTION OF TEMPERATURE

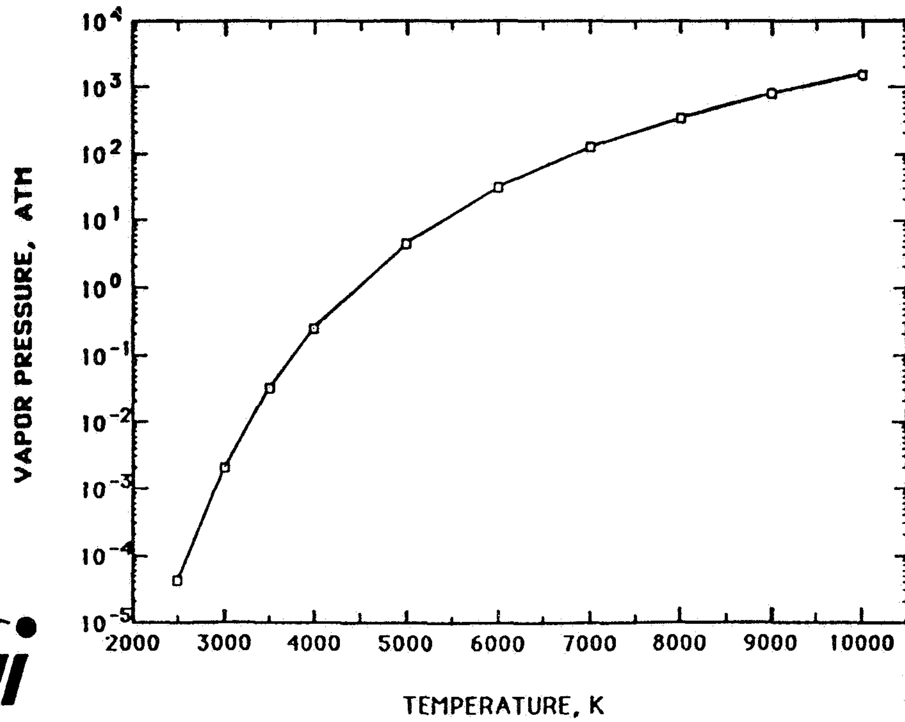

Figure 5

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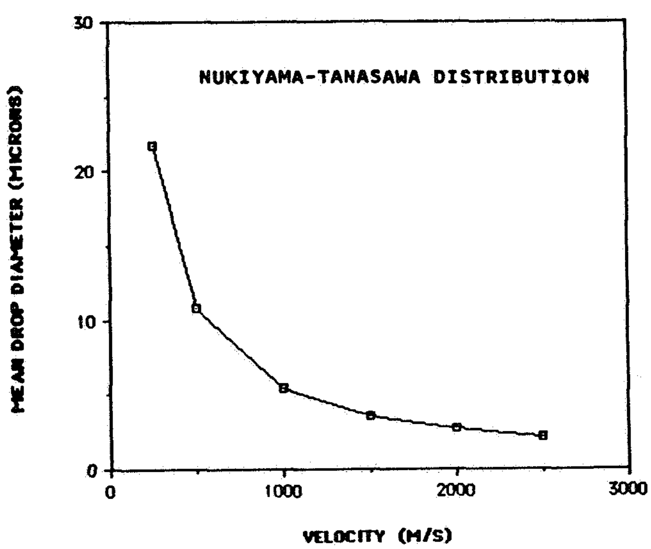

AVERAGE URANIUM DROPLET SI[illegible] AS A FUNCTION OF HYDROGEN FLOW VELOCITY IN THE NOZZLE S[illegible] SYSTEM

Figure 6

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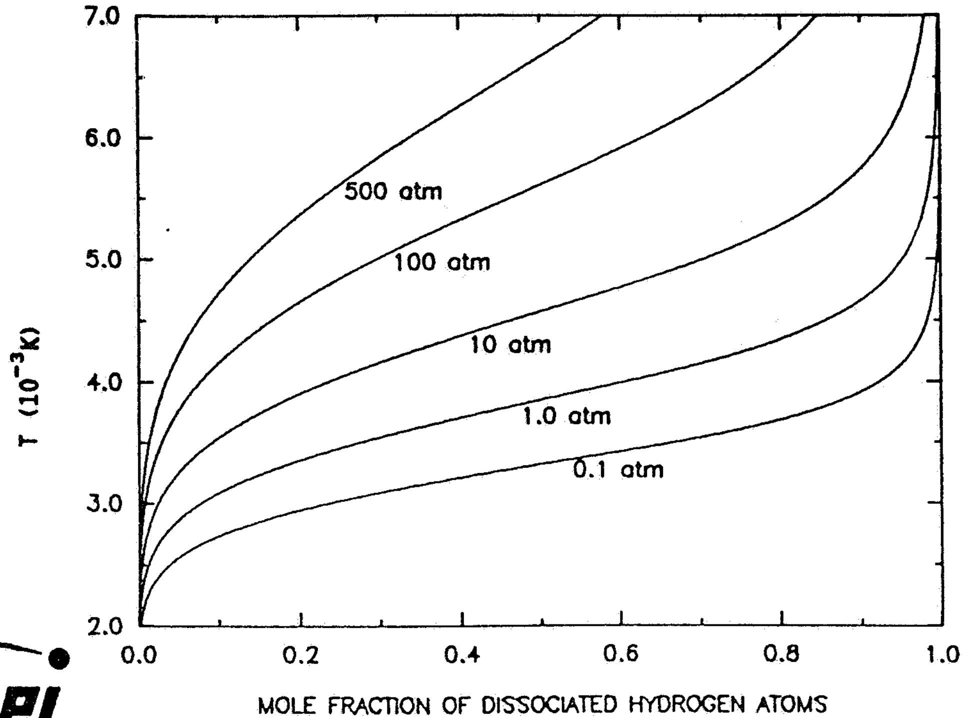

Figure 7

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NEW MATERIAL TECHNOLOGY

REQUIREMENTS: **SUPERIOR THERMAL CAPABILITIES AND ULTRAHIGH TEMPERATURE MECHANICAL PROPERTIES**

SINGLE CRYSTAL TUNGSTEN, W

SINGLE CRYSTAL TANTALUM, TA

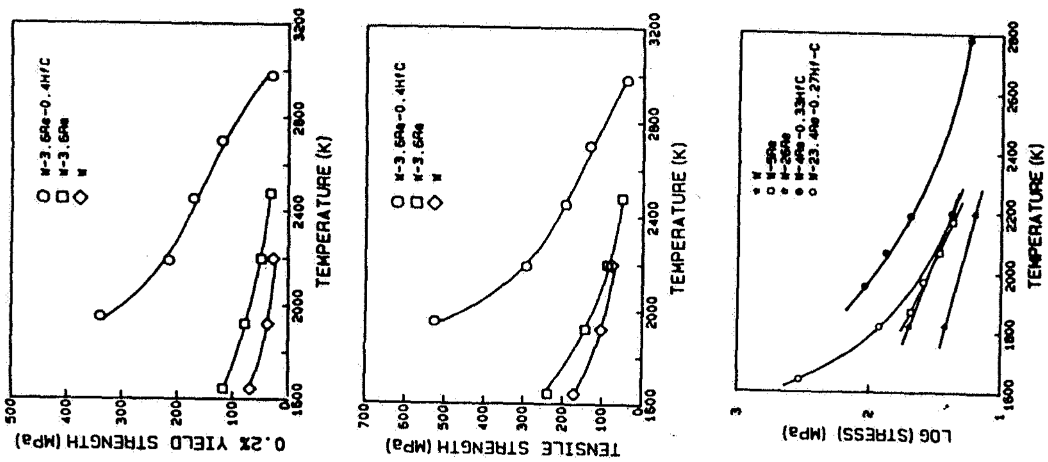
GLOSSY CARBON (~1% POROSITY)

W-RE-HFC ALLOYS (W-3.6RE-0.4HFC, W-4RE-0.33HFC, T_M ~ 3700K)

W-THO_2 ALLOYS (W-1THO_2, W-2THO_2)

T-222 (TA-10W-2.5HF-0.01C)

Figure 8



COMPARISON OF THE YIELD STRENGTH, TENSILE STRENGTH AND CREEP STRENGTH OF W-BASE ALLOYS

Figure 9

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DROPLET CORE NUCLEAR ROCKET (DCNR)

1. DROPLET FUEL AND PROPELLANT ARE INTIMATELY MIXED.
   - * ENERGY TRANSFER SURFACE AREA DENSITY ($>10^6$ M^2/M^3)
   - * ABOUT 1/2 OF FISSION ENERGY IS DIRECTLY DEPOSITED TO PROPELLANT.
   - * DROPLET FUEL PROVIDES MORE THAN 3 ORDERS OF MAGNITUDE IMPROVEMENT ON HEAT TRANSFER AREA.

2. HIGH PROPELLANT TEMPERATURES AND FLOW RATES WITH VERY LOW FUEL LOSS

   HYDROGEN TEMPERATURES ~ 3000 TO 7000K
   HYDROGEN FLOW RATES ~ 1 TO 1000 KG/S

3. HIGH DEGREE OF NONEQUILIBRIUM DISSOCIATION OF HYDROGEN MOLECULES DUE TO FISSION FRAGMENTS

   ISP = 2000 S (@ T=6000K AND 20% DISSOCIATION/RECOMBINATION)

4. VERY HIGH THRUST-TO-WEIGHT RATIO. (NUCLEAR THERMAL ROCKET, RADIATION SHIELDS AND ASSOCIATED POWER GENERATION SYSTEM)

   THRUST-TO-WEIGHT RATIO = 5 AT 75 KLB (333KN), 1500 MWE
   1.6 (SHIELDED)

5. IMPROVED SAFETY FEATURES
   - * IN-ORBIT FUEL LOADING
   - * A LARGE PORTION OF RADIOACTIVE FISSION FRAGMENTS LEAVE THE CORE
   - * LOW URANIUM LOADING (ABOUT 20 KG IN CORE AND 100 KG TOTAL)

Figure 10

INDPI

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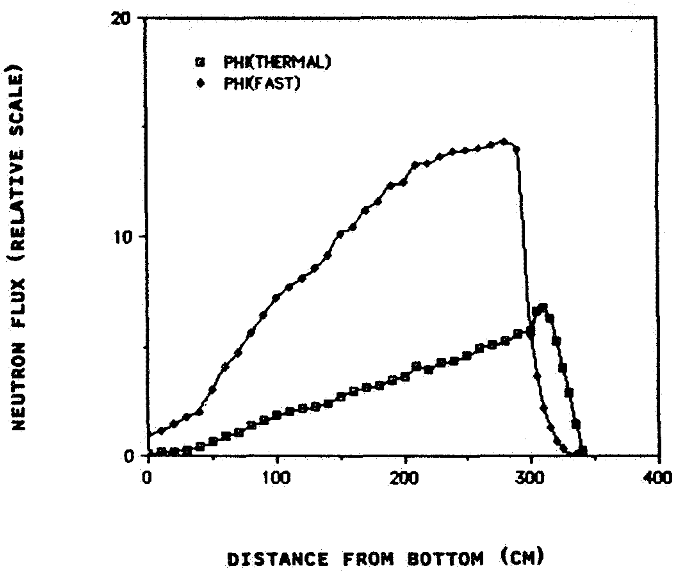

CROSS-SECTIONAL AVERAGE AXIAL FLUX DISTRIBUTION IN DCNR
(E_{TH}<1.8EV, E_{FAST}>1.8EV)

Figure 11

INDPI

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DROPLET CORE NUCLEAR ROCKET CHARACTERISTICS

* REDUCES MISSION DURATION TO LESS THAN 200 DAYS

* REDUCES HYDROGEN PROPELLANT CONSUMPTION RATE BY A FACTOR OF 2 TO 3

* SYSTEM DESIGN PARAMETERS

<u>ISP</u> 2000 SEC

<u>ENGINE DIMENSIONS:</u>

REACTOR I.D./LENGTH	1/3 M
REACTOR O.D./LENGTH	2/4 M
ENGINE LENGTH	13 M
THRUST/WEIGHT (UI ELDED)	5
THRUST/WEIGHT (SHIELDED)	1.6

Figure 12

INNOVATIVE NUCLEAR SPACE
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CRITICAL TECHNICAL ISSUES

* MODELING OF TWO-PHASE FLOW DYNAMICS AND ENERGY TRANSFER
* DROPLET FUEL SEPARATION AND URANIUM LOSS
* HYDROGEN DRIVEN URANIUM SPRAY NOZZLE DESIGN
* RECIRCULATION OF LIQUID URANIUM
* MATERIALS COMPATIBILITY IN ULTRAHIGH TEMPERATURE LIQUID U, LIQUID AND VAPOR LI AND HYDROGEN ENVIRONMENTS (CORROSION, EROSION, INTERGRANULAR ATTACK...)
* ROCKET NOZZLE DESIGN FOR 6000 TO 7000K HYDROGEN ENVIRONMENT OPERATIONS

Figure 13

N92-11106

COMMENTS ON DUAL-MODE NUCLEAR SPACE POWER & PROPULSION SYSTEM CONCEPTS*

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I. INTRODUCTION

These comments are based on a rather intense period of work in The Aerospace Systems Laboratory at Princeton University around the early 1970s and some later efforts by the authors and others in the Techno-Systems Analysis Corporation and RCA Astro through the mid-1970s and into the 1980s. We have been convinced for many years that some form of Dual-Mode Nuclear Space Power & Propulsion System (D-MNSP&PS) will be essential to spacefaring throughout the Solar System and that such systems must evolve as mankind moves into outer space.

II. BACKGROUND

The earliest work on the dual-mode derived from a need to dispose of the nuclear rocket reactor afterheat which was used to generate auxiliary power for long durations and reduces the quantity of hydrogen required for cooldown and the duration of cooldown thrusting. John H. Beveridge of Aerojet Nuclear Systems Company presented this first paper in 1971.

Subsequent thinking led to the mathematical modeling of concepts wherein large amounts of thermal power would be taken continuously at appropriate temperatures for conversion to electrical power from a specially configured rocket reactor. Dual-mode operation provides relatively high-thrust accelerations from the direct thrust mode and low-thrust accelerations with higher effective jet velocities from electric thrusters. Detailed conceptual designs of D-MNSP&PS for specific missions should be undertaken and compared with other means for carrying out such missions in the context of an overall evolving space program.

* Prepared for the Nuclear Thermal Propulsion Workshop - A joint NASA/DoE/DoD Workshop,July 10 - 12, 1990,Cleveland,Ohio.

III. DUAL-MODE NUCLEAR SPACE POWER & PROPULSION SYSTEM CONCEPTS

It must be kept in mind that space power and propulsion systems will evolve from present systems according to the demands of an ongoing program of diverse elements. Chemical rockets will continue to be used for the indefinite future for a variety of missions. After nuclear space power and propulsion systems and electric rocket concepts are proven they may be used advantageously for many classes of space missions especially those requiring high energy and complex flight paths.

A. Direct Nuclear Rocket Thrust

Nuclear rockets of the Rover(aka NERVA) Program were the most highly developed of nuclear space propulsion devices. The Small Nuclear Rocket Engine (SNRE) design introduced at the very end of this program was well proven and could be an excellent starting point at around 375 MWt for re-initiation of development and test work. With "slush" hydrogen as the propellant contained in insulated tanks for unmanned heliocentric missions, especially round trips, from and to transport nodes in long-lived Earth orbits lived Earth orbits. Much study and analysis is needed to identify the optimum nuclear rocket systems, vehicle configurations and flight paths for comparison with other means for performing such a mission.

B. Nuclear Electric Rocket Thrust

Nuclear electric rockets for primary propulsion have been delayed in their development by the lack of suitable nuclear space power systems which were denied funding following the cancellation of space nuclear propulsion and power programs in 1973. High effective jet velocities of electric rockets require large power supplies to provide even the low thrust accelerations and long thrusting periods that are characteristic of this form of propulsion. Their primary applications are missions that have flight paths in heliocentric space away from massive bodies.

The Kaufman electron bombardment ion thruster was an excellent development effort at the NASA Lewis Research Center; while arcjets, magnetoplasmadynamic and other electric rocket thrusters with a variety of propellants have also been developed. However an electric rocket thruster with proper characteristics for cruising throughout the solar system has yet to make its appearance.

C. Combinations

Mission analyses have shown that optimum performance of advanced Solar System missions, especially manned missions, requires combinations of propulsion (and power) systems.

D. Proposed Dual-Mode Reference System Design Concepts

1. Overall System

The Initial D-MNPSP&P Reference System should be based on (a) present(1990) and (b) advanced(1995) technology for use on comparable missions in the 2000 and 2005 time period respectively. The technology bases must assume a continuity of research, technology and advanced development work during the period on all vehicle subsystems; although this does not appear to be likely in the present funding circumstances world-wide, and especially in the USA.

Modification of the LANL Rover reactor at 1500 MWt or SNRE with 375 MWt full rocket power could also provide continuous (87,600 h) thermal power at lower levels of 150 to 35 MWt respectively and at a constant temperature of 1500 K for use by an efficient closed cycle electrical power generation systems producing between 45 and 10 MWe. The waste heat rejection subsystems would make use of deployable heat pipe radiators.

The advanced systems should be based on proposed concepts that have been clearly defined and appear to be realizable before the end of the 20th century for missions in the second decade of the next century. The overall character of these systems should be represented by new materials, sophisticated concepts, higher powers and temperatures, very high reliability and operational safety. These advanced systems require systems and mission analyses that are parametric, probabilistic and detailed, but must also be basically realistic.

Very advanced systems need to be defined and analysed, but should be handled on a separate basis that emphasizes research and technology aspects of major components; e. g., type of second or third generation reactor. Sensitivity analyses need to be conducted, and parametric studies need over appropriate ranges to give an overall understanding of the systems characteristics.

2. Major Subsystems

Dual-Mode Nuclear Space Power & Propulsion Systems can conveniently be broken down into a number of subsystems as given below. Although various concepts may have other subsystems or lack some of those shown; for the present purposes these should suffice for the Reference System definition.

a. Nuclear Subsystem

The Nuclear Subsystem includes the energy source and controls for the release of thermal power at elevated temperatures. In the D-MNSP&PS the thermal power is removed for two purposes. The lower power is released over the entire lifetime of the system once it has attained a long lived Earth orbit, and the lower power components will be maintained at a constant operating temperature of 1500 K (or higher for advanced systems).

(1) Reactor Types

There are a number of reactor types that are capable of being configured for dual-mode use. The gas (hydrogen) cooled epithermal carbide core that is fueled with enriched uranium oxide is heart of the LANL Rover and SNRE reactors. Some fuel element development for long duration, multiple thrusting periods at maximum temperature will be required. The capability for operation for 10 y at lower temperature and power levels will also represent a development challenge. Dual-mode operation of these and other types of reactors is a very substantial challenge that must be met by conceptual design effort and research and technology work including systems and component analyses. Mission analysis must also be performed before the D-MNSP&PS can be defined and related to the reactor type.

(2) Nuclear Radiation Shields

The D-MNSP&PS require much more substantial shields than the "shadow" shields that are ordinarily provided because the reactor operates continuously albeit at lower than rocket power levels. Something between a 2 pi and 4 pi tailored shield with cooling provisions will need to incorporated;however, although the additional mass must be accounted the transport of this mass can be discounted by Lunar exploitation activity. Vehicle conceptual design will be conditioned by nuclear radiation sheilding considerations.

(3) Thermal Power Source Heat Exchanger

One of the major problems in realizing the D-MNSP&PS capabitity is the removal of thermal power from the core for the generation of electric power. Depending on the type of reactor this may be accomplished in several ways. A pumped loop may be placed in or near the core and connected to the electric generation system. Specially configured heat pipes may be placed in the core where they would serve to remove thermal power and also act as supports for the core. One of the difficulties is to arrange for these elements to operate at a prescribed lower temperature even during rocket thrusting,and they must operate for the life of the vehicle with minimum mass and very high reliability.

b. Power Conversion Subsystems

(1) Direct Thrust Nozzles

Direct thrust would be provided by hot hydrogen flowing through a conventional nozzle as in a Rover engine;however the reactor would be fastened to the vehicle structure. The expansion portion of the nozzle would be movable at the throat and control the vehicle in pitch and yaw. Roll control would make use of auxiliary jets. Some research,technology and development work will be required to realize this capability.

(2) Thermal to Electrical Conversion Systems

(a) Closed Cycle Brayton Systems

The electrical generation systems that meet the requirements for dual-mode operation with high efficiency and long lifetime are the Brayton cycle gas turbine power systems that have had much development attention for other applications. Recent developments that are important for space use include new high temperature materials and foil bearings. A considerable amount of analysis and development aimed at specific characteristics are needed before the Brayton systems can be unqualifiedly selected for dual-mode application.

(b) Other Systems

Before other power conversion systems can be considered seriously much analysis and some technology work is needed so comparisons and selections can be made.

(3) Electric Rocket Propulsion Systems

Electric rocket thrusters are discussed in Section III.B. above,but a propulsion system includes other elements such as:power conditioning units,thruster clustering and control,plume control and emi considerations. Propellant tankage and control also need attention. The primary problem remains to identify the kind of thrusters for the D-MNSP&PS and to proceed with development for test and use.

c. Waste Heat Rejection Subsystems

Primary heat rejection radiators for D-MNSP&PS have large areas and temperatures around 1000 K;and often need to be deployed and provided with meteoroid protection. Advanced developments with new materials and working fluids have been made in recent years,but more work is needed. Mission specific radiators need to be designed,developed and tested. A number of auxiliary waste heat sources are found throughout the system and the vehicle but they are generally of lower temperature and can be dealt with locally.

d. Control and Safety Subsystems

The problems of control and safety are unusually severe because the D-MNSPS&PS is a very complex system and must be controlled over a wide range; in fact it is not altogether clear how it can be controlled and made safe. Much analysis and testing is needed to begin to answer these questions.

3. System Disposal Concepts

Work has been in progress for a number of years on the disposal of nuclear power sources in outer space by the United Nations Scientific and Technology Subcommittee and others. I has been generally concluded that they can be safely operated and disposed of if careful provisions are made and carried out responsibly. It is the United States position that nuclear reactors should not be started below altitudes with orbital lifetimes sufficiently long for radioactive species to have effectively decayed. D-MNSP&PS must have provisions for deployment to remote orbits where collision with orbital objects is nil and orbital lifetime is infinite.

IV. POSSIBLE 21st CENTURY DUAL-MODE MISSION APPLICATIONS

Generic Missions of the early years of the 21st century can make excellent use of the D-MNSP&PS vehicles and in a few years they become essential as the space program of the period evolves. The Table presented below shows missions of the 2000 to 2020 period where use of the Dual-Mode System should be evaluated; it should be understood that both the D-M and the missions will be evolved substantially in the course of the period.

TABLE

Possible Early 21st Century Dual-Mode Mission Applications

Geo-centric Operations
- Nuclear Operational Station
- Cargo Operations in Earth Orbits

Cis-Lunar Missions
- Unmanned and Manned Lunar Shuttles

Lunar Exploitation
- Lunar Resources
- Lunar Bases
- Lunar Observatory

Helio-centric Missions
- Asteroids and Minor Bodies
  = Unmanned Exploration
  = Manned Exploration
  = Manned Exploitation

- Martian Missions
  = Unmanned Round Trips
  = Manned Expeditions
  = Martian Bases

- Other Solar System Missions
  = Outer Planet and Moons Orbiters and Landers
  = Outer Planet Moons Explorers
  = Outer Planet Round Trips
  = Trans-Neptune Explorers

All D-MNSP&PS vehicles will depart from and return to a geo-centric operational station at an altitude that has an orbital lifetime of more than 300 years. This transport node will have an inclination that facilitates the nuclear vehicles that it will service.

It is anticipated that D-MNSP&PS vehicles will operate in cis-lunar space for training purposes with cargos of opportunity.

The first major dual-mode missions will probably consist of unmanned and manned asteroid explorations. Such missions could be of consequence for some period of time.

Martian missions will probably be carried out on a global basis with a very ambitious scenario that utilizes a wide variety of chemical and nuclear propulsion.

Other solar system and galactic missions will follow after the first two decades and will make maximum use of the power and propulsion technology that has been brought into being.

N92-11107

NUCLEAR THERMAL/NUCLEAR ELECTRIC HYBRIDS

B. D. Reid
Battelle Pacific Northwest Laboratory
Reactor Technology Center

First I would like to describe the nuclear thermal and nuclear electric hybrid. The concept isn't new; as our previous speaker indicated, there had been some work done in the early 1970's, and I will briefly describe some of that again.

We evaluated a hybrid concept, and I will be describing its specifications and its mission performance. Then, as requested by our workshop organizers, I will discuss technical status, development requirements, and we, like everyone else, will provide some optimistic cost estimates.

Essentially (Figure 1), we see the hybrid working as a concept whereby you have both thermal propulsion and electric propulsion. We see this concept being used when you would use a thermal propulsion, high thrust thermal propulsion for your trans-Mars/trans-Earth injection burns, and using electrical propulsion in transit (Figure 2).

This is all using one reactor. There are differences in the reactor performance when it's a one mode versus the other. In the thermal propulsion mode, the reactor is operating at, let's say, 1500 megawatts. In the electric production mode, it's of the order of 2 to 3 percent of that, which translates to about 35 megawatts thermal.

In looking at hybrids there are options in the design. One option uses common heat transfer passages, while the other uses independent heat transfer passages. There are pluses and minuses to each of these options. I will show a schematic of each of them to explain them a little more fully. With the independent heat transfer concept, you do have the possibility of providing electrical power during your thermal propulsion cycle.

This is a rough schematic of a common cycle hybrid (Figure 3). During open cycle propulsion, you close two valves and it essentially operates just like a standard NERVA. In the power production mode, the other two valves are closed. Argon is used both in the closed cycle power conversion and to provide your propellant for your electric propulsion; in this case we have chosen MPD thrusters.

Here is an independent cycle (Figure 4). In this schematic for the open cycle propulsion, the hydrogen flows through the reactor, and again the open cycle propulsion just like a NERVA. During the power conversion cycle, you can operate this cycle independent of the open cycle propulsion, using the argon through separate cooling passages and also through your MPD thrusters.

With this slide I am going to repeat some information that you already have seen today (Figure 5). As part of the NERVA/ROVER program, dual mode reactors were looked at. The primary incentive was to reduce propellant losses which were required to address decay-heat removal.

In this case the NERVA concept was operating at 365 megawatts in its thermal propulsion mode. For the electrical propulsion, it was operating at only 1 megawatt with only approximately 25 kilowatts electricity to an organic Rankine cycle. For this there were very few engine modifications required, primarily materials.

This is another cartoon of the same concept that you saw previously (Figure 6). During the thermal propulsion mode, the power conversion cycle is essentially cut off and the turbine is driven by hydrogen flow through the core support tie-tubes. When the reactor is not in the thermal propulsion mode, this circuit could be energized for power conversion.

We evaluated a hybrid concept for the manned Mars mission. It is based on a 1500 megawatt NERVA with 850 seconds specific impulse. We chose Brayton cycle with argon, with 8 megawatts electricity, and 35 megawatts thermal. And again, we are using the MPD thruster at 5000 seconds specific impulse. In the concept that we have chosen here during thermal propulsion (Figure 7), two valves are closed: it operates just like a standard NERVA. During the power conversion cycle, the other two valves are closed, and we have a valve in the nozzle.

Now, we recognize there are some significant challenges with this, and we were a little loath to put this up here because we thought it might fail the "snicker" test. But after careful study, we think that it's a concepts that will work. Again, the advantage with this type of cycle is that you can achieve a reasonable power production with this concept. The earlier concepts were very limited in the power you were able to obtain from the core because you had very limited heat transfer surfaces. This concept operates at about 1400 Kelvin, which is well below the operating temperature of the core in the thermal propulsion mode. The Brayton cycle will operate at about 150 psi.

This chart shows some of the performance specifications of our concept (Figures 8 & 9). These thrust/weight numbers here, I would like to caution you, are only for the base reactor. The lifetime for this concept in the thermal propulsion mode is similar to NERVA, or approximately 10 hours. In the electric propulsion mode, we have convinced ourselves that you have a life of at least 2 years, which is sufficient for the reference mission. Figure 9 is a mass breakdown. This is for a mission that was less than 600 days, 555 days to be exact.

One of the advantages of the hybrid concept (Figure 10) is that you do have the ability to provide electricity for housekeeping loads and, with space power beaming, you have the ability to provide significant power to meet other mission objectives such as supporting a base on the Martian surface.

As noted, this concept is primarily based on the NERVA technology, and there have been a lot of improvements in that technology that can be incorporated here, primarily in fuels and materials for nozzles and core materials. We believe that the primary uncertainties with this concept are going to lie with the other part of the power conversion system; with the Brayton cycle.

Figure 11 summarizes in chart form our anticipated technology readiness. As most speakers have touched on today, the reactor is at readiness level approaching 6. Our greatest uncertainty lies in integrating the two power conversion cycles, and we have identified that accordingly (Figure 12).

Here is our cost estimate (Figure 13). It's a rough order of magnitude, and it's as optimistic as the next guy's.

Summing up, in our mission benefits (Figure 14), we have identified modest reductions in trip time compared to the nuclear thermal only, at the same masses. Said another way, you have reduced masses compared to nuclear thermal options with similar trip times. Now, a key to all this is that incorporation of the dual mode concept into a thermal reactor, you must not significantly degrade the specific impulse of your thermal reactor, or you have lost everything that you are going to gain with your electric propulsion. So, a key in coming up with this design is making sure that you haven't hurt yourself in Isp.

As to safety issues, these are primarily those associated with NERVA, and other people can probably address those more capably than I. However, there is a minimal additional risk associated with incorporating this additional system that we can't ignore. It does provide some finite increase in risk.

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Dual Mode Hybrid Concept

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PNL NTP/NEP CONCEPT DESCRIPTION

- Based on 1500 MWth NERVA for thermal propulsion
  - 850 Isp using hydrogen as propellant
- Generation of 8 MWe from Brayton cycle at 35 MWth during electrical propulsion
  - Employs MPD thruster at 5000 Isp using argon as propellant and Brayton cycle working fluid

Battelle

Figure 1

NTP/NEP HYBRID DESCRIPTION

- Nuclear thermal propulsion for TMI and TEI burns supplemented by nuclear electric propulsion during transit
- Characteristics of NTP/NEP Hybrid
  - Considerable power differences between open cycle mode and closed cycle mode
  - Option for common or independent heat transfer passages in reactor
  - Possibility of providing electrical power while in thermal propulsion mode

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Figure 2

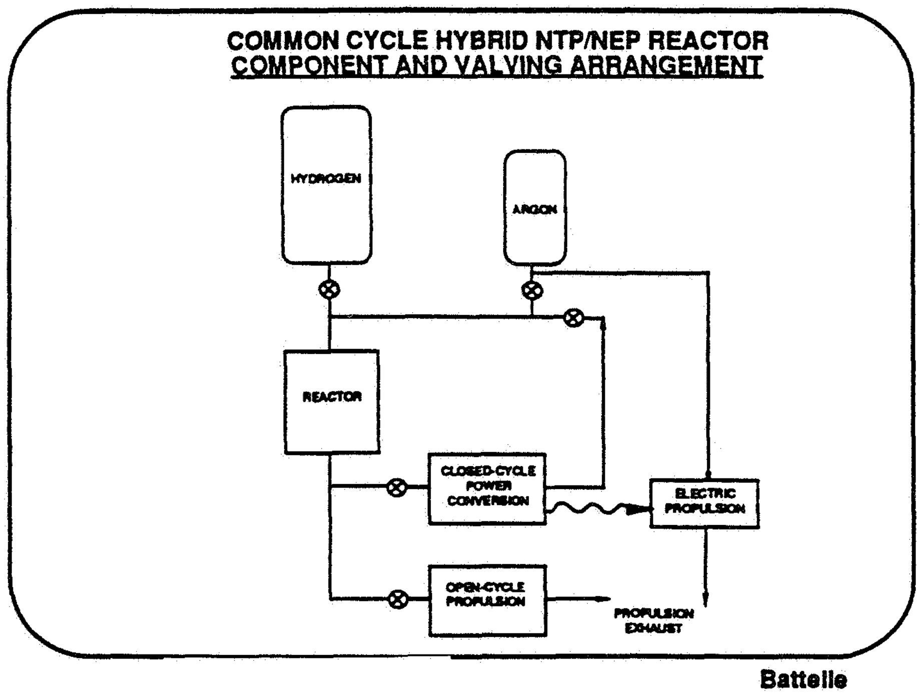

Figure 3

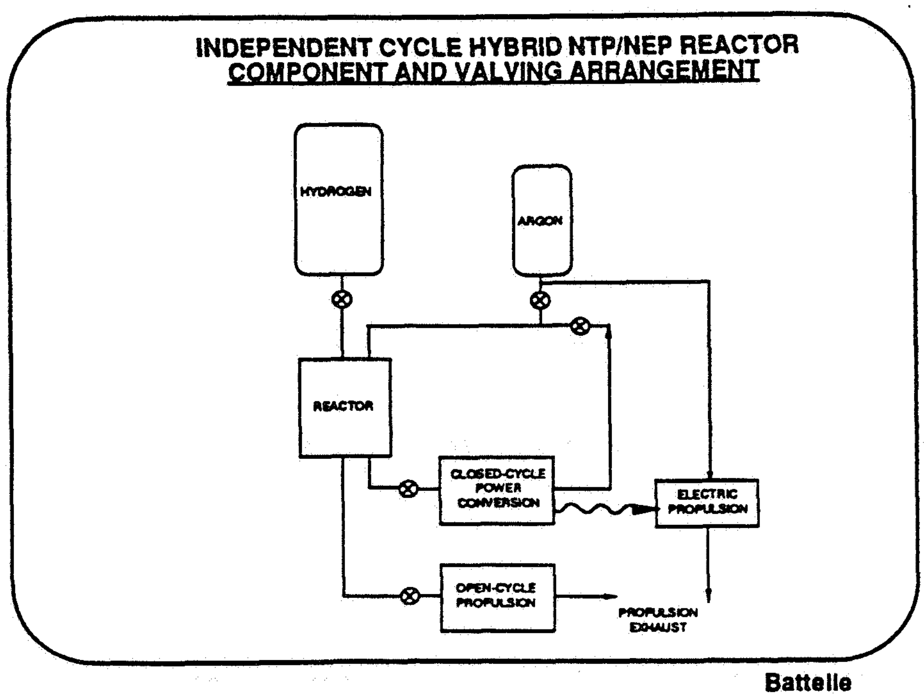

Figure 4

HISTORY OF NTP/NEP CONCEPTS

- Dual mode Nuclear Rocket proposed during ROVER Program in early 1970's
- Based on 365 MWth NERVA
  - Operating at 1 MWth generating approximately 25 kWe from Rankine Cycle
- Concept required few engine modifications
  - Change reactor dome design
  - Change tie tube support line materials

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Figure 5

EARLY DUAL MODE NUCLEAR ROCKET CONCEPT

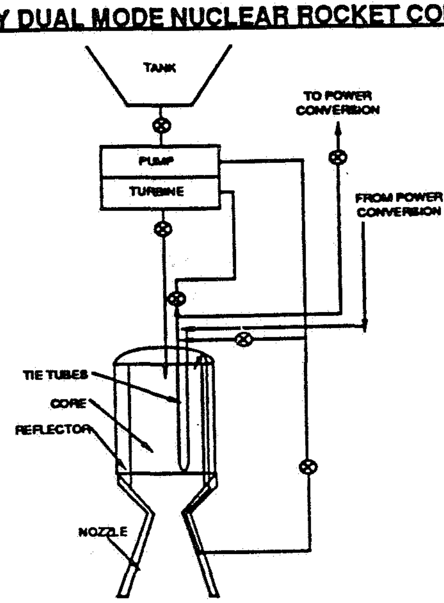

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Figure 6

PROPOSED HYBRID NTP/NEP REACTOR CONCEPT

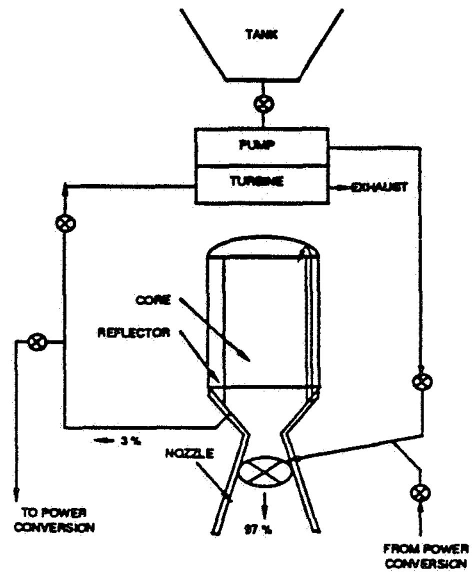

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Figure 7

NTP/NEP HYBRID PERFORMANCE

Propulsion:

Thermal	850 Isp	75,000 lb thrust
Electric	5000 Isp	15 lb thrust

NTP Thrust/Weight

With shield	4.7 without electric thrust
Without shield	7.7 without electric thrust

Lifetime:

Thermal - approximately 10 hours
Electric - greater than 2 years

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Figure 8

NTP/NEP HYBRID PERFORMANCE (CONTD)

IMLEO:

Payload	124 MT
Propulsion Subsystem*	16 MT
Propellent (hydrogen / argon)	384 MT
Total	524 MT

*reactor, shield, radiator, armor, thrusters

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Figure 9

MISSION PERFORMANCE OF NTP/NEP HYBRID

- Trip time
- Initial Mass in Low Earth Orbit (IMLEO)
- Increased commonality and redundancy
  - Electric production for housekeeping loads
  - Redundant / alternate means of propulsion
  - Electric production for surface application through space power beaming

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Figure 10

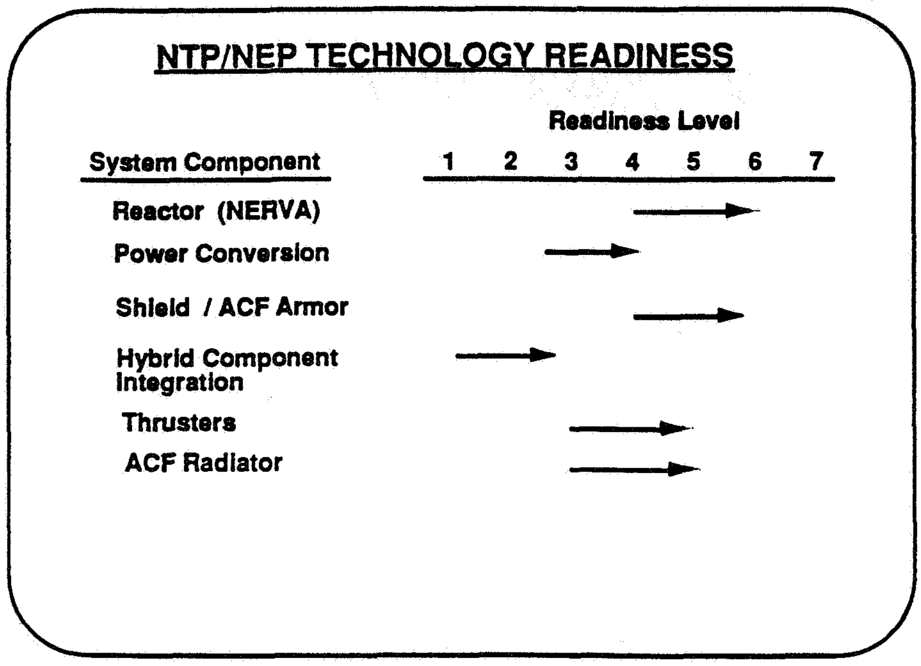

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Figure 11

TECHNICAL STATUS AND ESTIMATED COST

- Reactor technology based on proven NERVA Technology from early 1970's

Therefore:

- Principal technology uncertainties and primary development costs associated with power conversion and integration of power and propulsion modes

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Figure 12

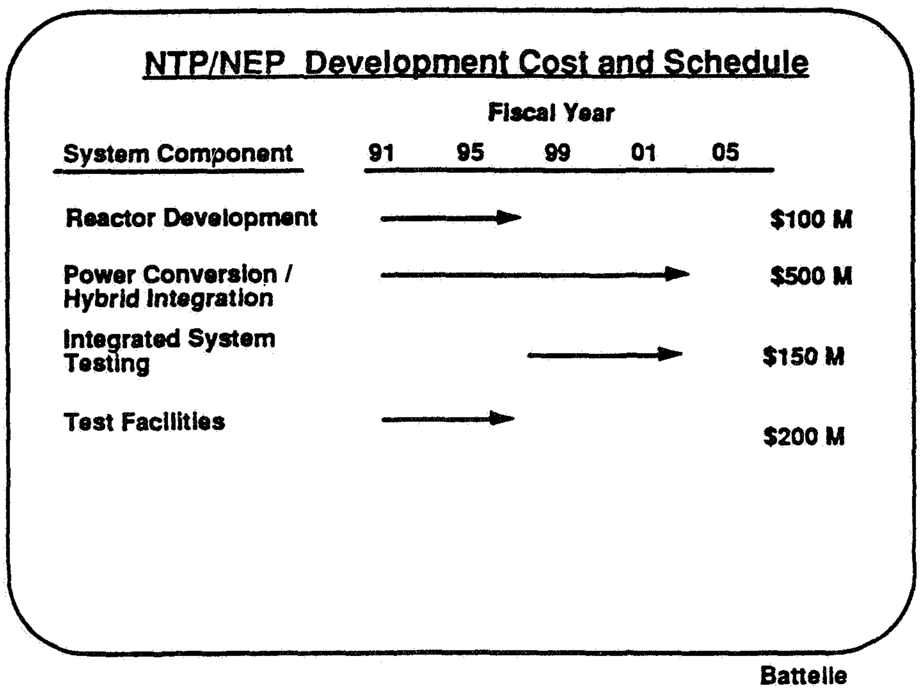

Figure 13

Conclusions ON NTP/NEP HYBRID

Mission Benefits

- Reduced trip time compared to NTP options with only propulsion mode
- Reduced IMLEO compared to NTP options offering similar trip times
- Offers redundant means of propulsion with enhanced mission flexibility

Technical and Developmental Risk

- Reactor technology based on proven NERVA technology from early 1970's

Safety

- Safety issues primarily associated with NERVA Technology
- Minimum additional risk compared to NTP alone
- Additional complexity does result in some increased safety / reliability risk

Battelle

Figure 14

OPEN CYCLE GAS CORE NUCLEAR ROCKETS

Robert Ragsdale
Sverdrup Technology, Inc.

I am going to walk you through some prior Lewis Research Center work. The work started about 32 years ago, and ended about 17 years ago, so this is not something that was done yesterday. However, that's true of everything we are talking about. It was part of a supporting research and technology program funded out of SNPO, the Space Nuclear Propulsion Office. It was not a development program like NERVA. Work was done at Lewis on the open cycle Gas Core concept, and parallel work was also funded by NERVA money, done at United Aircraft Research Laboratory. That work will be described in the following paper by Tom Latham.

The basic concept is shown in Figure 1. The open cycle gas core engine is a nuclear propulsion device. Propulsion is provided by hot hydrogen which is heated directly by thermal radiation from the nuclear fuel. This is the entire engine. Critical mass is sustained in the uranium plasma in the center. It has typically 30-50 kilograms of fuel. It's a thermal reactor in the sense that fissions are caused by absorption of thermal neutrons. The fast neutrons go out to an external moderator/reflector material and, by collision, slow down to thermal energy levels, and then come back in and cause fissions.

The hydrogen propellant is stored in a tank. It runs through a turbo pump system, regeneratively removes all of the gamma and neutron heating in the moderator/reflector region, cools the nozzle and then flows into a cavity.

There is a direct contact in this open-cycle concept between the uranium and the hydrogen. The transfer of heat is primarily by a photon wave, a thermal flux that radiates outward. It is intercepted by the hydrogen propellant that comes in through the wall so that it's optically black in there. The wall doesn't see this very high-temperature incandescent nuclear plasma. The heat is intercepted by the hydrogen at low temperatures by adding what is called seed material. Seed material is very small dust particles which could be carbon, tungsten, or U235 itself. The hydrogen becomes optically opaque and begins to absorb the radiation. The hydrogen flow path is generated through the wall at controlled angles. The hydrogen depends on the flow field setting up essentially a stagnation pressure point which causes a slow recirculation dead zone in the center. That's why the uranium "sits" there instead of escaping. A lot of work was done on the fluid dynamics of how you set up a flow pattern to cause that to happen.

The advantage of the concept is very high specific impulse because you can, in principle, take the plasma to any temperature you want to by increasing the fission level by withdrawing or turning control rods or control drums.

The model you might picture is very much like a small contained "sun," radiating its heat outward, with very high temperatures in the center of the plasma. Temperature is fairly constant near the center, but drops near the edges and then drops through the hydrogen. The heating process is fission, not fusion, so it is not really like a sun. It is an optically-thick, radiating, incandescent heat source. In principle, you can reach any specific impulse that the hydrogen can attain without burning out the nozzle or the wall.

The nuclear issues are: containment of the plasma, the nuclear criticality effects, the power levels, and the control system. The hydrodynamics are mainly related to flow. The heat transfer concerns the seeding of the hydrogen and the protection of the wall and the nozzle.

The work was following a step-wise path which involved neutronics, fluid dynamics and heat transfer (Figure 2). We had not progressed to the point of moving beyond the neutronics, fluid dynamics and heat transfer, and beginning to couple those things together. The work stopped in 1973. There were cold critical experiments, fluid dynamics experiments, and heat transfer experiments done. The heat transfer experiments concentrated on the optical properties of the gases themselves when they were ionized. In 1973, cold flow experiments were beginning which combined the understanding of cold flow and nuclear issues into a cold flow critical experiment. There were also hot flow experiments that combined RF heating with the cold flow. The next step would have been bringing together all three of those in small-scale fission experiments, and then a full-scale test equivalent to running a NERVA engine out in Jackass Flats. There was, right near the end of the program, a PER (Preliminary Engineering Report Study), which began to look at how you would really test one of these things on the ground when you got to where you knew how to build one.

The work was done primarily at Lewis Research Center, but it was supported by a large number of relatively small research grants and contracts. Figure 3 shows who the actors were, what they were doing in the areas of criticality, radiative heat transfer, nuclear fuel containment and systems studies. The A designates analytical work, the E is for experimental work.

Figure 4 shows a list of new technologies that could be used in the development of the open cycle engine. The flow was one of the big problems experimentally. We were always out in front of the analytical techniques that could be used to analyze those kinds of experiments. CFD could help a lot by modeling some of the old cold flow data and hot flow data. Maybe we could model the entire engine concept itself. CFD techniques have advanced a lot since 1973. Also, structural ceramics didn't even exist then. Space radiators and heat pipes have also advanced a lot since the days this work was done by Lewis Research Center. Non-intrusive instrument technology would be very valuable in the flow experiments.

The key questions haven't changed at all in 15 or 20 years (Figure 5). Can containment be achieved? That's a tricky question. Primarily it comes down to an acceptable fuel loss rate. And there are a lot of semantics in what is acceptable -- cost, public perception, using up a natural resource, safety and radiation. And the amount of fuel that is in the reactor, of course, determines the pressure level. The next question is could a 5,000 second Isp nozzle be cooled even if you could heat the hydrogen? This is a key problem we really didn't address. Finally, can it be ground tested within today's constraints and at an acceptable cost and risk? It is easy to write the questions down, but difficult to answer them. However, it is not difficult to envision ways to get at the answers.

Base line engine performance for a 5,000 megawatt reactor is shown in Figure 6; 5,200 seconds specific impulse, 50,000 pound thrust, engine weight 250,000 pounds. The entire engine was contained within about a 14-foot pressure vessel. Nozzle area ratio was about 50 to 1, but it could be whatever you chose.

Man rating features required nothing special in this engine other than the usual turbo pump duality and things like that. The engine weight included the pressure shell, moderator and reflector (which actually constituted the shielding). The gamma rays are all trapped by the large mass around the fuel.

The mission/systems status of the work that was done by Lewis as of about 1973 showed the potential for a 60- to 80-day round trip mission, which did not deliver a payload ("courier" mission), to Mars (Figure 7). Performance was unmatched, unsurprisingly, by NERVA and nuclear electric, but somewhat surprisingly even by fusion, because of the way Lewis modeled fusion on that very first trip.

An engineering design study of an engine in about 1972 disclosed areas for potential improvement primarily in terms of what the fuel would be. It is not necessarily conclusive that you use Uranium 235; you might use 233. Other improvements could affect moderator/reflector material, the liner itself, the inside liner that the hydrogen flows through, and finally the space radiator. A first cut through that preliminary engineering report study disclosed no real fundamental reasons that you can't test a gas core reactor.

Engine/mission characteristics are show in Figure 8. It's for the Mars "courier" mission. Specific impulse is shown as a function of engine thrust and this is total engine weight.

The nominal engine picked for the mission was 50,000 pound thrust. Isp is reduced because you have to cool the nozzle transpirationally and that led to the reference engine. It took about a 100,000 pound command module and left a 600 kilometer Earth orbit. It parked into an eccentric Mars orbit with about 1.1 Mars-radius, came back, and reparked into the same 600 kilometer Earth orbit. Figure 8 is for that mission. It shows trip time versus the initial mass in Earth orbit in kilograms, from zero to two million. Trip time follows the kind of curve that you would expect. You can cut initial mass in

Earth orbit in half by going out 80 days from 60.

Figure 9 shows what ought to be done. First year activities would be to setup the CFD models, looking at both thermal Isp limits, and the containment and flow process. Also, you would reestablish an engine system model to give you a crack at the trade-off studies between weight, pressure, and critical mass as a function of Isp level. Then, you should go back and update that 1972 facility study.

In the near term (maybe the first year or two), one critical experiment would be to reestablish a benchmark cold flow test (Figure 10). I would urge moving into a five to ten megawatt RF heated Isp nozzle test.

A one-megawatt RF flow containment test would also be valuable. Technology to do that was already demonstrated at the end of the program. Finally, a spherical ZPR (zero power reactor) test using flow within the cavity should be run.

For long-term critical testing, you would have to do a flowing critical test to show containment and reactivity control (Figure 11). First cold, then warm, then hot. Also, you should perform a low-power engine test (a reactor test) to show the Isp and effluent handling capability. Finally, run a full power prototype engine test.

What would all that cost? Figure 12 shows a guess. Start up studies at first should retrieve the original data. Come up with a preliminary program plan and then a final program plan. Then, in what is perhaps an optimistically short time period, technology development starts where the program ended before and moves fairly fast to a point I call "technology readiness." Technology readiness means you don't need any more new technology, any more research in creating new knowledge. What you do need is a lot of engineering to develop the system.

The big bucks are spent on engine development. This items includes an early cut at redoing the PER on the facility, then do an official PER as a part of the usual NASA Construction of Facilities procedure, begin ground testing and finally reach some point at which the engine is ground-qualified. At this point, it is as qualified as you can do in a one-g environment. Because you are at such high pressure levels, you really don't need to exhaust into a vacuum. The biggest factor here may be the presence of a one-g field in an engine that would be operating more like 0.1 to 0.01g. You can't simulate that, so then there is space development required.

Figure 13 breaks out the costs as to how much you would spend to get to key decision points.

Assuming you start this fall, by next spring with a fairly small expenditure you can gather the old data base and take a better look at what I am discussing here. If this is a serious contender at the research level then initiate the focused technology program development around the spring of 1992. By that point would you have spent a million dollars.

Spring of 1993 would bring you up to a "go, no go" decision based on the facility PER and the high Isp nozzle cooling test. You spend five million bucks or so to get to that point.

Then you step up the expenditure level around 1997. You would be halfway through ground development and at that point you would have enough information to make some kind of a decision.

At about this point, you reach the end of the technology readiness plan where you don't intend to develop any more technology. Only engineering remains. Around 2000, 2002 you would decide how well the solid core is going, how well does the Gas-Core Nuclear Rocket look like it is being developed?

Figure 14 shows a test facility. I have assumed all the way through this that you wouldn't be working on gas cores unless you are working on solid cores, so the basic facility would be there. You would need to add a very large scrubber (which in today's environment you need for Solid-Core reactor anyway) to the engine test.

What are the risks? There are two kinds of risks: One is programmatic-which may be tough to deal with (Figure 15). First of all, the time to technology readiness. It takes a lot of guts to buy into this program and invest the money you need to before you know if you are going all the way. I think that's a risk. Other risks include public reactions to fuel release in space, and getting ground testing approval. I think those are very significant program risks, especially for this kind of concept.

Then there are technical risks. I put these in my opinion of order of priority. These risks include nozzle cooling, nonnuclear simulation reliability, fuel containment and reactor dynamics, and ground testing and scrubbers.

After 15 years of research at Lewis Research Center I think the idea remains very attractive (Figure 16). Obviously the idea of high-speed, low trip-times to Mars is very attractive. The key questions for the open cycle are: Can containment be achieved, can it be ground tested and can a rocket nozzle handle 5,000 seconds Isp flows? Finally, I think restarting the effort with a reasonable efficiency would be a fun challenge for somebody.

A VOICE: Why does it weigh so much and what ideas do you have to bring the weight down?

MR. RAGSDALE: It weighs so much because of the radiator, moderator, and reflector material around it and the pressure vessel that has to contain it. The way you can bring it down would be devise ways that don't require as much mass. I don't really think you are ever going to bring the weight down very much. You just need that much moderator material around it to thermalize the neutrons.

A VOICE: What is the moderator material?

MR. RAGSDALE: Beryllium, beryllium oxide, heavy water, possibly graphite.

A VOICE: Perhaps zirc hydride on that nozzle could be more effective?

MR. RAGSDALE: Possibly, but I think you are not going to significantly affect the weight. These things are just big and heavy.

A VOICE: Half of your engine weight was radiator.

MR. RAGSDALE: Right. And today's technology may provide lighter radiators. I am not sure that the weight is a fantastic problem. The real question in my mind is not bringing the weight down but does it really work and produce 5,000 seconds of impulse. If it does, you are going to get to Mars and back in 60 or 80 days. Even if you made the weight zero, you are not going to do a lot better than that.

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<u>Open Cycle Gas Core</u>

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OPEN CYCLE GAS CORE ENGINE - THE CONCEPT

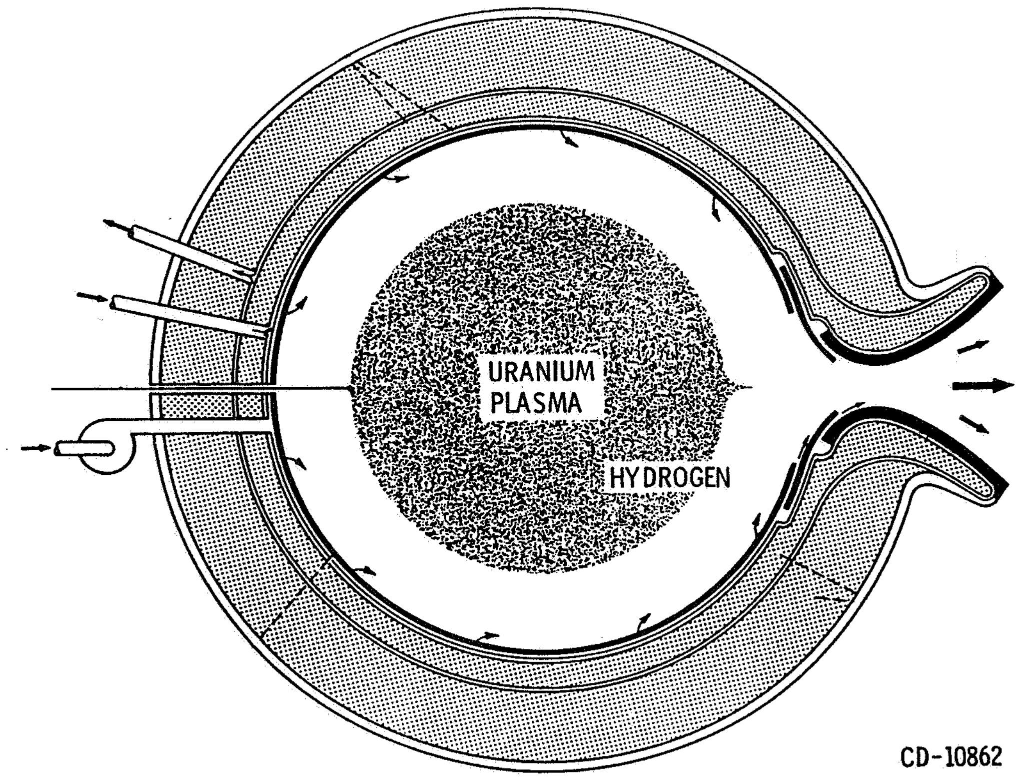


Figure 1

THE WORK WAS FOLLOWING A STEPWISE PATH

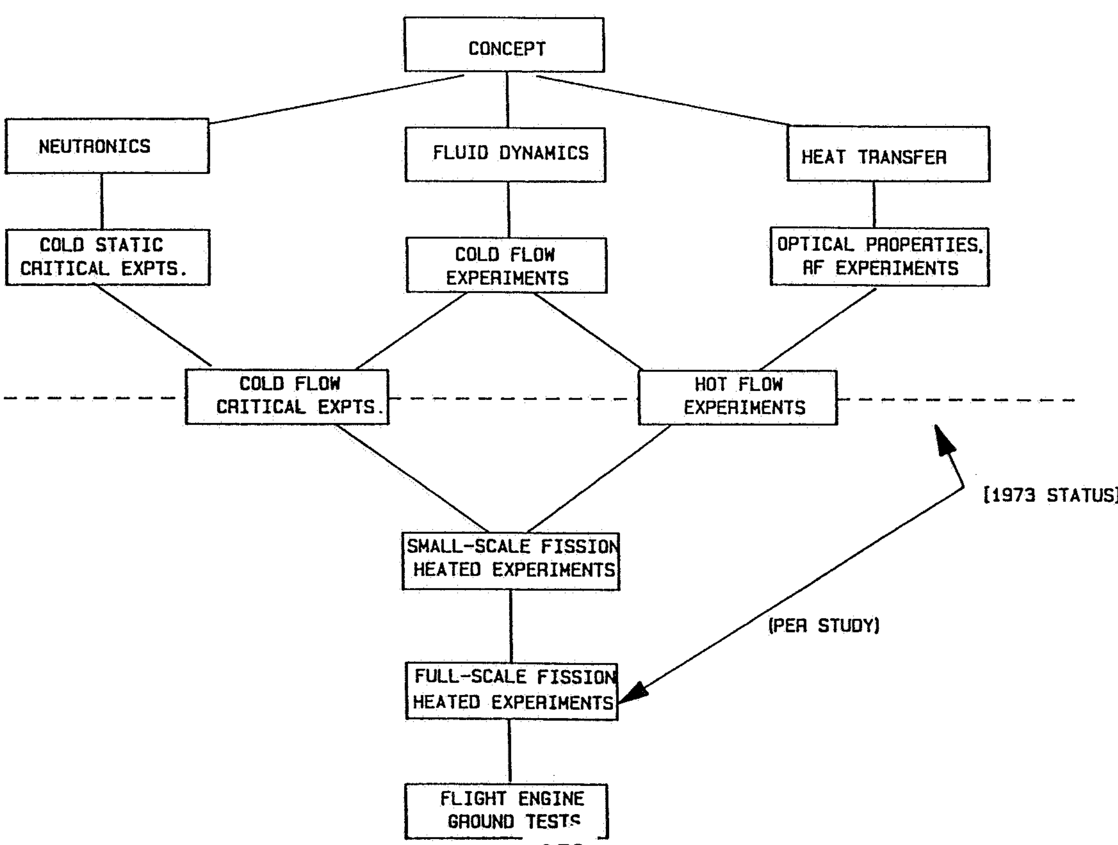


Figure 2

WHO DID WHAT

Criticality of Gaseous Fuel	Radiative Heat Transfer	Nuclear Fuel Confinement	Systems Study
NASA-Lewis (A)	NASA-Lewis (A/E)	NASA-Lewis (A)	NASA-Lewis (A)
Aerojet Nuclear (E)	TAFA Division Humphreys Corp. (E)	Illinois Institute of Technology (A/E)	United Aircraft Research Labs (A)
United Aircraft Research Labs (A)	Georgia Institute of Technology (E)	Cornell University (A/E)	Georgia Institute of Technology (A)
Versar, Inc. (A)	AEDC (E)	United Aircraft Research Labs (A/E)	University of Florida (A)
	United Aircraft Research Labs (A/E)	Aerojet Nuclear (E)	Computer & Applied Sciences, Inc. (A)
	University of Florida (E)	University of Arizona (E)	
	University of Maryland (E)	NASA-Ames (E)	
	NASA-Langley (E)		

Figure 3

POTENTIAL NEW TECHNOLOGY IMPACTS

o COMPUTATIONAL FLUID DYNAMICS
  - MODEL OLD COLD FLOW EXPERIMENTAL DATA
  - MODEL OLD RF HOT FLOW DATA (?)
  - MODEL THE ENGINE CONCEPT FLOW

o STRUCTURAL CERAMICS

o SPACE RADIATORS/HEAT PIPES

o NON-INTRUSIVE INSTRUMENTATION
  - COLD FLOW
  - HOT FLOW

Figure 4

KEY QUESTIONS

* CAN CONTAINMENT BE ACHIEVED ?
  - ACCEPTABLE FUEL LOSS RATE
  - ACCEPTABLE REACTOR PRESSURE
* CAN A 5000 SEC Isp NOZZLE BE COOLED ?
* CAN IT BE GROUND TESTED ?
  - WITHIN TODAY'S CONSTRAINTS
  - ACCEPTABLE COST/RISK

Figure 5

BASELINE ENGINE PERFORMANCE DATA

SPECIFIC IMPULSE	5200 SEC
THRUST	50,000 LB
ENGINE WEIGHT	250,000 LB
ENGINE DIAMETER (sphere)	14 FT
NOZZLE AREA RATIO	50:1

MAN RATING FEATURES - nothing engine-unique is included - usual pump duality, etc, is assumed

* includes pressure shell, moderator, reflector, shielding, space radiator

Figure 6

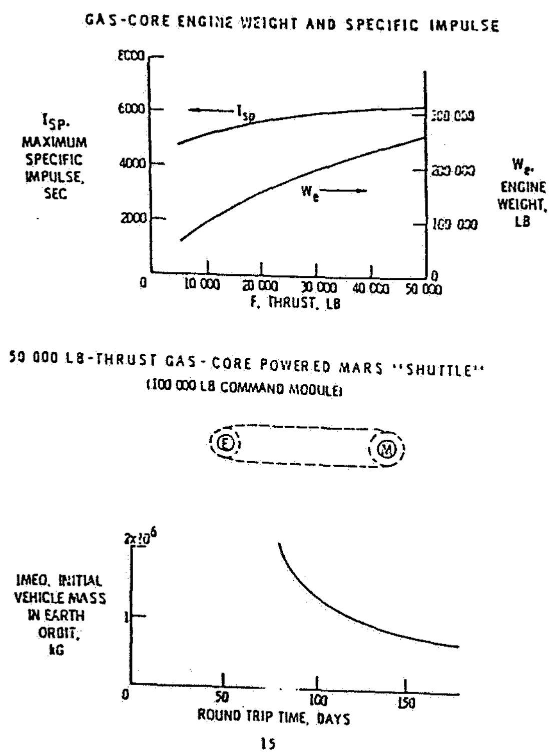
MISSION/SYSTEMS STATUS

o MISSION STUDIES SHOWED POTENTIAL FOR 60-80 DAY ROUNDTRIP COURIER MISSIONS TO MARS - PERFORMANCE UNMATCHED BY NERVA, NUCLEAR-ELECTRIC, OR FUSION SYSTEMS

o AN ENGINEERING DESIGN STUDY OF AN ENGINE DISCLOSED AREAS OF POTENTIAL IMPROVEMENT - FUEL, MODERATOR, LINER, RADIATOR

o A FIRST CUT AT A GROUND TEST FACILITY (A "PER" STUDY) BY LeRC DISCLOSED NO INSURMOUNTABLE ISSUES, BUT LEFT MUCH TO BE DONE

Figure 7



ENGINE/MISSION CHARACTERISTICS

Figure 8

KEY FIRST YEAR ACTIVITIES

* SET UP CFD MODELS
  - THERMAL Isp LIMITS
  - CONTAINMENT

* ESTABLISH ENGINE SYSTEM MODEL
  - WEIGHT, PRESSURE, CRITICAL MASS, Isp

* UPDATE 1972 FACILITY PER STUDY

Figure 9

CRITICAL TESTS - NEAR TERM

* BENCHMARK COLD FLOW TEST

* 5 - 10 MW Isp NOZZLE TEST

* 1 MW RF HOT FLOW CONTAINMENT TEST

* SPHERICAL ZPR TEST OF CFD FUEL DISTRIBUTION

Figure 10

CRITICAL TESTS - LONG TERM

* FLOWING CRITICAL REACTOR TEST TO SHOW CONTAINMENT AND REACTIVITY CONTROL
  - COLD, THEN HOT

* LOW POWER ENGINE TEST TO SHOW Isp and EFFLUENT HANDLING

* FULL POWER PROTOTYPE ENGINE GROUND TEST

Figure 11

TOP LEVEL SCHEDULE AND COSTS

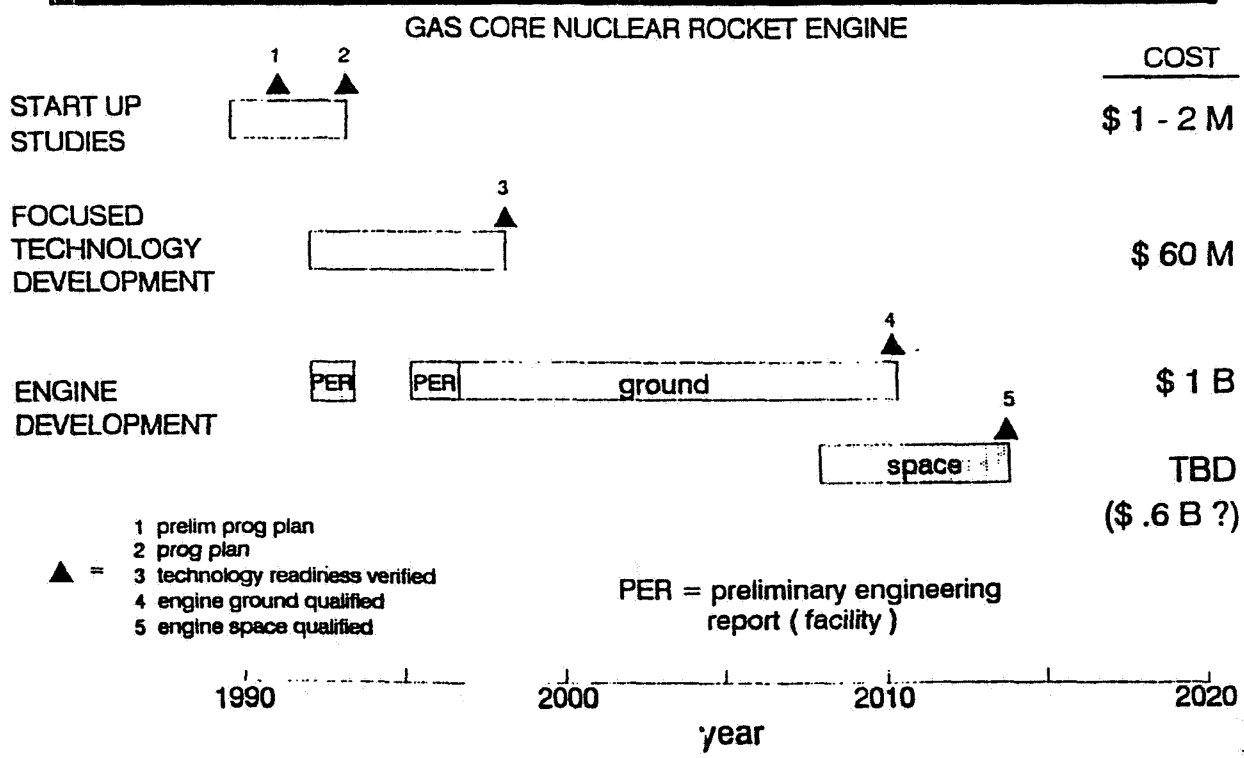


Figure 12

MAJOR DECISION POINTS

DATE	DECISION	COST TO DECISION
APR 1991	RETAIN/DROP GNR OPTION	$ 200 K
SPRING, 1992	INITIATE FOCUSED TECHNOLOGY DEVELOPMENT	$ 1 M
SPRING, 1993	GO-NO-GO DECISION BASED ON FACILITY PER & NOZZLE Isp TESTS	$ 5 M
1997	INITIATE ENGINE DEVELOPMENT BASED ON TECHNOLOGY READINESS ASSESSMENT	$ 75 M
2002	SOLID CORE/GNR DECISION	$ 700 M
2008	INITIATE SPACE QUALIFICATION	$ 1 B

Figure 13

ENGINE TEST FACILITY

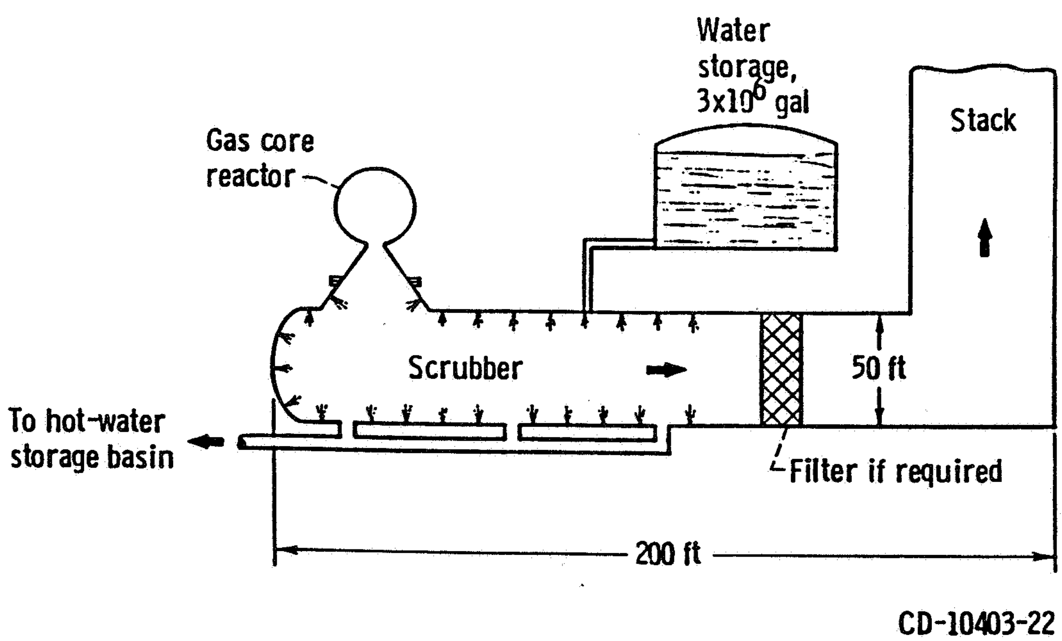


Figure 16. - Addition to test stand ETS-1 or test cell C at Nuclear Rocket Development Station required for 10 000-megawatt gas core test facility. Hydrogen and services supplied by existing systems.

Figure 14

DEVELOPMENT RISKS

PROGRAMATTIC

* TIME TO TECHNOLOGY READINESS DEMO
* REACTIONS TO FUEL RELEASE IN SPACE
* GROUND TESTING APPROVALS

TECHNICAL

* NOZZLE COOLING
* NON-NUCLEAR SIMULATION RELIABILITY
* FUEL CONTAINMENT/REACTOR DYNAMICS
* GROUND TESTING - SCRUBBERS

Figure 15

CLOSING REMARKS

- AFTER 15 YEARS OF RESEARCH, THE IDEA REMAINS VERY ATTRACTIVE, BUT HIGH RISK

- THE KEY QUESTIONS FOR THE OPEN CYCLE GNR ARE :

  - CAN CONTAINMENT BE ACHEIVED ?
  - CAN IT BE GROUND TESTED ?
  - CAN A ROCKET NOZZLE HANDLE HIGH PRESSURE, HIGH Isp FLOWS ?

- RESTARTING THE EFFORT WITH REASONABLE EFFICIENCY WOULD BE AN EXCITING CHALLENGE !

Figure 16

VAPOR CORE PROPULSION REACTORS

Nils J. Diaz
University of Florida
Innovative Nuclear Space & Propulsion Institute

Figure 1 describes INSPI's charter, participating institutions and projects. Essentially we have been in existence for five years and we have been funded to work on advanced nuclear space power reactors, including gas cores. A significant amount of work has been done recently.

Earlier research used small amounts of UF6, flowing it in argon. We did critical experiments at 20 kilowatts. One time we went to a hundred kilowatts for a few seconds, got there and came down. Most of the fuel was in the cavity, about 4 kilograms of uranium hexafluoride. We still had solid fuel elements, but we did have a critical system with a significant amount of UF-6 in it. A lot of testing was done on it -- both steady state and dynamic testing. We looked at the characteristics of the system and we validated many of our codes.

Initially, we were having as many as four groups doing different gas core analysis, and eventually a decision was made to focus research on just one concept.

Many research issues were addressed (Figure 2). For example, it became obvious that uranium tetrafluoride is a most preferred fuel over uranium hexafluoride. Every time we start with uranium hexafluoride and go to even lower temperatures (700 K), we end up with uranium tetrafluoride. So why fight mother nature? UF-4 doesn't have the problems UF-6 has, and it has a very attractive vaporization point; 1 atmosphere at 1800 degrees Kelvin. So it is a temperature that's not enormously high, yet hot enough.

We also looked at materials compatible with uranium tetrafluoride, like tungsten, molybdenum, rhenium, carbon. We find that in the molten state, UF-4 and uranium attacked most everything, but in the vapor state they are not that bad. A lot of materials contained them in the vapor state. We identified compatible materials for both the liquid and vapor states.

We actually established a series of analyses to determine how the cavity should be designed. For example, unless your central fuel region is as large as 0.85 of the cavity, the system could be inherently unstable.

We did a series of experiments to determine the properties of the fluid, including enhancements of the electrical conductivity of the system. We now have CFD's and experimental programs that deal with most of the major issues.

We also said that if we do not do nuclear testing from the beginning, eventually we would going to lose our credibility. We performed some small, controlled nuclear testing that gives the gas core credibility.

The beauty of the gas core is that nuclear experiments are easily done. They are rapidly assembled. That is one of their advantages. You can make a gas core with an aluminum tank and a barrel of UF-6. We can design and build and put a gas core in operation in three months.

The only issue is safeguards. Do you have access to the kilograms of uranium that you need?

When working on solid cores, the problem comes down to the fuel. And it will always be like that because as long as you have solid fuel, you are limited (Figure 3).

We nuclear engineers should be using radiation. We stopped using radiation to use heat. That's the way we were trained, that's what we learned. We can use radiation because the vapor cores have an unlimited fuel temperature (Figure 4). They have inherent hot spot equalizers, tremendously high burnup. Limitations are established by the wall cooling, not by the temperature of the heat you can transfer.

The things that are no longer available to you with the solid cores, become the heart and the furnace of your system. You can do all of the things better with vapor cores.

NERVA, which is the standard, is a solid fuel reactor. It might be the only one that we will be able to put out there. My point with this technology is that regardless of how we go, the high temperature technology needs to be developed. It should be done in steps, not jumping to an enormously high temperatures, because temperature costs money and takes time.

What would be the best gas core reactor? Vapor core reactor (Figure 5). Well, it's very simple. The best gas core reactor is one in which the fuel is as hot as you can get it and the fuel is separated from the propellant. The fuel is confined, the propellant goes out and that would be great. So that dream of these two wonderful substances, one fissioning, depositing the energy, and the other coming out -- they cannot coexist unless you want to have really significant separation.

The second best gas core will be one in which you have intimate mixing of the fuel and the propellant, and then you can separate it by some means that might include more than just a vortex flow. You might include some mechanical means of separation.

I am going to initiate the technical part of my talk by looking at what can be done with gas cores and what they are.

We have heard of gas core concepts in which the temperature of the fuel is very, very high, and the mode of heat transfer is preferably by radiation heat transfer. So, you need extremely high heat transfer rates. Then you need a series of complicated yet potentially achievable containment techniques. But, because it is radiative heat transfer, you have to work at very, very high temperatures, and that creates a significant problem as far as how soon we can get this.

There is a second region in which the fuel and the propellant are mixed. The mixture uses some of the best things of the gas cores; the intimate contact, direct molecular collision using direct fission deposition and everything else. Here you have to separate the uranium because the cost of the uranium would be prohibitive, if not economically, then politically.

At less than 5000 degrees Kelvin, there is a region that is of enormous interest to the development of the gas core. That is a region in which the gas core is completely separated from the propellant by a physical wall. The minute you do that you reduce the potential of the gas core. You are now almost a solid core, but not quite. What this allows you to do is to get rid of the limitations of solid cores (Figure 6).

Here you start with a vapor fuel. It has the capacity to occupy different geometrical shapes. The vapor core is a better fuel than standard fuel elements because temperature is no longer limited. Heat transfer is limited by conductance only.

The minute you use a physical barrier, you are severely limiting the heat transfer area. We can use all of the energy transfer mechanisms: direct molecular conduction, fission fragment energy deposition, molecular collision, and radiative heat transfer. Again, the area and the mode of heat transfer are very, very important considerations.

The fundamental features of the vapor reactor (Figure 7) include the fact that energy conversion is not limited by fuel temperature, but rather by wall cooling. The core fission power density is not limited by fuel thermal-mechanical or thermal-hydraulic considerations. There are no geometric constraints on the fuel configuration. If we want to trap it a little bit inside the wall, there are no limitations on the lifetime of the reactor due to the fuel. It also has a much higher burn up because you can burn not only the outside of a fuel element, but you can actually burn the entire load. The gas core has the capability of doing direct ionization, so we can improve the optical and electrical properties of the gas.

We have talked about the advantages of the design features (Figure 8); high fuel utilization, no fuel fabrication, simplified fuel management. There is also inherent hot spot compensation, which means that if you have a hot spot, as fission increases, it gets hot, and it moves into another region. It's a very interesting effect. Density decreases with temperature, which will decrease power in that region. Moving on then, fission product removal is possible.

There is inherent stability in the fuel. The fuel core geometry constraints are minimized. Fuel density can be varied and you can have different power densities in different regions by separation.

Disadvantages include confinement, containment, and recirculation of fuel. Fuel recirculation loops create new problems. One of the main reasons for having a solid fuel is to keep the radioactivity in the fuel. With circulated fuel we have an added problem.

Figure 9 is going to be very familiar to you all. We just took the NERVA core and decided we were going to do a few things to it. What we did is create a cell with a hydrogen core. We put in a graphite wall made of carbon-carbon. Reactivity dropped effectively from 1.4 to 1.07. It really took a beating, but it's still critical. So the cell is arranged with hydrogen, carbon-carbon wall, moderation, uranium tetrafluoride and helium. UF-4 is a poor heat conductor. We added helium to improve heat transfer.

This is a very simple design. Instead of the fuel being dispersed through the matrix, it is now a vapor. But it basically uses all of the NERVA technology.

We do have some significant changes. We put a beryllium reflector on the top because we needed reactivity. We put a graphite reflector on the bottom. This does make the system heavy. Now we could probably do away with those two things if we put 25 centimeters of NERVA fuel in here at the very top. In our calculations, the system K effective increased to about 1.7, and we generated almost two-thirds of the power in here. Why would you do that? Because the solid core has that high power density.

To summarize, Figure 10 shows baseline system parameters for two systems, one is the NVR, which we could call a super NERVA. The other is a generic Vortex Confined Vapor Reactor (VCVR), which is sketched out in Figure 11 (and was discussed by S. Anghaie).

As far as technology readiness is concerned, practically everything is a 2 or 3, except for fuel confinement with internal heat generation (Figure 12). That's not been done at all. We have now capabilities to do research both with very high temperature and nuclear and nonnuclear testing in lab prototypes. Figure 13 and 14 list tasks that should be preformed to rapidly come up to a level in which we can determine what the options are.

Figure 15 lists critical test requirements and safety issues. Many things have been done in the previous five years that actually impact the cost and schedule. We have the fluid dynamics, the high temperature cross sections, and the capabilities of doing experiments at very high temperatures (up to 10000 degrees Kelvin). We also have the facilities.

I think it's critical to get this concept going.

Figure 16 shows research and development costs. It's not a very expensive thing. However, the numbers do not include facilities, so whenever you have to use a facility, you have to add the cost in. I don't know what the facility costs are. They keep changing all the time. So you could easily add \$100 million to this for facilities, and once you start doing your prototype you might have to add \$200 million to it.

I terminate this at ten years. Beyond this point the gas core and solid core cost the same. The reason is that in the first ten years we don't have to have fuel fabricated, tested and qualified. We can have fuel tomorrow.

UF-4 is a nice substance. We can do things more quickly and more economically than anybody else because we have the fuel in the form that we want it. We don't have to do anything to it; we don't have to test it; we don't have to verify it. That provides an enormous saving in time and money.

Note: For Bibliography, See DCNR (S. Anghaie)

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INSPI WAS CHARTERED AND SPONSORED BY SDIO/IST, BEGINNING IN SEPTEMBER, 1985. IT HAS BEEN OPERATING AS A NATIONAL CONSORTIUM OF UNIVERSITIES AND HIGH TECHNOLOGY BUSINESSES PERFORMING MULTI-DISCIPLINARY, MULTI-INSTITUTIONAL RESEARCH ON ADVANCED AND INNOVATIVE NUCLEAR SPACE POWER REACTORS AND ENERGY CONVERSION SYSTEMS. ITS GOVERNMENT CONTRACTS ARE ADMINISTERED BY WPAFB. INSPPI IS AN INSTITUTE OF THE STATE UNIVERSITY SYSTEM OF FLORIDA.

THE INSTITUTIONS AND PROJECTS FROM SEPTEMBER '85 - PRESENT INCLUDED:

MMW GAS CORE REACTORS

- California State University, Long Beach
- University of California, Los Angeles
- University of Florida, Gainesville
- GA Technologies
- Maxwell S
- J. Dorning Assoc. (Univ. of Virginia)
- Pacific Sierra Research Corp.
- AVCO
- Richard Rosa, MSU (consultant)
- LANL
- RTS, Inc.
- Space Power, Inc.
- SRI, International

INDUCTIVE COUPLING THERMIONICS (TRICE)

- Rasor Associates
- Oregon Graduate Research Center (OSU)
- Space Power, Inc.
- Thermoelectron

NUCLEO-CHEMICAL CONVERSION

- Ann Arbor Nuclear, Inc.
- University of Michigan

METAL VAPOR TURBO-ALTERNATOR

- SPACE POWER, INC.
- UNIVERSITY OF NEBRASKA

Figure 1

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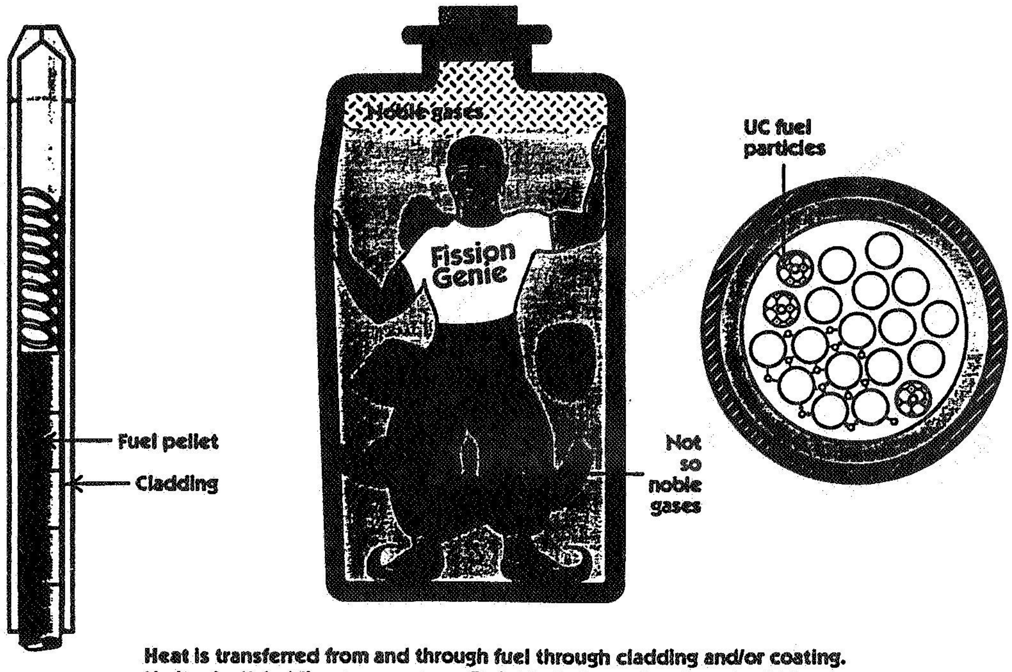
RECENT VAPOR CORE REACTOR RESEARCH

- INSPI HAS BEEN ADDRESSING THE CRITICAL SCIENCE AND TECHNOLOGY OF VAPOR (GAS) CORE REACTORS SINCE 1985 FOR SDIO.
- MANY CONCEPTS WERE EXAMINED AND A PROGRAM CENTERPIECE CONCEPT (AND ALTERNATE) SELECTED TO FOCUS THE RESEARCH IN OCTOBER 1988.
- A UF_4/KF VAPOR CORE-MHD SYSTEM IN A CLOSED RANKINE CYCLE IS THE PRIMARY CONCEPT.
- RESEARCH PROGRAM IS NOW FOCUSED ON EXPERIMENTAL VERIFICATION AND MODELING OF SCIENTIFIC FEASIBILITY AND CRITICAL TECHNOLOGY ISSUES.
- SIGNIFICANT RESEARCH ACCOMPLISHMENTS
  - UF_4 IS PREFERRED CHEMICAL FUEL FORM FOR T<5000K, U-METAL DROPLETS FOR 3000K<T<7000K, U-VAPOR FOR T>6000K
  - W, MO, RE, C AND THEIR ALLOYS & CARBIDES IDENTIFIED AS MATERIALS COMPATIBLE WITH UF_4 ABOVE 1800K
  - NEUTRONIC STABILITY OF EXTERNALLY MODERATED GAS CORE INCREASES AS FUEL DENSITY DISTRIBUTION APPROACHES CAVITY WALL
    - -- FOR CENTRALLY-PEAKED DISTRIBUTION, $V_{fuel}>0.85\ V_{core}$ FOR STABILITY
  - ENHANCED ELECTRICAL CONDUCTIVITY & MHD ELECTRICAL PRODUCTION CAN BE ACHIEVED VIA DIRECT CHARGED PARTICLE IONIZATION
  - EXPERIMENTAL AND COMPUTATIONAL FACILITIES ESTABLISHED FOR HIGH TEMPERATURE VAPOR CORE NEUTRONICS, FLUID FLOW, HEAT TRANSFER, MHD & MATERIALS ANALYSIS.

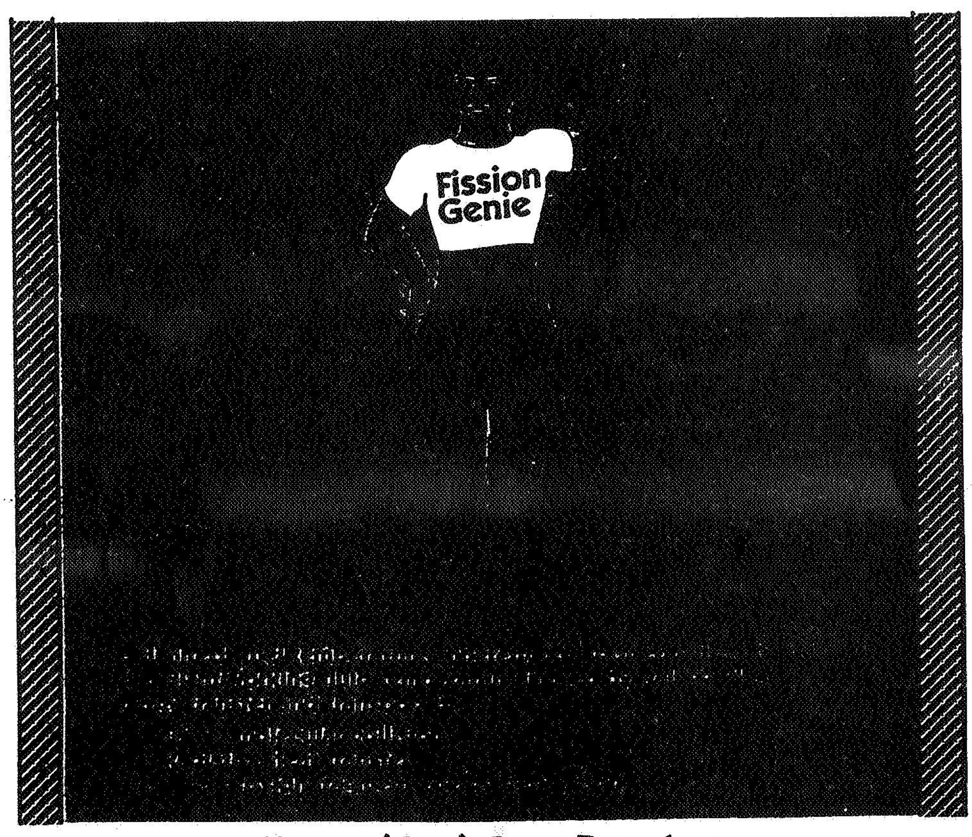
Figure 2



Heat is transferred from and through fuel through cladding and/or coating.
Limits: fuel/cladding temperature, fission/cc³, thermal and mechanical properties, burnup, peaking factors, hot spots, etc.

Solid Reactor Fuel

Figure 3



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Vapor (Gas) Core Reactors

Figure 4

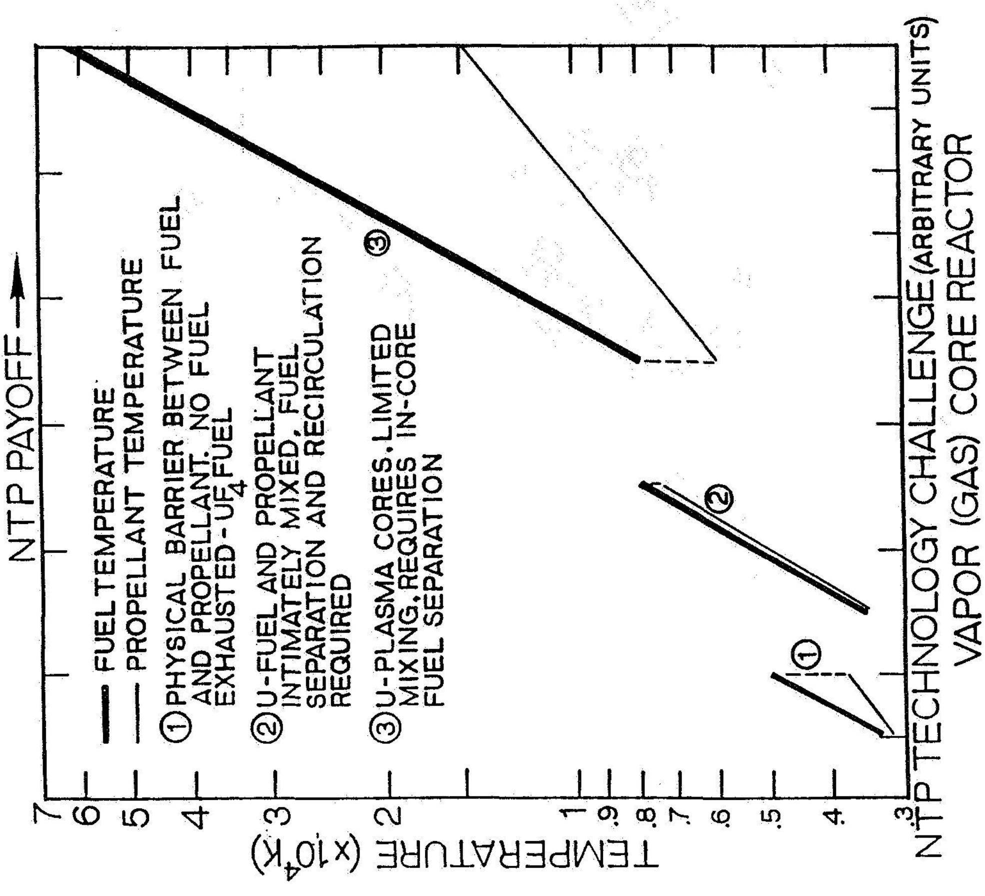
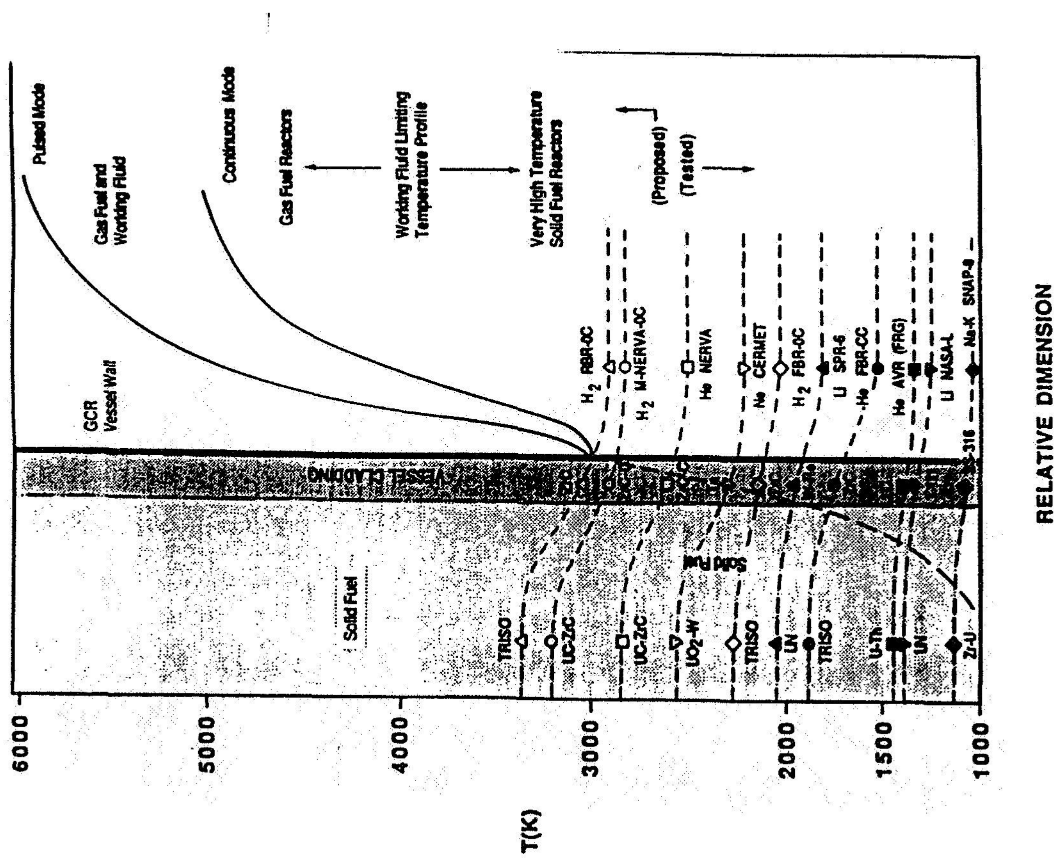


Figure 5



Limiting Temperature Profiles For Solid and Gas-Fuel Reactor

Figure 6

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VAPOR (GAS) CORE REACTORS: FUNDAMENTAL FEATURES

- THE NUCLEAR REACTOR AND THE ENERGY CONVERSION SYSTEM ARE NOT LIMITED BY FUEL TEMPERATURE BUT BY WALL-COOLING CAPABILITIES.
- CORE FISSION POWER DENSITY IS NOT LIMITED BY FUEL THERMAL MECHANICAL OR THERMAL HYDRAULIC CONSIDERATIONS.
- THERE ARE NO GEOMETRICAL CONSTRAINTS ON FUEL CONFIGURATION.
- DIRECT IONIZATION (NONEQUILIBRIUM) OF WORKING FLUID CAN IMPROVE THE OPTICAL (LASING) AND ELECTRICAL PROPERTIES OF THE FISSIONING GAS
- THERE ARE THREE ADDITIONAL ENERGY TRANSFER MODES BEYOND THOSE OF SOLID FUEL REACTORS:
  - DIRECT FISSION FRAGMENT ENERGY DEPOSITION
  - DIRECT MOLECULAR COLLISION BETWEEN FUEL AND PROPELLANT
  - RADIATIVE HEAT TRANSFER VIA BLACK BODY AND LINE RADIATION

Figure 7

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VAPOR (GAS) CORE REACTOR: DESIGN FEATURES

ADVANTAGES

- HIGH FUEL UTILIZATION (BURNUP ~ 200,000 MWD/MT)
- ELIMINATION OF FUEL FABRICATION, TESTING, VERIFICATION
- SIMPLIFIED FUEL MANAGEMENT
- INHERENT HOT SPOT COMPENSATION; DENSITY DECREASES WITH TEMPERATURE, DECREASING POWER
- FISSION PRODUCT REMOVAL POSSIBLE WITH UF_4 SLIP STREAM (POTENTIAL RADIOISOTOPE RECOVERY)
- INHERENT STABILITY DUE TO EXPANDING FUEL; POWER DISTRIBUTION CAN BE SHAPED BY DENSITY VARIATIONS (NVR)
- FUEL CORE GEOMETRICAL CONSTRAINTS MINIMIZED
- FUEL DENSITY CAN BE VARIED REGIONWISE TO FIT POWER DENSITY AND TEMPERATURE DISTRIBUTION

DISADVANTAGES

- CONFINEMENT, CONTAINMENT, RECIRCULATION OF FUEL
- FUEL RECIRCULATION LOOPS CREATE NEW AND UNIQUE PROBLEMS RANGING FROM EX-CORE CRITICALITY CONSIDERATIONS AND SHIELDING TO MULTIPLE COMPONENT MATERIAL LIMITATIONS.

Figure 8

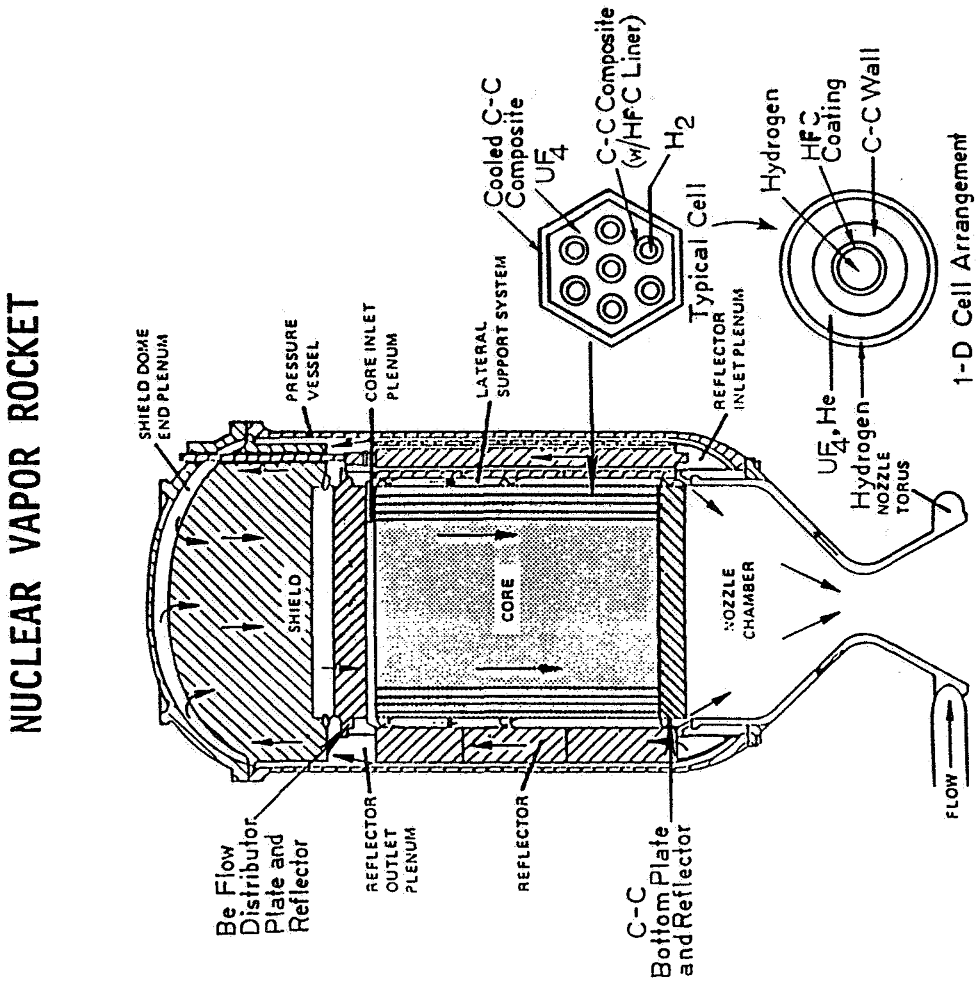

Figure 9

VAPOR CORE REACTORS

BASELINE SYSTEM DESIGN PARAMETERS

REQUIREMENT PARAMETER	UNITS:	NVR	VCVR
ENGINE AVAILABILITY	YEAR	2015	2020
THRUST PER ENGINE	KLB (F)	75	75
NUMBER OF ENGINES	NUMBER	1-7	1
REACTOR POWER (THERMAL)	MW(T)	1250	1650
DUAL MODE-LOW ELECTRIC POWER	KWE	50	25-50
DUAL MODE-HIGH ELECTRIC POWER	MWE	-	1-3
CORE WEIGHT - UNSHIELDED	KLB	33	31
CORE WEIGHT - WITH 0.75M SHIELD	KLB	55	53
ENGINE THRUST/WEIGHT	KLB(F)/KLB(M)	1-2	1-2
SPECIFIC IMPULSE	SECONDS	1280	1810
NOZZLE EXPANSION RATIO	RATIO	50:1	50:1
PROPULSION OPERATING TIME/MISSION	MINUTES	PROPELLANT-LIMITED	PROPELLANT-LIMITED
NUMBER OF MISSIONS	NUMBER	2	1
NUMBER OF STARTUP CYCLES/LIFETIME	NUMBER	12	6
AVERAGE MISSION DURATION	DAYS	310	240
RELIABILITY		?	?

Figure 10

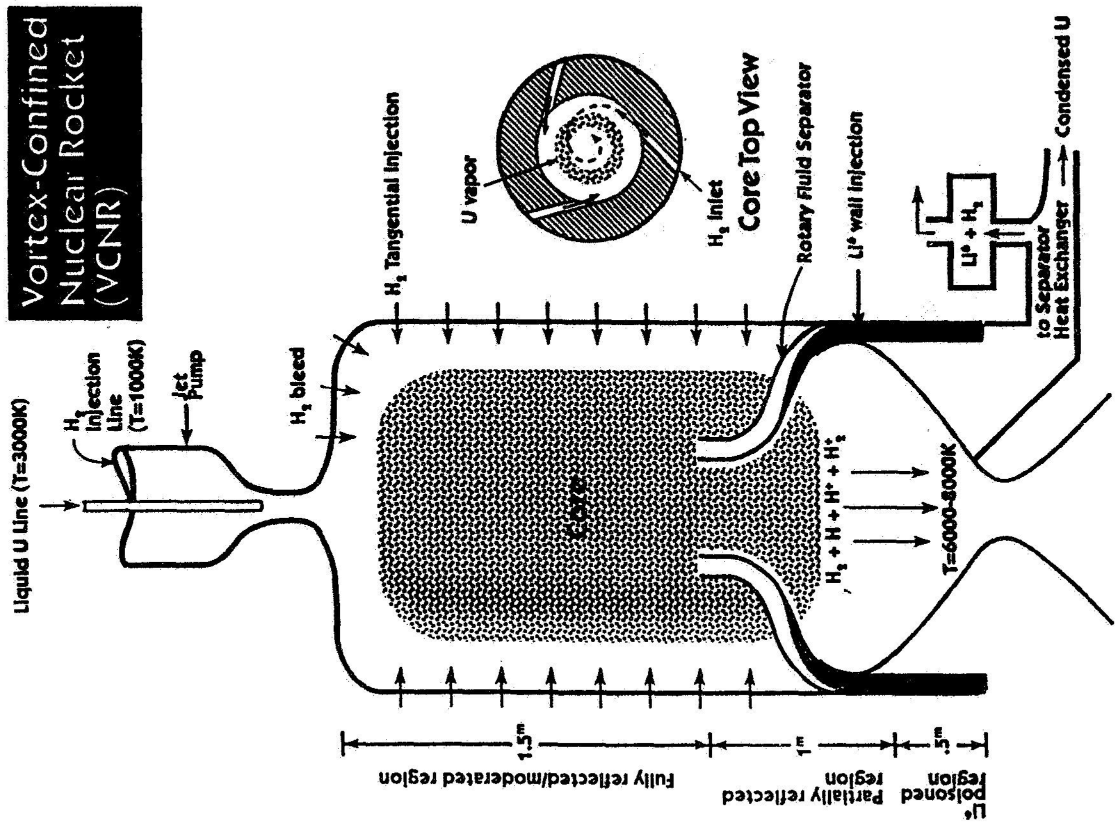

Figure 11

VAPOR CORE PROPULSION REACTORS

SCIENTIFIC/TECHNICAL ISSUE	TECHNOLOGY READINESS LEVEL NVR	VCVR
• CORE CRITICALITY & DYNAMICS	3	2
• CAVITY OR CHAMBER WALL COOLING	3	2-3
• FUEL CONFINEMENT/SEPARATION		
- WITHOUT INTERNAL HEAT GENERATION	-	2
- WITH INTERNAL HEAT GENERATION	-	1
• VAPOR FUEL		
- RECIRCULATION	2*	2
- VAPORIZATION & CONDENSATION	2*	2
• FUEL/WORKING FLUID CHARACTERIZATION	3/4	2-3
• ENERGY TRANSFER AND TRANSPORT	4	2-3
• MATERIALS COMPATIBILITY	2-3	2-3
• THERMAL MANAGEMENT	3-4	2-3

*SLIP STREAM ONLY, FOR REACTOR CONTROL & POTENTIAL RADIOISOTOPE RECOVERY

Figure 12

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REQUIRED TECHNICAL DEVELOPMENT

91-92

- UF_4 FUELED MINI-CAVITIES
  - TEST AT HIGH TEMPERATURE, NO NEUTRONS
  - TEST AT LOW TEMPERATURE, 10^8-10^{12} N/CM^2SEC
  - TEST AT FFTF, 1000 K, HI ϕ
    - τ, $\Delta\rho$ VS T, ρ
    - DESTRUCTIVE ANALYSIS

- UTREC FACILITY
  - NOZZLE TEST FACILITY UPGRADE
  - RUN WITH UF_4-CF_4-HE

Figure 13

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REQUIRED TECHNICAL DEVELOPMENT (CONT'D)

93-95

LOW POWER (~20-100 KWTH), UF_4 FUELED, FLOWING CRITICAL FACILITY

- USE BE (KIWI) AND PLASMA CORE CAVITY (PCC) AT PAJARITO SITE, LANL (SHIELD PCC, RUN @ 20-100 KWTH OR HIGHER, UF_6) @ 500K

- UPGRADE CAVITY DESIGN, MATERIALS TO T = 2500K, FLOWING UF_4

Figure 14

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CRITICAL TEST/FACILITY REQUIREMENTS, SAFETY ISSUES

CRITICAL TEST REQUIREMENTS

- THERMOPHYSICAL PROPERTIES OF UF_4-CF_4-He SYSTEM
- UF_4 HANDLING, RECIRCULATION AND FLOW
- MATERIALS INTERACTION/COMPATIBILITY
- REACTOR/REACTIVITY DYNAMICS AND STABILITY VIA FUEL DENSITY FEEEDBACK CONTROL
- ACHIEVEMENT OF REQUIRED POWER DENSITY
- CRITICALITY AT POWER, TEMPERATURE CONDITIONS; INTERNALLY MODERATED AND CAVITY REACTOR
- INTEGRAL REACTOR/COOLANT ENGINE TEST

SAFETY ISSUES

- REACTOR TRANSIENT RESPONSE
- OUT-OF-CORE CRITICALITY
- FUEL PLATEOUT (MASS TRANSPORT)
- TUBE RUPTURE ANALYSIS

Figure 15

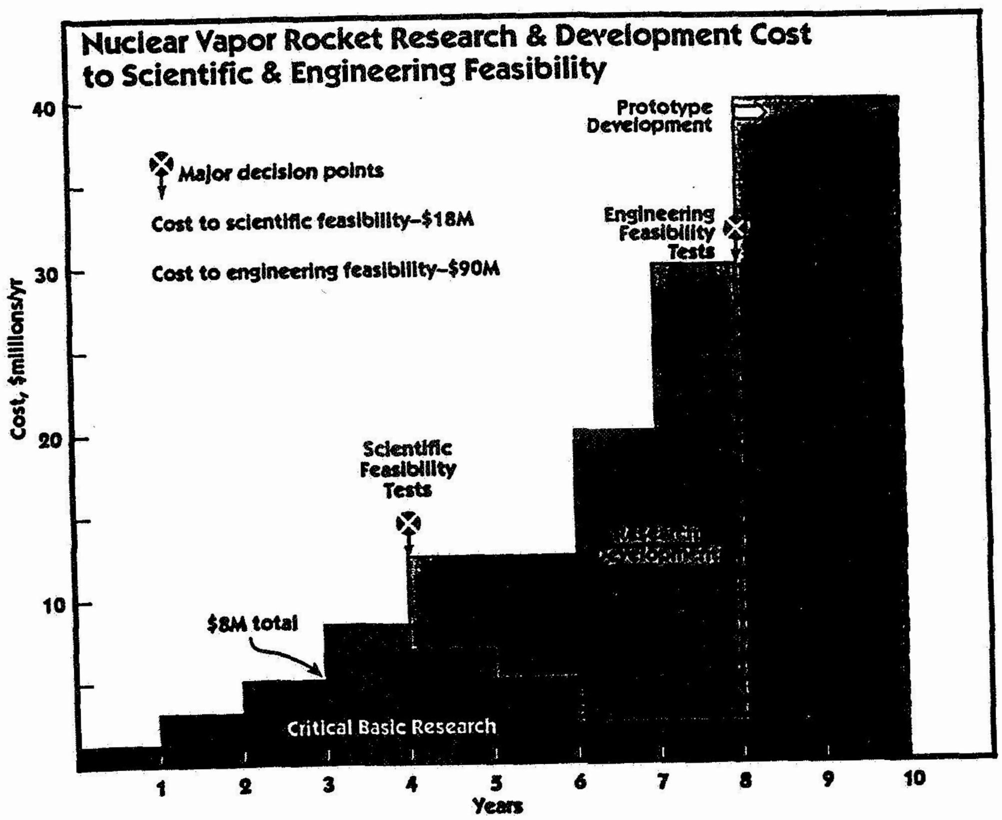

Figure 16

N92-11110

NUCLEAR LIGHT BULB

Tom Latham
United Technologies Research Center

The nuclear light bulb engine is a closed cycle gas core concept. United Technologies made a policy decision in the early days of gas core reactor development that we were not to work on any concept that didn't have the potential of complete containment of the nuclear fuel.

During that era we did support NASA-Lewis with contracted open cycle gas core flow test work and shared a great deal of technical information from the nuclear light bulb program.

The nuclear light bulb concept provides containment by keeping the nuclear fuel fluid mechanically suspended in a cylindrical geometry. Thermal heat passes through an internally cooled, fused-silica, transparent wall and heats hydrogen propellant (Figure 1). The seeded hydrogen propellant absorbs radiant energy and is expanded through a nozzle.

Internal moderation was used in the configuration which resulted in a reduced critical density requirement. This result was supported by criticality experiments. If, in addition, we used U233 nuclear fuel instead of U235, we gained about a two-thirds reduction in overall fuel loading.

A reference engine was designed that had seven cells and was sized to fit in what was then predicted to be the shuttle bay mass and volume limitations (Figure 2).

The pressure vessel, the hydrogen cooling pumps, the secondary cooling system, fuel handling systems and thrust nozzles fit into a bay that measures about seven meters long by four meters in diameter. The total engine weight is around 70,000 pounds (Figure 3), the engine power is around 4,600 megawatts, and the thrust-to-weight ratio is 1.3.

These numbers were chosen relatively carefully. We chose these operating levels so that we did not have to use space radiators in the system to remove excess heat from the moderator or pressure vessel. If you go much beyond this performance, you do have to start using space radiators to remove extra heat.

If you increase specific impulse to 2,500 to 3,000 seconds, you have thermal radiation dominated heating of the nozzle throat. There were studies done of nozzle throat cooling schemes to remove the radiant heat. That's an important technical question to tackle.

A VOICE: Radiation from the gas?

MR. LATHAM: Yes. The gas and the seed that is in it. The hydrogen flow through the nozzle is optically thick because it has tiny tungsten seed particles in it.

Elements of the nuclear light bulb program included closed loop critical assembly tests done at Los Alamos with UF_6 confined by argon buffer gas (Figure 4).

We also showed that transparent fused-silica, when subjected to a high intensity ionizing dose rates, exhibit a radiation damage annealing effect that restores transparency.

We did some work that showed that the fuel region could be seeded with constituents that would block UV radiation from the uranium plasma. That reduces radiation energy absorption in the fused-silica wall at wavelengths below the UV cutoff. That has to be verified experimentally.

Argon seeded with sub-micron tungsten particles to simulate seeded propellant was heated by thermal radiation from a high power dc-arc. The radiant energy passed through a fused silica wall to a propellant channel. A peak outlet temperature of 4500K was reached, which is equivalent to a specific impulse or 1,350 seconds for hydrogen.

It was shown by a combination of calculations and experiments that internal moderation produced a critical mass reduction (Figure 5).

In a 1.2 megawatt RF facility at the United Technologies Research Center, we used uranium hexafluoride and tungsten hexafluoride as the simulated fuel. We seeded the argon buffer gas with some fluorine gas to react with any fluorides that approached the containment walls. In final experiments, we were getting only milligrams of deposits in tests that ran about 40 minutes. The uranium fluorides are fuel forms that need to be considered for these applications, at least as initial fuel concepts.

A level 3 technology readiness for this concept is estimated.

What are the effects of new technologies (Figure 6)? Certainly modern computational fluid dynamics are going to tell us a lot more. We need to look at nozzle cooling designs and what the upper limit is on specific impulse. There are a whole host of materials that need to be readdressed: coatings, transparent materials, and composites, for example. Space radiator redesign should reduce some weight; we need to look at the reference engine generally with 1990's technology in mind. Mission architectures have changed and we have to work with new regulations with regard to testing, crew safety, and space operations.

Key technical issues include reactor and system stability (Figure 7). We didn't examine failure modes and safety, and we don't have estimates of operating lifetime. Fuel and buffer gas separation, handling and recirculation are areas that also must be addressed.

We don't know much about overall system reliability either. Correlation of fission versus electrically heated tests has to be addressed and verified. We also need to do experiments that validate that you can seed an optically thick plasma and control the spectral distribution of emitted thermal radiation.

We did some missions analysis for a Mars mission back in 1971. The characteristics of the systems used, which of course should be updated, are show in Figure 8. The assumed transit times were 140 days out and 245 days back, with an 80 day stopover (Figure 9).

The mission required four impulses; one impulse to get there, one to stop at Mars, one impulse to leave and one impulse to return to Earth. The reference engine required an initial mass in Earth orbit that was between a third and a quarter that of the solid core nuclear rocket (Figure 10).

The numbers in parentheses are the number of engines needed to leave Earth, number of engines needed at Mars, number of engines needed to leave Mars and, finally, the number to return. No notation means you can do it all with one engine.

For the next steps (Figure 11 and 12), more fluid dynamic analysis and nozzle cooling design work is needed. We should look at materials such as composites, coatings, transparent wall materials, and evaluate the NASP database to see what kind of materials are of use. We should redesign the reference engine using 1990's technology. Modern mission analysis should be done, as well as environmental assessments of the effects on crews by space and test operations. Then, we should define how to proceed.

What are the critical tests (Figure 13)? Cold flow and more electrically heated tests are needed to develop fuel recirculation and handling systems and also for demonstrations of fluid mechanical confinement. Using the same kinds of tests, we should investigate fuel and buffer gas circulation and reprocessing and measure the effects of spectral tailoring.

In the long term, nuclear criticality tests must be continued. Small scale low power tests and small scale high power tests can be done using the solid core facilities for fuel element tests. You can do a lot of proof-of-concept validation before you have to get to full scale testing.

The key point here is that you can piggyback nuclear light bulb experiments using solid core test reactors and facilities for small scale in-reactor proof-of-concept tests, thereby saving money.

Here is a cut at costs and schedule (Figure 14).

In closing, it's hard to review all the work that was done. But a lot of technology was considered some 10 to 20 years ago and in all cases, the feasibility of the nuclear light bulb concept continued to be demonstrable.

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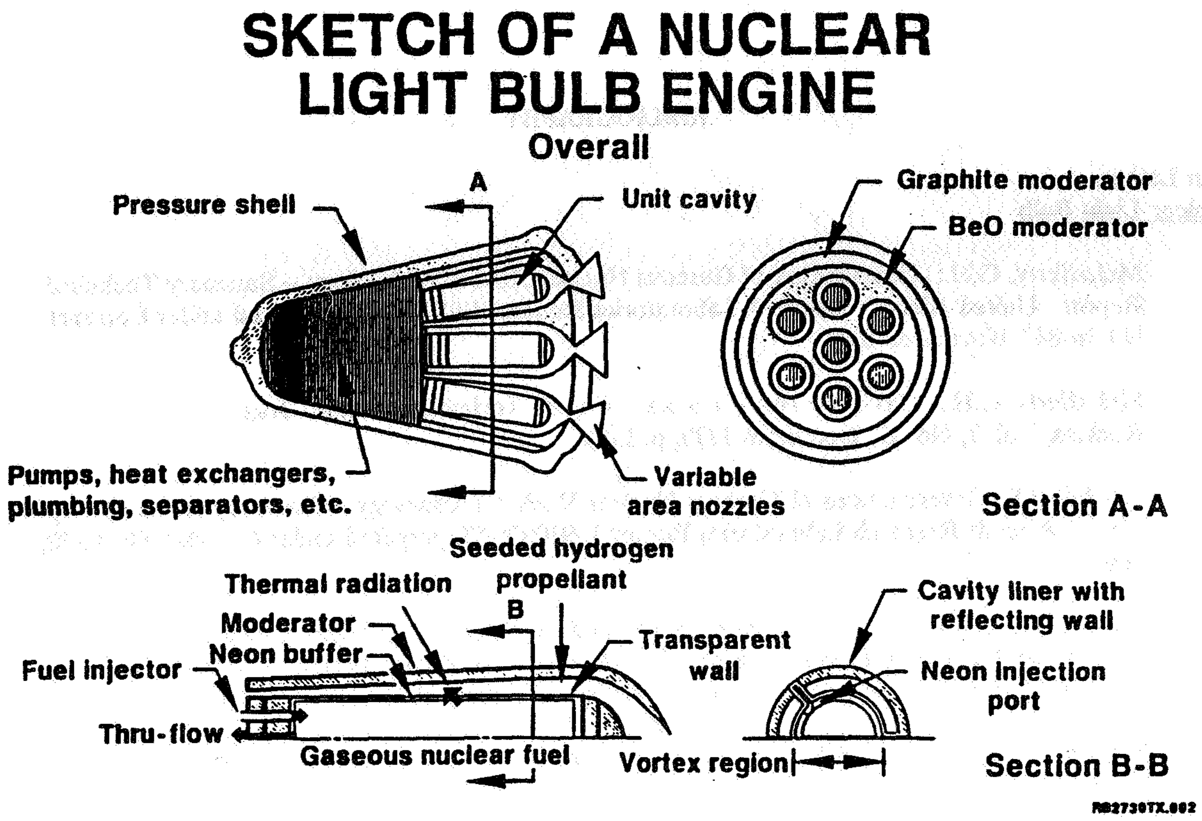


Figure 1

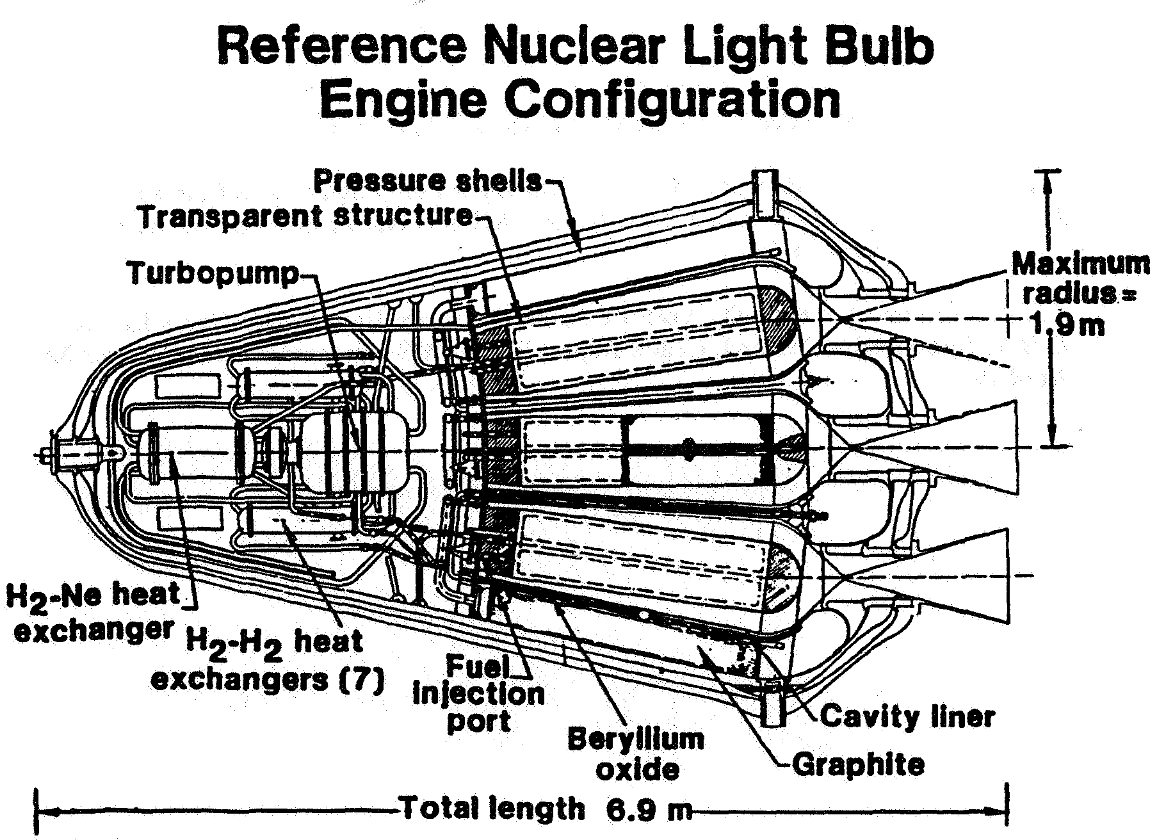


Figure 2

PERFORMANCE CHARACTERISTICS OF REFERENCE NUCLEAR LIGHT BULB ENGINE

Engine weight	70,000 lb
Engine power	4600 MW
Total propellant flow	49.3 lb/sec
Specific impulse	1870 sec
Thrust	92,000 lb
Engine thrust-to-weight ratio	1.3

Figure 3

GAS CORE NUCLEAR REACTOR

Program Achievements

- Flow Containment Demonstrated
  - Cold Flow
  - RF Plasma
  - Closed Loop Critical Cavity Assembly
- Energy Coupling Demonstrated
  - RF Plasma
  - Radiation Annealing Effect
  - Buffer Gas Tailoring
  - Seeded Propellant Heating Test
    - Equivalent Isp Approx 1350 sec.

Figure 4

GAS CORE NUCLEAR REACTOR

Program Achievements (Cont.)

- Internal Moderator Benefit Confirmed
  - Almost 3:1 Reduction in Critical Mass
- Flow Rate Control Demonstrated
  - Closed Loop Argon-UF6 Vortex Flow Syst.
    - Los Alamos Critical Cavity Assembly
    - Seven Tests
    - Achieved 20 KW for Approx. 100 sec.
    - No Unexpected Fluctuations
- Technology Readiness Level = 3

Figure 5

IMPACT OF NEW TECHNOLOGIES / SAFETY REGULATIONS

- Computational fluid dynamics
- Cooled nozzle design
- Materials
- Space radiator design
- Reference engine with 1990's technology
- Mission architectures
- Environmental and crew safety

Figure 6

KEY TECHNICAL QUESTIONS

- Reactor / system stability over all operating conditions
- Failure modes and safety impacts
- Operating lifetime / performance envelope
- Fuel / buffer gas separation / recirculation system performance
- Overall system reliability
- Correlation of electrically heated demonstrations to fission heated operation
- Validation of spectral tailoring of radiant heat flux

Figure 7

ENGINE SPECIFICATIONS

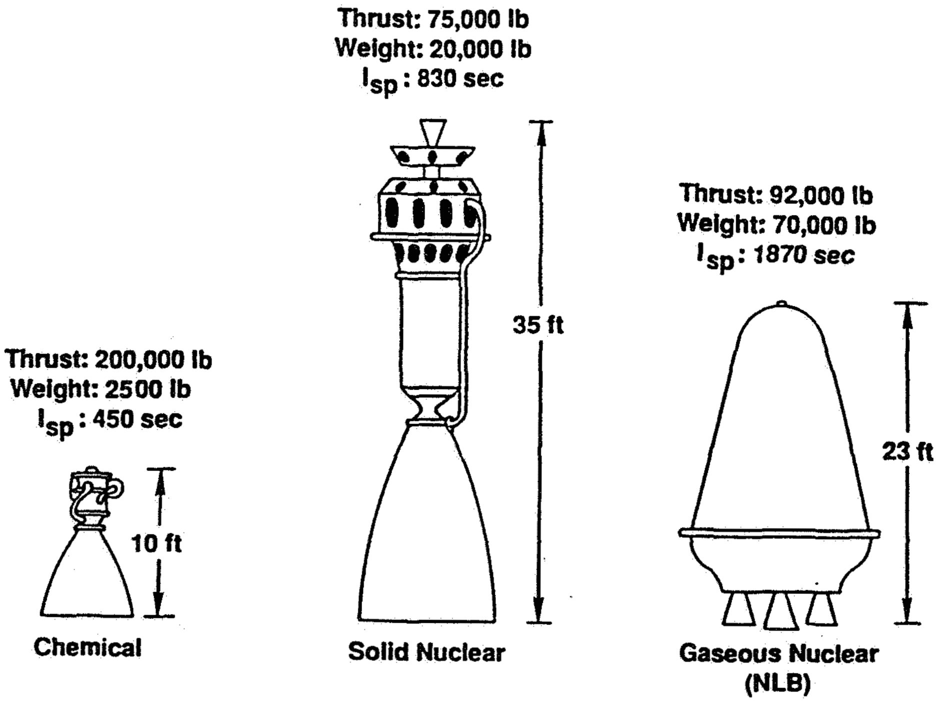


Figure 8

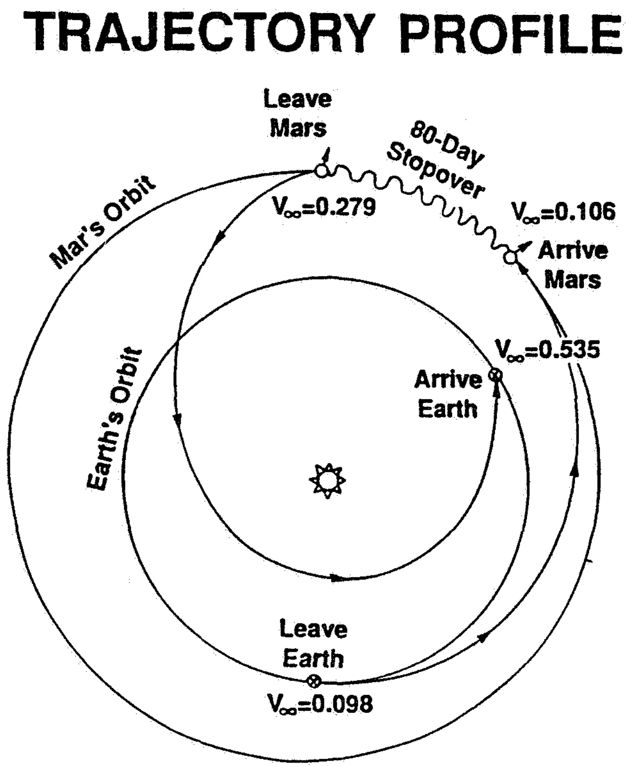


Figure 9

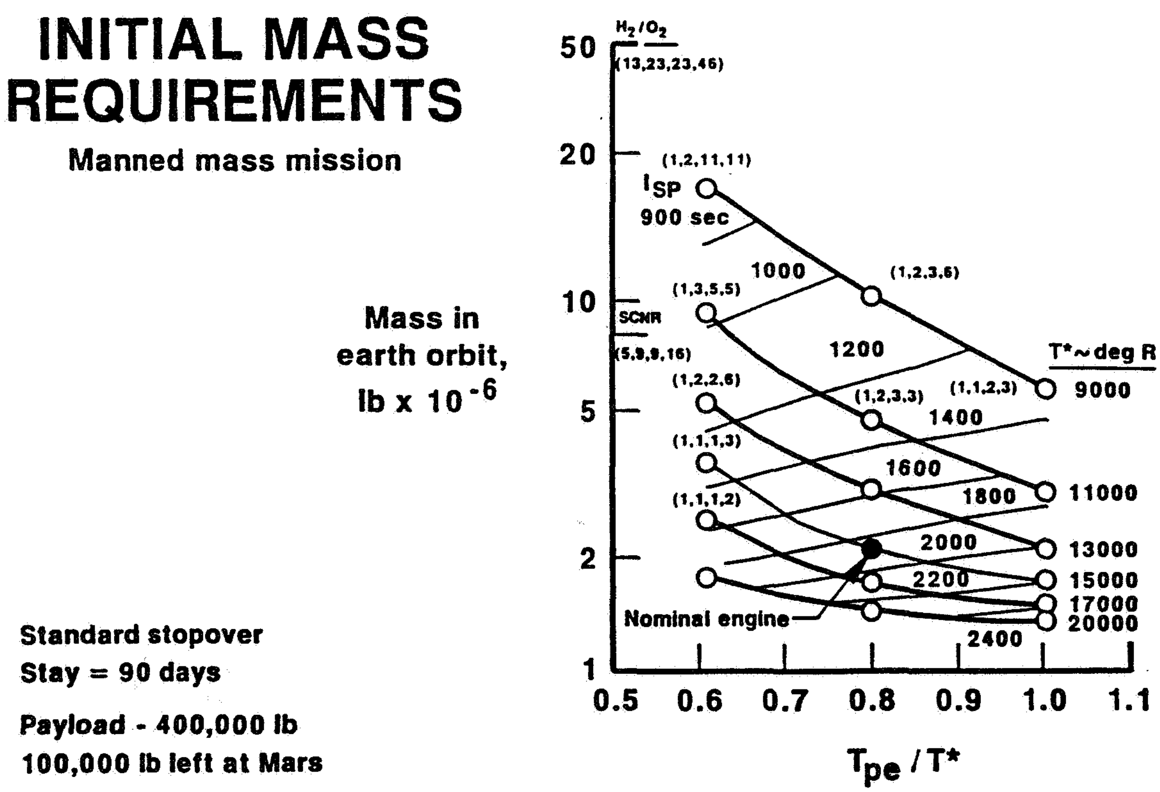


Figure 10

GAS CORE NUCLEAR REACTOR

The Next Step

- CFD Analysis/Design of Cavity
- Cooled Nozzle Design
- Materials Evaluation
  - Radiation Damage
  - Composites and Coatings
  - National Aerospace Plane Data Base
- Redesign Reference Engine
  - 1990 Technology Level
    - Advanced Turbopump Concepts (SSME)
    - Advanced Diagnostics
    - Fiber Optics
  - Launch and On-Orbit Operations
  - Fuel Reprocessing System
- Mission Performance Analyses

Figure 11

GAS CORE NUCLEAR REACTOR

The Next Step (Cont.)

- Environmental Assessment
  - Earth Development Facilities
  - Launch Facilities and On-Orbit Operations
  - Operations and Crew Impact
    - Direct - Radiation Exposure, Vehicle Design
    - Indirect - Mission Profile, Duration
  - Lunar and Planetary Outposts
- Test Options Evaluation
- Test Program/Facilities Definition

Figure 12

CRITICAL TESTS

Near term (non-nuclear)

- Cold flow model validation
- Electrically heated, hot flow confinement tests
- Fuel/buffer gas separation and recirculation
- Spectral tailoring
- Nozzle cooling limit

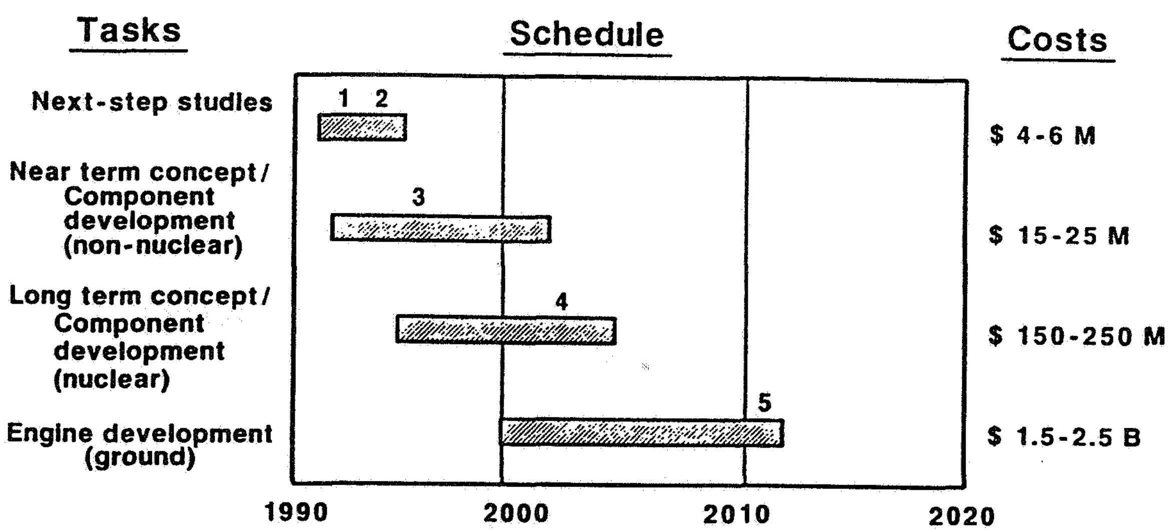
Long term (nuclear)

- Reference engine zero power criticality
- Small scale, low power, flowing critical tests*
- Small scale, high power (fission plasma) flowing critical tests
- Unit cell, high power (fission plasma) flowing critical test **
- Full scale, full performance reference engine tests

* Control, stability and confinement
** Control, stability, confinement, fuel handling, spectral tailoring, propellant heating

Figure 13

NUCLEAR LIGHT BULB DEVELOPMENT SCHEDULE AND COSTS



1 - Program plan
2 - Facilities/Test plan
3 - 2000's reference engine design
4 - Technology readiness verified
5 - Engine ground qualified

Figure 14

QUICK TRIPS TO MARS

R. HORNUNG
Boeing Aerospace
and Electronics

We started out with a point design. We used the chemical propulsion weight statement for option five that Boeing Huntsville had been using, which totaled out to 731 tons. We took a shot at applying nuclear thermal propulsion to this, but not to exceed their weight estimates (see Figure 1).

We put together a vehicle essentially of two components that would have to be launched by two very heavy lift launch vehicles.

In what I call the second stage, which provides the Delta V to Mars, there is one group of tanks and then the upper part which has the habitat module and some storable propellant for MEV, which is in the second upper part of this vehicle (see Figure 2).

Once the vehicle is assembled in low Earth orbit, you have three NERVA boosters with a fourth in the center that acts as a dual mode system. The fourth generates electrical power while in route, but it also helped lift the vehicle out of lower Earth orbit.

I thought it was a good idea at the time to have three NERVAs here basically because of shielding. At that point in time I thought it would reduce the shielding. But based upon advice I have had from some of the shielding experts, most of the gamma is gained or emitted during the time you are firing, and so getting rid of those boosters before you head for Mars doesn't help your shielding problem that much. I would suspect in the future when the vehicle is optimized you would probably end up with maybe one NERVA in the middle and it will also be in dual mode.

The major portion of gammas are produced when you are firing. They are still there after firing but not as serious as they were: that's what the shielding people told me. I thought it would be a good idea to get rid of those boosters before you left for Mars, but it is not that beneficial.

You first fire all four of these engines for about 40 minutes, each one using about 100 tons of hydrogen. If you recall in a previous table there were 435 tons of hydrogen on the chemical vehicle. So you fire of those 40 minutes and get a Delta V for about 70 kilometers per second.

As you are firing, the center dual mode system continues firing longer than the three outside ones and eventually fly away from them. I show a very strong hard back through the core, up to the Apollo module, where there is another tank. It is designed such that when you are flying away from it there is an incline ramp that pitches your three other boosters off so they separate, and you continue on towards Mars. You have this strong back and when it gets through firing, the final three tanks come off. These three tanks hold what only one of the first three provide.

To give you some idea of scale, the first three tanks on the cylindrical part are each 100 feet long. They are 26 feet in diameter. The other tanks are also 26 feet in diameter. So each one of these stages holds about 100 tons of hydrogen, for a total of about 400 tons.

After they separate, then you go into a dual mode operation. I have not applied any nuclear electric propulsion in this mission. This particular electric power generation capability has been sized to generate 2.5 megawatts of electricity.

Now, I have said it is housekeeping power. But something I would like to evaluate in the future is what could you do with that 2.5 megawatts in the way of course corrections through electrical propulsion. The RCS's and ACS's generally use some kind of mono or bi-propellant and I would like to look into that. I think there is enough power there to do something beneficial for those subsystems, at a lot higher specific impulse.

There has been some discussion of dual mode here at this workshop and so I would like to present our case for that later. You are on your way to Mars and you have this electric power available for housekeeping and propulsion.

On the forward end of this vehicle we have a Mars Ascent Vehicle (MAV) and a Mars transfer vehicle stage. Mars Exursion Vehicle (MEV) is on the front also. They both have heat shield designs on them. The data from the Boeing workshop on aerobraking has a number of different concepts for aerobraking, heat shield and so forth. My thought is that this could be either a deep dish type, conical, or spherical, but the main feature this one would have, that the ones in those workshops didn't have is a way of deploying extra fins to increase cross-sectional area to at least the diameter of the aero shield they talked about. The reason I do that is to keep this vehicle here a total diameter of 56 feet. That's the outside diameter of the vehicle. But when it deploys, you are up close to 100 feet, like a 30 meter aerobrake.

This diameter would require a very heavy lift launch vehicle. It doesn't exist. I don't know if you know what the ALS of Boeing looks like, but they have a module that is recoverable on a tank that's expendable. I would see a number of those stacked around a central core tank like an ET, only maybe even larger than an ET. They think they could do a 56 foot diameter. In other missions, missions they called hybrid, you use the liquid oxygen to burn the solid so you can control them. They haven't drawn up any concepts yet. I hope by the first part of September they will have and I can make a configuration out of that. But that's why the 56 foot is about as large as I thought it could go based on what they told me about those potential boosters that they could put together.

I would like to discuss the unfolding of the heat shield. These fins can be either deployed thermally in a passive mode or electrical thermally in an active mode. You can't see it on these small drawings (see Figure 2), but there is something I would call flecture backing on the fins that are made out of this memory metal. When it gets up to its transition temperature it goes from Martensite to Austenite and then returns to its original shape. When it is in the Martensite structure and you bend them down to start, so that when you go through the yield region you don't yield beyond an eight to ten percent, you bring them back down and clasp the whole thing with a circumferential strap. When you get near Mars you pop the strap off, using what we call Nitinol actuators that they have been building at Boeing for other programs (so we know they work).

I mentioned passive and active operation. In order to make this passively stable, I have put this skirt so the center of gravity is up as far to the nose as possible. When you do hit an atmosphere with it and it starts heating, say it starts heating on this side, when it reaches 350 degrees Fahrenheit, this material will transition and straighten out a fin. Then you start braking more. If it starts flipping over, the other side would heat more and it would start deploying and if you weren't happy with that you could thermally deploy these before you hit the atmosphere or deploy the whole thing or parts of it by heating that metal electrically. Since I have 2.5 megawatts available, I can do a lot of heating; this metal will transition as fast as you can heat it.

The MEV goes down to Mars on its own. The skirt of the heat shield becomes its landing gear and it stays behind when the subsequent stage goes away. When the subsequent stage goes away, the middle part of that heat shield comes back up with it. I was told that this adds an extra penalty or scar weight on the propulsion system. Now, at this point in time I haven't tried to change any of the weights of the MEV other than what was on that table for the chemical. I just used their weight statement. I realize there might be scars there that hurt the system.

The rest of this vehicle did propulsion brake with a storable propellant. If we had hydrogen I would be working with 900 seconds for Isp. The propellant we used through the reactor is a much heavier molecule and I am guessing its weight density is about 45 to 50 pounds a cubic foot.

Its Isp will only be about 480 seconds and that's the reason I said the quick trip thing is in quotes, that 480 Isp hurt us on the return.

We still burn that same propellant with the same reactor when we return to Earth. It goes back into the dual mode operation with the deployable radiator that recovers again. The Nitinol is also used again. It reminds me of those things you have at a New Year's Eve party that blow out and come back automatically.

I had originally thought it was a good idea to bring as much of this vehicle back as possible for refurbishment in the space station orbit. However, I have been told that it is not necessarily a good idea for NTR to bring a mass penalty back with us. That can be decided in the future.

When we separate again, part of the vehicle becomes the heat shield aerobrake. It might go into a long elliptical the first time around and brake, but I show it here once around. Then you drop off the Apollo type capsule and reenter back to Earth.

Figure 3 shows the typical orbit we would have: outbound and inbound with a potential Venus swing by.

Figure 4 has an error here. Line four in mass allocation should read "LH2 and storable propellant" and the number should be 557. There are about 438 tons of LH2 in the propellant budget.

The trip time outbound didn't come out as well as I thought. I had been doing some calculations and I thought it was going to come out closer to 154 days. It didn't when we ran it on the program. However, the person running the program didn't have time to do any optimization. He picked what he thought was the best trip start time. I noticed he picked February of 2016 which is the right year. But it looks like April would have been better.

The return trip is not good; it is 300 days. As I said before, the Isp killed me going back to 480 from the 900. However, I don't have to carry liquid hydrogen all the way through this trip. I think that might be a problem. I am not convinced you can store liquid hydrogen that long without a considerable loss.

I would like to speak a little bit about the dual mode. The center part of this vehicle, the central core reactor, would be a system that's laid out in Figure 5. This was actually laid out for a LTV that was stowed in the shuttle. When we got to sizing it, we found it didn't leave much room for payload.

You see in Figure 5 the hydrogen source. We have done some trades for other gases. Hydrogen, Helium, Xenon and so forth. However, during the closed mode you have a valve that has to close; that's one of the technology problems. There are concerns over the valve being in the line of a direct nuclear propulsion system. We think this configuration can be designed based on some technology that exists for the Pegasus engine used in the Harrier aircraft. They have some ducting that controls the thrust vector on their jet engine. They think they can do that same type of technology for a little bit higher temperature. We are in dual mode. We are up to 700 degrees coming out of reactor, so we think that valve can be developed without a lot of risk.

The generators sit around the end of the design unit (it is a Brayton and closed cycle incidentally). They are being driven by a turbine. This system was originally designed to have some burst power. Figure 6 shows a schematic of that system and it shows you the burst power capability.

Figure 7 shows the variables we keep track of when we are doing the evaluation on this.

In Figure 8 we see the results for a hydrogen working fluid. It wasn't laced with anything. With a radiator sink temperature 0 degrees Fahrenheit, you can see this was done by the mechanical engineer based on the units (BTU's per second). Efficiency came out 29 percent. Specific weight is 5.4 kilograms. Notice on Figure 8 the system weight, and radiator area. That's why I used the size for the one I used on the Mars mission.

The radiator is 58,000 square feet. Keep 50,000 in your mind. Based on what fluids you use, it is around that. It is a low temperature radiator.

We are only coming out of the reactor at 1,700 degrees Fahrenheit. It is more efficient with the regenerator. Like I mentioned before there is a valve; a technology area. This radiator is something we needed. I think I can develop the concept for that, but we need to do something to test that in conjunction with Battelle looking at fabric radiators. If I take the regenerator and I put the radiator in, the efficiency drops way down.

In Figure 9 we use a helium xenon working fluid. The turbine people like a heavier molecule, but when you optimize the whole system, the previous one with hydrogen was better. You see the efficiency dropped a little bit. I think this is a little lower, the mass is a little lower.

When we were working this, turbine people wanted to spin faster and so forth to get their system smaller. If you put multiple turbines in, you don't really have to worry about the size so much. The generator people don't like to spin so fast.

The hydrogen system provides slightly more efficiency on the overall system, but it's heavier. It depends which way you want to go with the system.

Figure 10 is one without the regenerator; the efficiency went way down and you also are heavier.

The next step here is (and we have talked about this at JPL) you have to find out what kind of power you need. Power conditioning here needs to be married into this. If you are going to use NEP you need those thrusters and so you need the propulsion people and the power conditioning people to get together. That's a big headache.

With the turbines and the generators that you see in Figure 5, you might be able to give the thrusters the kind of power they want directly without much power conditioning. This is a closed Brayton cycle. In the burst power it is open Brayton. If you want a lot of electrical power in the burst mode you are dumping the hydrogen into space.

When you are talking about storing hydrogen for a long time it is a scar weight or you have a refrigerator to carry along with you: refrigerator plus electric power requirement to run them, especially if you are going to use them there for a long time.

Boeing has looked at a couple of areas. That tank I showed on this system at the beginning is about a 12 thousand cubic foot tank. Boeing has designed a tank a long time ago about that size that they thought would lose about seven pounds of hydrogen a day in space. I figured out the numbers once and even with that tank, they put a lot of MLI on it and it is not a dewar. I have a feeling if you are going to carry hydrogen around until you leave Mars, you need a dewar. I don't know what the weight penalty is on that, so I can't say.

A VOICE: It you went with lighter weight tanks and compensated the light weight tanks and a little higher thermal input with electric powered coolers, you could meet the same requirements.

MR. HORNUNG: Do you know what they weigh?

A VOICE: That's just it, I don't think anybody ever looked at that.

MR. HORNUNG: That's the question. We were working on one where you pump hydrogen gas through a membrane and you can get very deep cooling; down below the typical minus 423 degrees Fahrenheit. That looked good but those things start stacking up and if you want any large quantity it becomes a horrendous weight. Somebody has got to look at that part if you want to carry a tank along.

A VOICE: You have a thing here that's talking about radiation sink temperatures of zero degrees F. Don't you think that's bit conservative?

MR. HORNUNG: Yeah, the guy that did this is conservative. He has been around Boeing some 35 years and he has been burned a few times, he was a little reluctant to do this analysis because he didn't know all about the application and so he was conservative and we only had 10-K. So I didn't have money to go back and have him do it again. At the time I forgot to tell him what I thought the space temperature might be and thus it is conservative system, overdesigned in a sense.

A VOICE: I noticed the vehicle swings into Venus orbit and I would think that that might be optimistic. You are saying you can orient the radiator?

MR. HORNUNG: I was hoping during most of the orbit the sun would be over in the right spot so the radiation would be looking at the radiator on edge but I don't know that to be true. This incidentally is a double sided radiator.

A VOICE: You are not thrusting, so unless there is a crew requirement or a heating requirement on the tanks, it doesn't matter.

MR. HORNUNG: Even if I put in NEP in a certain region of the mission for course correction, I can vary those as long as you put it out near the CG somewhere.

A VOICE: I guess the other thing is that the radiator is just a figurative depiction there?

MR. HORNUNG: Well, I took a little bit of artistic leeway.

A VOICE: I am worried about when it is deployed. I am not sure you have a two part shield on your reactor.

MR. HORNUNG: I am not worried about radiation on the fabric of the radiator.

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Quick Trip to Mars

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Mars Vehicle Mass Statement

(Drop aeroshell; Earth return via ECCV)

2016 MASE. #2 delta V set: E dep $\Delta V = 4281$, M dep $\Delta V = 3400$

25 t surface cargo 12/1/89 run: marsbsl15.dat;156

All masses in kg.

Crew of 4	Option 1	Option 5
MEV crew module	3478	3478
Ascent stage inert mass	3099	3099
Ascent stage usable propellant	17292	17292
Asc RCS propellant	208	208
Ascent stage at liftoff w/o samples	24077	24077
Boiloff	163	163
Ascent stage landed mass	24240	24240
Landed surface cargo	25000	25000
Total descent payload	49240	49240
Descent stage inert mass	6012	6012
Descent RCS propellant	3023	3023
Descent propellant	13841	13841
Lander aeroshell	9376	9376
MSRV	0	4000
Mars excursion vehicle (MEV) gross	81496	85492
Mars transfer crew module & equipment	30384	30384
Consumables	5808	5808
Trans-Earth Injection Stage (TEIS) inert mass	11857	11870
Earth Crew Capture Vehicle (ECCV)	7000	7000
Earth return cruise mass	55049	55062
TEI usable propellant	61108	61121
Inbound midcourse maneuver propellant	1081	1082
TEI vehicle departure mass	117238	117265
Mars capture aerobrake	20717	20717
MEV gross	81496	85492
Mars capture mass	219451	223474
Boiloff	4457	4458
Outbound midcourse maneuver propellant	5938	6043
Comsat (separation before Mars capture)	3000	3000
Interstage structure	1000	1000
Trans-Mars injected mass	233846	237975
TMI stage inert mass	55000	55000
TMI propellant	435252	438459
Initial Mass in Earth Orbit (IMEO)	724098	731434

Figure 1

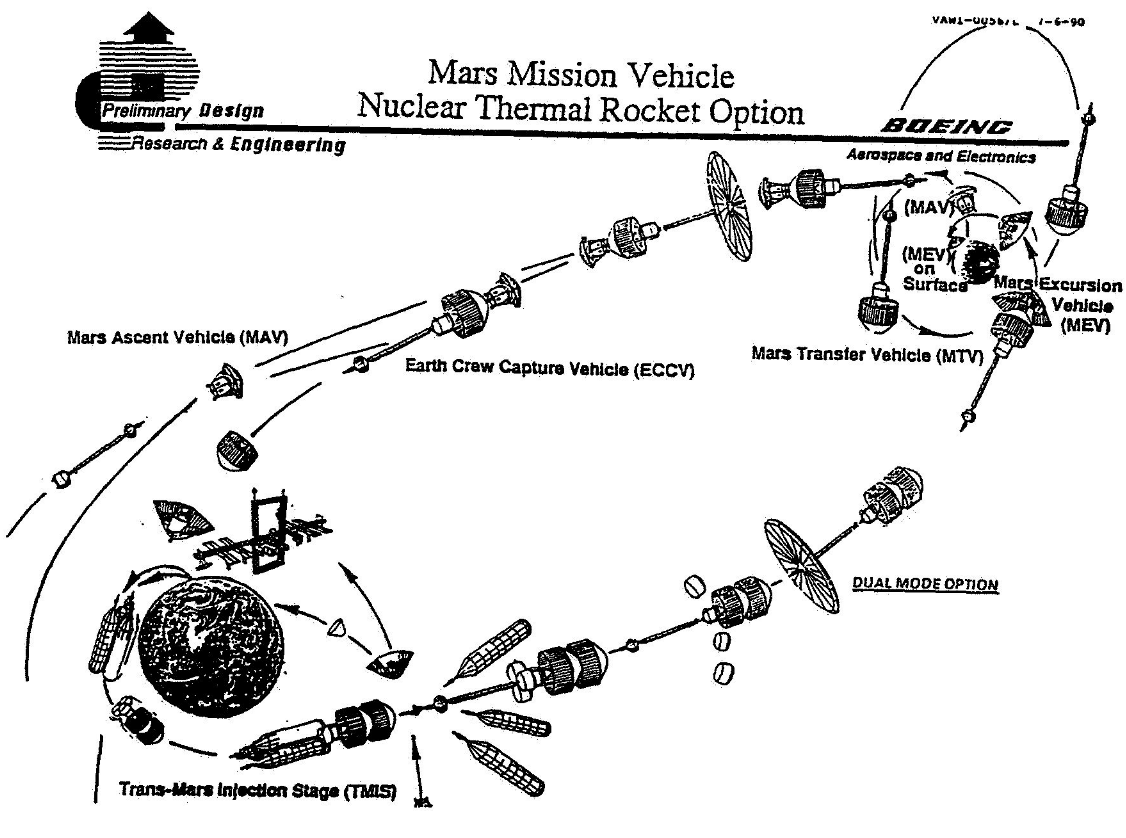


Figure 2

Aerospace Systems Technologies

NTR MARS MISSION

Boeing Aerospace & Electronics

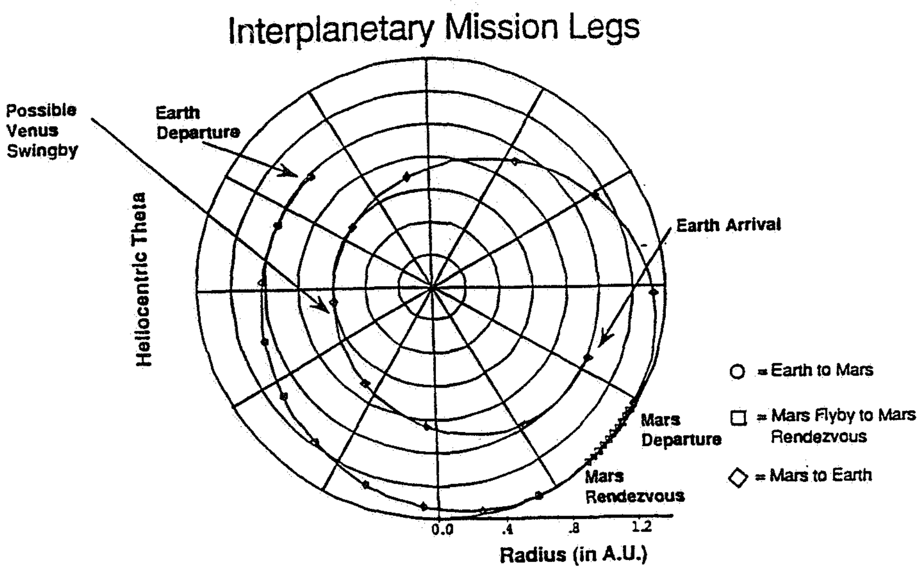


Figure 3

Aerospace Systems Technologies

NTR MARS MISSION SUMMARY

Boeing Aerospace & Electronics

Mission Parameters

Specific Impulse-NTR	900 sec

Lunar and Planetary flybys are being investigated for possible performance gains. At present, the vehicle must fly by the Moon in order to receive a gravity boost, enabling a quicker transfer time to Mars. A quicker Earth-Mars transfer reduces subsequent delta vee requirements and provides the opportunity for a Venus flyby.

Mass Allocation

IMLEO (SSF orbit)	732 MT	
Payload Outbound	84 MT	
Payload Inbound	40 MT	
LH2 Propellant	577 MT	** Preliminary data
Stage Mass	55 MT	
Vehicle Dry Weight (after staging)	16 MT	

Mission Summary

Outbound Trip Time (Including Earth escape)	200 days
Inbound Trip Time	300 days
Stay Time	30 days
Departure Date	February, 2016

WGV7/09/90
Disk 1

Figure 4

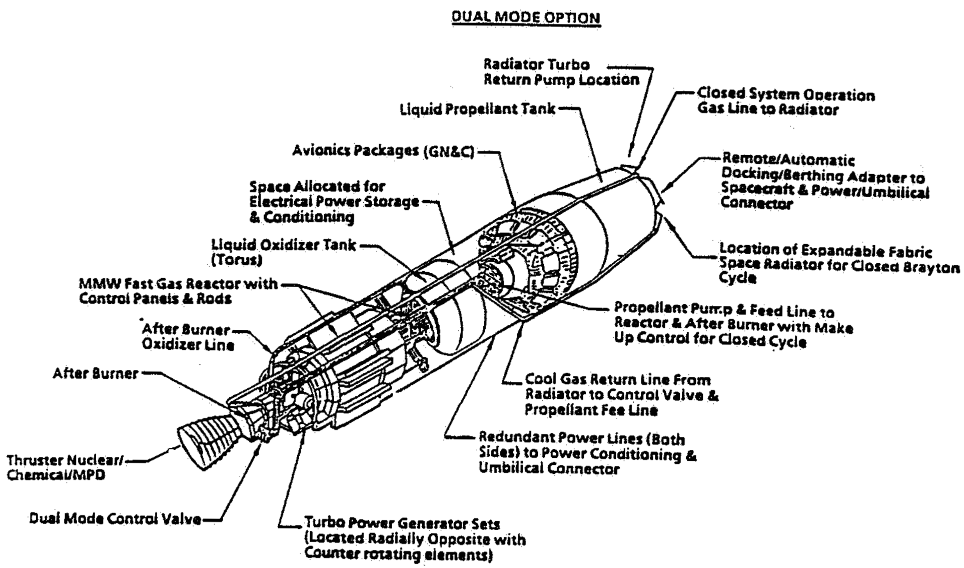


Figure 5

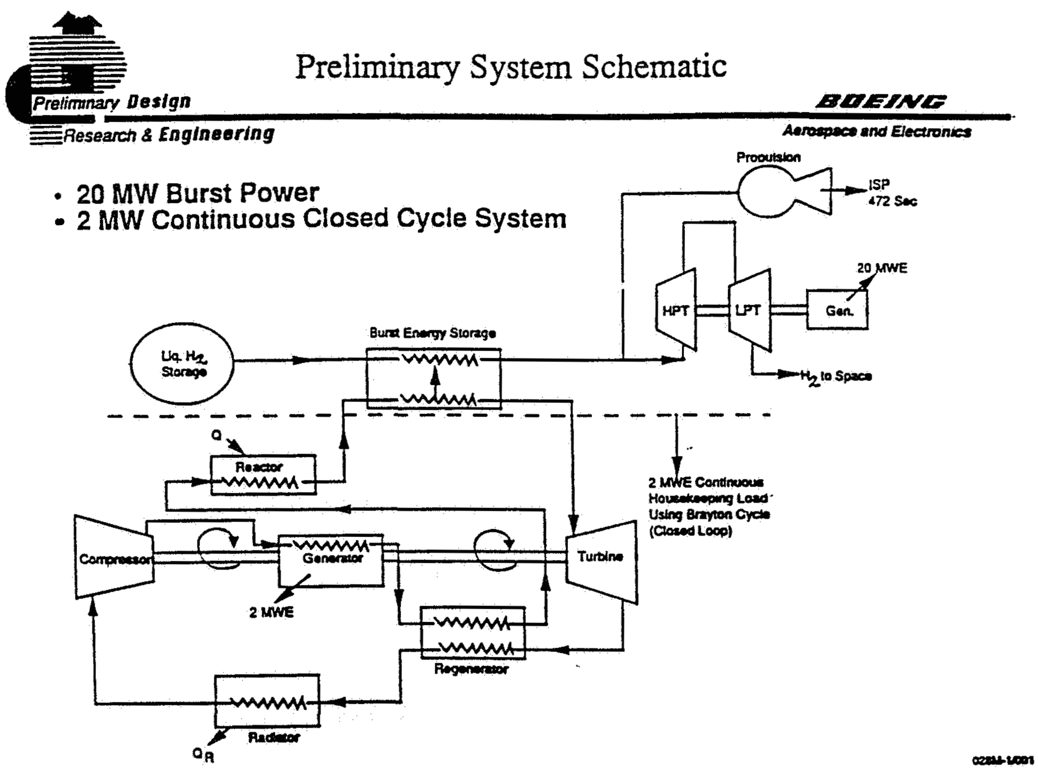


Figure 6

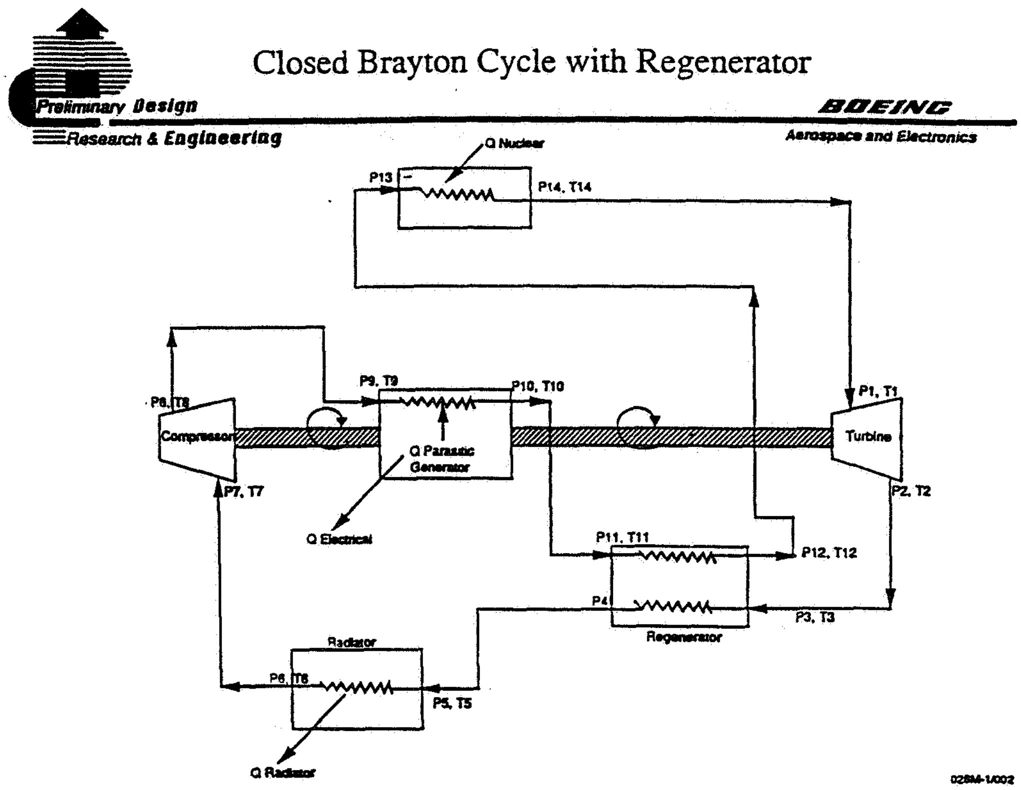


Figure 7

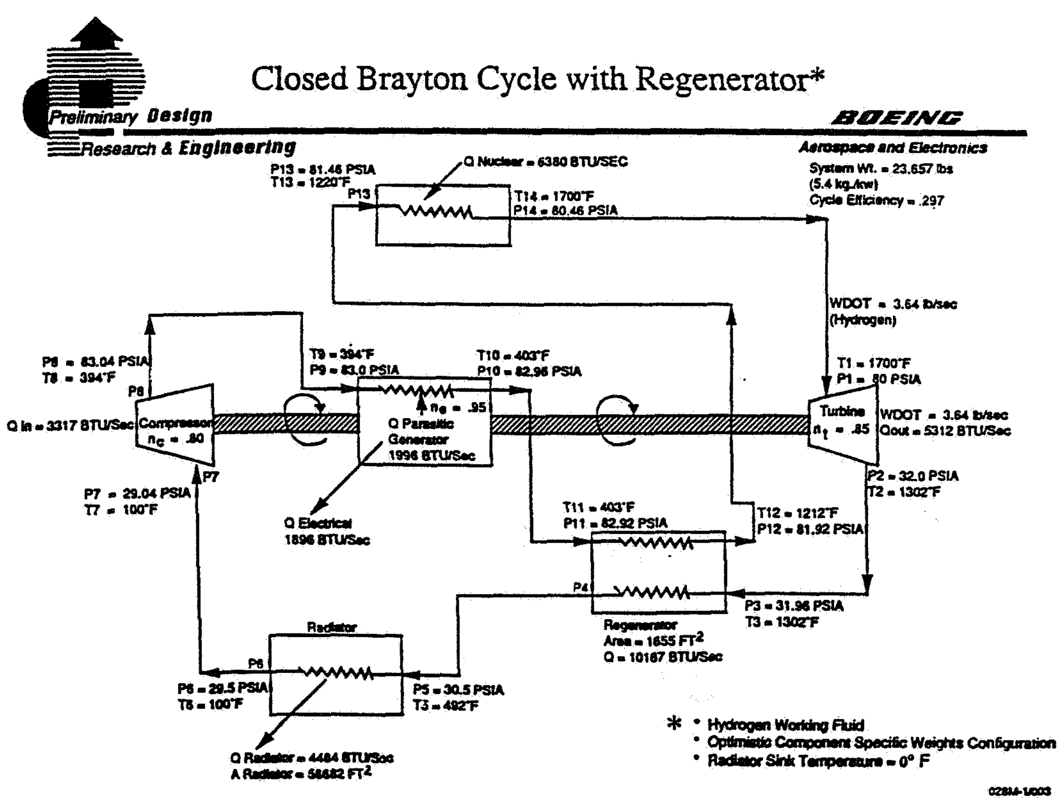
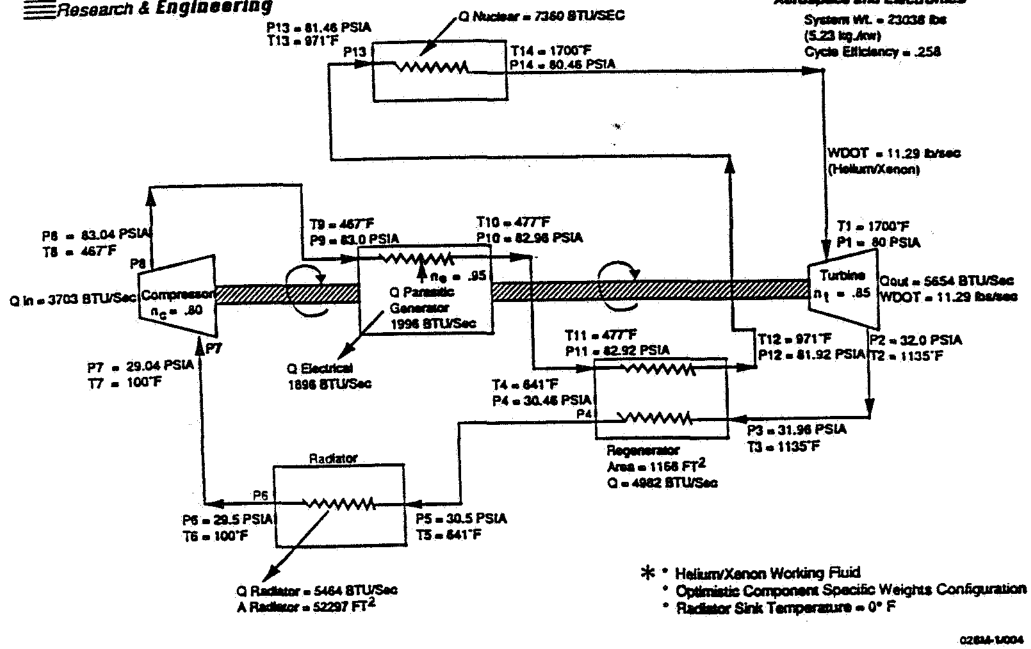


Figure 8

Preliminary Design
Research & Engineering

Closed Brayton Cycle with Regenerator*

BOEING
Aerospace and Electronics



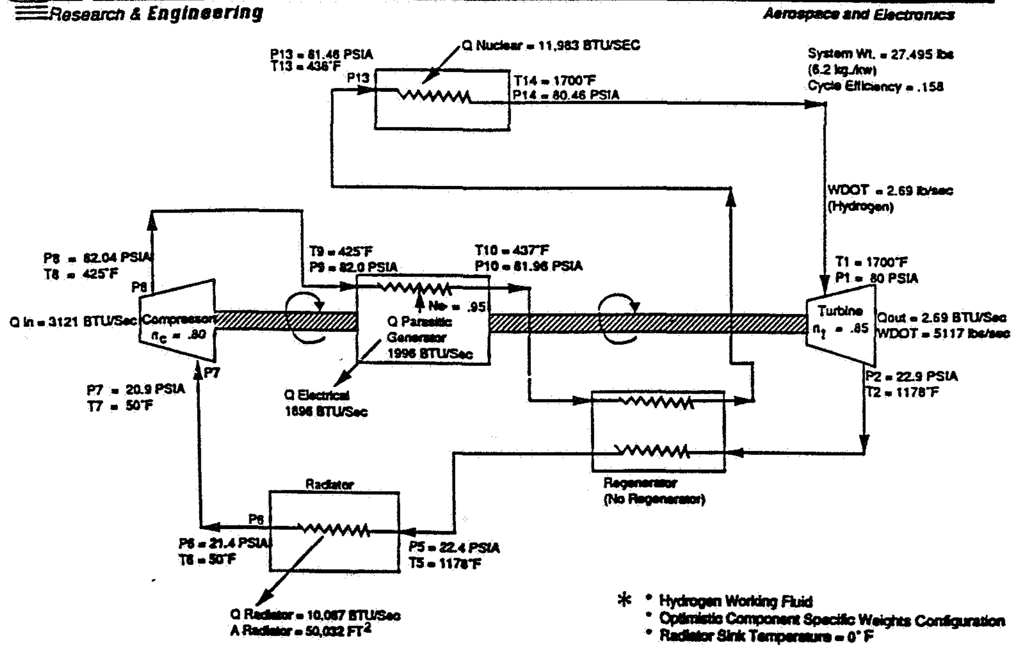
* • Helium/Xenon Working Fluid
• Optimistic Component Specific Weights Configuration
• Radiator Sink Temperature = 0° F

Figure 9

Preliminary Design
Research & Engineering

Closed Brayton Cycle Without Regenerator*

BOEING
Aerospace and Electronics



* • Hydrogen Working Fluid
• Optimistic Component Specific Weights Configuration
• Radiator Sink Temperature = 0° F

Figure 10

PROPULSION SYSTEM NEEDS

STANLEY GUNN
Rocketdyne

I would like to pick up where I left off in the presentations yesterday relative to the needs of the designer of a solid core nuclear rocket engine; things that he needs to do or needs to understand in order to focus his design effort and his analysis.

Now, I will put the "strawman" propulsion requirements and engine cycle up to illustrate a couple of points. Chamber pressure is first.

This particular design (see Figure 1) was very arbitrarily selected at about 1,000 psi and I will go through the factors that led us through that selection. There is a lot of flexibility going up or down in the chamber pressure.

The impact of this parameter is mainly going to be in the size of the engine (particularly the size of the nozzle assembly). It is important that we understand what the trades are in terms of the size and the configuration of the overall envelope of this engine.

Assuming that we wanted to go for an expander cycle (see Figure 2) in order to maximize our chance to get reasonably high Isps, we did not want to have some of the total propellant flow not contributing fully to the thrust and the Isp we can realize. If all of the weight flow is heated to the maximum temperature and is expanded to the full nozzle expansion ratio (epsilon), you will always do better.

The point is that the designer has several sources of energy to be able to heat the working fluid that's going through the turbine. It doesn't have to be all of these sources -- just those that heat the working fluid to the desired turbine inlet temperature before joining the main flow when it goes down through the core on the final pass. The source of energy might come from the cooling of the nozzle.

Now, that's a poor place to get the heat if you can avoid it because it is coming out of the gas that is expanding through the nozzle of the main thrust. It represents a small, but still finite impulse loss. It is better to take it out above this station because any energy you take out in the form of expansion across the turbine stages can be added back in by the core. You can get that for free, except for the weight of the structures involved.

One attractive place to get it is from a coolant loop in the core. This loop is used solely for the purpose of heating that portion of the total flow that is used to drive the turbine. In the kind of reactors that we looked at in the past, the tie tubes were also an attractive way to get that energy.

In this particular system diagram (see Figure 2), we shown a split. I will assume, for the total flow coming out of each pump, 50 percent goes down through the tie tubes and 50 percent goes down to cool the nozzle, and up through the reflector. Of that portion that goes through the turbine, a portion bypasses the turbine for control. The flow needed to actually develop the shaft horsepower goes down through the turbine and then down into the core, joining the flow from the reflector. All of this becomes one approach.

It is only necessary that we provide the measures needed in order to get adequate turbine pressure ratio and adequate pressure drop across the passages. They don't have to be balanced. They will get balanced when they come together. At that point there may be some imbalance, but it doesn't really hurt you any.

Now, we really have two very powerful knobs, in addition to weight flow, to get the necessary shaft horsepower.

In Figure 3 we have plotted chamber pressure as a function for various pressure ratios across the turbine (e.g., 1.1, 1.25, 1.52) and for various temperatures coming into the turbine. From this you can pick off the case where there is turbine bypass and which represents the maximum power that you can get.

Now, in this particular case you can see the trend. Fairly cool gas with a fairly high pressure ratio can get you to reasonably good chamber pressures. If you go on up in temperature of the gas, you can get higher chamber pressure and if you go on up to higher pressure ratios, you can get higher yet. If you go up to 1,000 R, your way is open to go to very high chamber pressures.

The question really is where do we want to be in pressure chamber? Only a portion of flow is going through the turbine because we don't need it all: e.g., less than 50 percent of the case we want through here. We can get the shaft power.

If the control valve is shut, the result is the condition of maximum power that I can develop from the portion that I have diverted into the turbine loop. Remember now that I took the other half of the total core flow and said I am going to use it to cool the nozzle and the reflector.

So we have a capability of going on up in chamber pressure and coming way down in the size of the nozzle, if that's what we should do from the engine size point of view.

Now, there are some that feel high chamber pressure equates to high risk. Well, I would like to say that I believe that's a myth. It's doesn't having anything to do with chamber pressure; it has to do solely with design margin.

A VOICE: Off the top of your head, would you accept as you went up in chamber pressure that flow -- required to power that cycle would have to and become a point where you can't go any higher.

Don't you run into a point where your area of the throat really becomes too small and you have a heat transfer problem on it?

MR. GUNN: What we did in our Phoebus designs is to go into tube splices or joining together of coolant passage design so you can cool down to very narrow slots in the design of the throat area. But you are right, there is a limit.

But I am trying to make the point that at 1,000 psia or 1,500 or even 2,000, there is no reason that you can't pick up and get an adequate design with an adequate throat area. For the hundred thousand pounds of thrust we have talked about, we have an eight inch float area.

A VOICE: You are really limited in chamber pressure by what you are comfortable with and the maximum heat flux of the throat?

MR. GUNN: True. But, I am not advocating up to 3,000 or 4,000 thousand psia. I am saying it is within the range of 1,000 to 2,000. I am comfortable that we can come up with a design that will work. But the real question is do we want it that high for reasons of size?

Now, there is another benefit that comes out if you go up in chamber pressure and that is the density of the working fluid that is going down through your core is increased and therefore the pressure drop across the core is reduced. But you are removing more heat per channel and the thermal stresses in that fuel element are going up. At some point, it is going to be the power density that limits the increase in chamber pressure/thrust level.

I should say that this is based upon a solid type of core where you have the thermal stresses associated with where the heat is generated how it gets to the surface. You are right.

This chart (Figure 4) starts off with the old famous Phoebus 1B test that we ran. What we did to come up with these parameters was simply take the test data that we had from that run, relative to the reactor, and put an engine cycle around it that was an expander cycle.

Now, note on this particular setup we have started off with a reactor exit pressure that was about 750 (735) psia. We ran that test and we said if we had simply sped up the pump, gotten more pump discharge pressure, we would have gotten to a higher chamber pressure and higher power level.

We had put design margin in the Phoebus 1B test test hardware to go to 1,000 psi. We would have gotten the 2,000 megawatts. One route to get more thrust out of your engine and your given reactor is to simply speed up the pump and get you to the higher discharge pressure. Then you will automatically get the higher power density, up to the limit of what the core can deliver.

There is another way that you can upgrade: composite fuel (see Figure 5). Instead of going on up in chamber pressure, you might choose to open up the throat area.

That will get you to a higher thrust about the same pressure at the pump outlet. The pressure drops across the nozzles coolent passages and the reflector, but it does result in an increased delta P across the core. For an assumed flow rate of 108 pounds/second, we had the difference between chamber pressure in this case of 1,333 over a 1,000, as compared to the reactor inlet pressure of 1,506 over 1,231. The difference here you can see is 231 psi, the difference here is about 170. So there is that effect.

And if you try to get your higher thrust and for the same core get a higher thrust to weight ratio by opening up the throat area, you are going to increase the pressure drop across the fuel elements.

Now, let's talk a little bit about nozzles (Figure 6). In this particular case we are looking at a 75 K engine and we are using composite fuel element, 4,860 degrees R in the thrust chamber and the Isp of 918 seconds, and 1,000 pound chamber pressure.

Well, as I indicated to you yesterday, if I maximize Isp, and I want to get the maximum in terms of expansion process, I get a very long nozzle. In this case 14 feet by 26 feet. This configuration creates concerns about where you store this thing and so one possible way to do that is to embrace extendable nozzles. The other way is to invert the nozzle skirt, but then you have the problem of getting the astronauts out there to bolt everything together.

But as you see, this package is fairly long. You might want to make a double truncation of the nozzle as a way of making the packages smaller. You could also consider going to expansion deflection nozzles or a torroidal nozzle. You can really pull down this size with such a technique, but that's adding the complication of looking at a more advanced configuration.

Another factor is that a portion of the diverging section of the nozzle now is going to see neutrons coming out of the core. They are going to be scattered and there is going to be some contribution of this projected source area of neutrons that have to be contended with relative to interaction with the hydrogen in the tank, and producing secondary gammas. If you are trying to cover yourself on that one, you might have to extend the shield to a larger diameter to be able to effect that.

A VOICE: Do you have a feel for what kind of neutron flux would be there without the shield?

MR. GUNN: I don't have that today, no.

However, one of the Russian scientists I talked to in May suggested a way around this problem, which was to go to multiple nozzles: nest together a group of short nozzles rather than one big one.

Then again you get into not only the throat heat load problem, but you also get yourself into a situation where you have more drag and more boundary layer to contend with and that's going to be an Isp loss. So these are factors to worry about in the design of the nozzle and this again points out the need to understand the payoff, as far as the vehicle contractor is concerned, on the package size.

A VOICE: The nozzle would seem to be significantly shorter, if you went to multiple nozzles.

MR. GUNN: That's right, and for some applications, that would be a neat way to do it.

A VOICE: If the nozzle is going to weigh more, you aren't going to get the full benefit.

MR. GUNN: That's part of it. I tried to see how much net benefit comes out. You have to go through that and find out how far do you want to push that. If you have carbon carbon or carbon composite, a light weight structure, you might elect to go farther than you would on rhenium or something like that. You would also be limited by manufacturing facilities.

Now yesterday I talked a little bit about dual turbo pumps and you asked the question. I said, yes, it was done and you were with us out in Nevada.

A VOICE: I know NERVA did. I said any real rocket.

MR. GUNN: We thought that was a real rocket.

A VOICE: It never flew.

MR. GUNN: It didn't fly.

A VOICE: You are telling me there was never, that there has never been a chemical rocket with multiple turbo pumps?

MR. GUNN: I am not sure, because the Russians have been pushing for multiple turbo pumps in some of their approaches and I am not sure where they stand.

A VOICE: My understanding really comes back to the business of multiple engines that I would argue strongly that multiple engines are better than multiple components on one engine, but that's another issue. That's really what I was asking.

MR. GUNN: I thought, had you ever done this, and the answer is, as far as pumps running together and working harmoniously, and designing a system that you could have a pump failure and you keep right on going, that was done.

Now, I have shown two. Is that the limit? No, it is not. We are looking at three, and if you put three in these things you can put a nice balanced thrust structure and fuel delivery lines and you can run a full thrust if you had a failure of one of the units. It's like on an airplane with multiple jet engines on a transport that we go across the country in. How many do we want to put on: two, three, four, eight? In some cases it is two.

A VOICE: They don't have multiple fuel pumps.

MR. GUNN: No, but I am trying to make a point; multiplicity. I think you are raising a good point, but I think one could argue that if you examine all of the components in the engine and ask yourself what gives you the most grief or gives us the most concern, I am going to say and surprise the people by saying I don't think it is the reactor. I think the reactor can be designed very robust, and forgiving to a certain extent, because we saw that in Nevada. We saw malfunctions occur. There can be some degradation, some erosion, some cracking of the fuel if it is a solid and still it will meet its job. But the turbo pump failing is catastrophic.

A VOICE: In the aircraft industry with engine out capability, those engines are designed for 30,000 hours of operation. We are talking about a maximum of ten hours. Now, with that sort of a situation, there can be a lot built into the design because you are really down very, very low on what I would call the life requirement.

MR. GUNN: Against what components are you looking at for failure?

A VOICE: I am looking at only the turbo pumps, the turbo compressor?

MR. GUNN: If you look at a blade on a turbine and look at the vibrations it can undergo on your Goodman Diagram curves, you find within minutes you can get yourself way out on the curve because you have such high vibration rates.

A VOICE: That has to be worked out in the design but in the airport industry we are talking about 30,000 hours of operation on those engines.

A VOICE: In the rocket industry, when we have had failures, the bottom line has been the support systems which can cause the catastrophic failure and not the engine. The RL-10 is probably the best engine ever built.

A VOICE: What I am trying to get away from if at all possible would be the concept of dual turbo pumps because of the short time of operation. I recognize the vibration problem.

A VOICE: Part of the problem in solid and liquid rocket engine development and with the shuttle is that they are technologies; the engine is too full. By doing a full systems test, we would have probably done a lot better. In other words, try before you fly.

With the dual turbo jump concept, I really question what we are gaining because we do have the added components and the complexity.

MR. GUNN: You are into the question of redundancy versus complexity. It turns out that the weight of the turbo pumps is a very small fraction of the total thing. You go to dual pumps, each one a little smaller, so it becomes lighter, but the two are heavier.

MR. HANNUM: We keep running down to Johnson to ask what does it mean to be man-rated: we keep asking them and the answer always comes back rather vague. But there are two points that they make consistently. One is that astronauts like redundant systems. You could argue that redundancy by virtue of it being there reduces liability. And it sometimes does. Redundancy sometimes does reduce liability.

Now, all the things that you all are saying about redundant turbo pumps and the pain and agony that goes with them is all very true. What we need to do is make the trades that Stan is arguing about and be prepared then to ask what does it do to reliability to have these? Redundancy is considered as "goodness" until you can prove it otherwise.

MR. GUNN: It's possible for us to design an engine system that requires no shielding to protect itself against its own created environment. The driver on that is to get the weight on the engine down and get the thrust to weight up because some of the shield's weight are not trivial.

Now, it still may be that you need to have the shield located in the engine area and maybe it's within the dome. Maybe it is above the dome. It depends on what temperatures they can stand relative to shielding against the neutrons and interacting with secondary gammas. In any case, I contend that it is possible to engineer every piece of equipment in this engine so it can take the full flux of reactor radiation and keep right on going.

I am going to point toward a system that doesn't use electronics. If it is necessary for redundant controls to go a separate system that is electrical and have them still work in a way that they could operate successfully, then you either have to shield the sensitive electrical parts or move them somewhere else to get them out of the radiation field.

Let me just address this first issue, which is pumping hydrogen. It is traditional that you have to have positive NPSH, and in dealing with liquid oxygen and some of the other propellants, that's true. Hydrogen is unique; you can pump it in a boiling phase even ingest up to 30 percent of the volume being received as vapor, and still pump alright (Figure 7).

When you trade that capability through the tank pressure, etc., you can convince yourself it is possible to pump saturated fluid in a tank at full thrust conditions, provided you get up above a certain minimum level in tank pressure.

I am not prepared to say that I am taking the tank pressure down to five psi, but I think anywhere from certainly 20 and above psi you can do that. And here again we need to have the people responsible for the tank design for the mission to tell us what working pressure they are going to go to. Then we can see if we can do this.

A VOICE: I was saying that one of the important things in this would be to work with the triple point hydrogen to get the better impulse density out of it and take all of the advantages.

MR. GUNN: I showed you earlier this control system (Figure 8) that's insensitive to the radiation environment. The fundamental parameters we need are pressure and temperature. We can get that. From that we can get the weight flow and the temperature. We know then what the reactor is doing, relative to its scheduled delivery gas temperature. Then we can operate on a schedule of thrust buildup, holding for thrust and thrust decay and meeting the mission requirements.

Shown also is a flux sensor. I think we need that I think for two reasons. One is that when we start the buildup, we start evolving from a very low power level until we start to see some significant power. You need to understand where you are and how fast that rate of power increase is occurring.

Then after the firing is over and you have shut off the propellant valve, the core now starts to heat up. You need to know when to introduce propellant flow again to pulse cool if that's the mode you are going after.

And one of the things that could give you that is the flux sensor. Perhaps the parameter I will show you next is the better way to do that. The question is what part of the core will tell you that other parts of the core are getting up close to the limit you want to see it operated at?

I make the contention that I am making a primary measurement of temperature and the neutron flux. I am going to use that measurement as a primary input, along with my pressure measurement, to determine everything else I need to do in that system.

Once you have made the shutoff and you have closed the propellant valve, you follow a curve on the decay power. That power is going to cause the core (after you have undercooled or overcooled it) to creep up again in temperature. Then you have to try to extract maximum Isp from the coolant gas, if that's what you want to do.

There is a neutron flux sensor that was conceived and worked on that back in the 1960s.

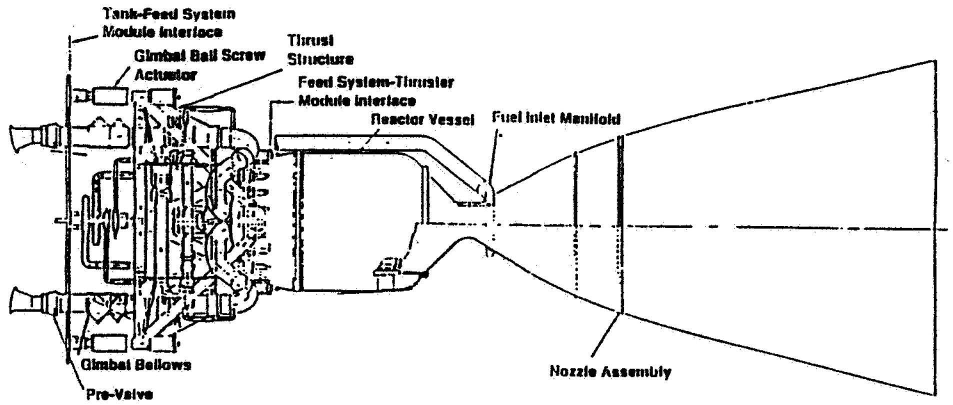
It is a temperature sensor that senses the output of the gas coming down out of the core directly and gives a pneumatic signal that's proportional to the temperature. We were also looking at pyrometors to get in and be able to measure surface temperature and maybe the core cylinder that needs to be monitored to say whether the internal core is getting too hot. We also did a lot of work on radiation resistant valves and actually deployed some of those in the NRX test series.

One last area that might deserve attention is an alternate way to get rid of this decay heat; that is a core cooling system. This was an old concept back in the 1960's. It was basically trying to tie into the heat removal capability of the tie tubes. With a closed cycle system, involving a turbo compressor and radiator, we could take the problem of having to use propellant to be able to remove the decay heat, and convert it into a means of radiating heat to outer space. We could thereby save the propellant for use in the later burns in the mission. It trades off that advantage with complexity because here is an added system of added weight. I am not sure that that's a smart thing to do, but it is a possibility.

Much of the improvements I have talked about here are a way of trying to get better specifications for the engine: that is Isp and engine thrust-to-weight ratio.

You have to be specific on what part you are talking about on this thing. Some of the things we know enough about so that if you retrieve the information, you can just go ahead and do it. A lot of other things are going to take development.

100K FLIGHT THRUST MODULE/FEED SYSTEM MODULE ASSEMBLY



Is (ε =500:1, UNCOOLED SKIRT) 922 SEC

WEIGHT..18,000 LB

FLOW RATE.................................108 LB/SEC

ε (COOLED SKIRT)...........150:1

POWER DENSITY.............85 MW/FT^3

ENGINE CYCLE...............EXPANDER

Figure 1

100K NTR, Expander Cycle, Dual T/P

Tie Tube Turbine Power

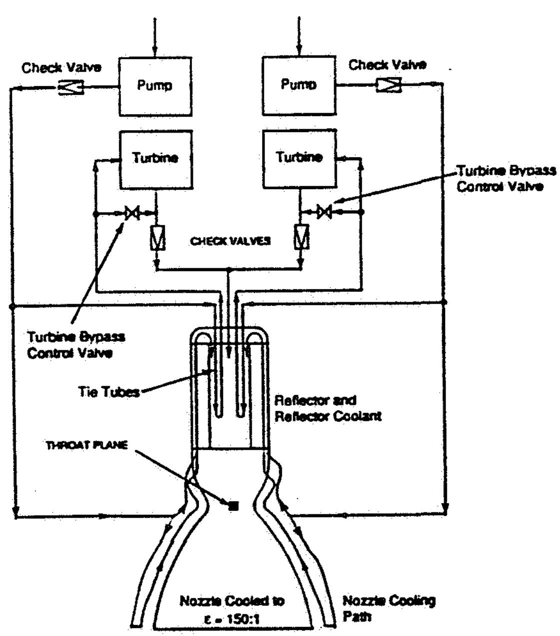


Figure 2

Turbopump Operating Conditions
75 Klbf NTR, Ae/At = 500

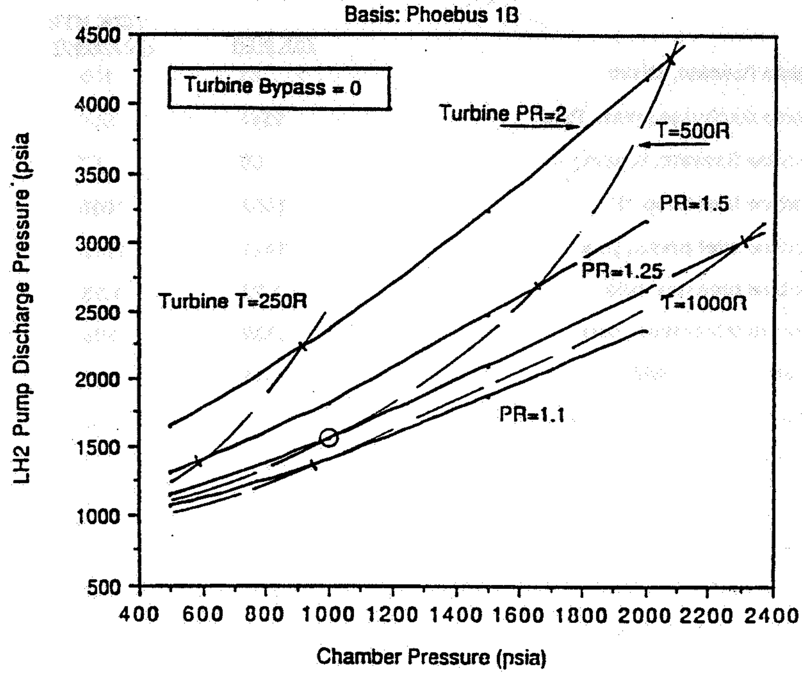


Figure 3

Typical System Parameters

	Phoebus 1B 75K Thrust Test	75K NTR Composite Fuel Elements
Pump flowrate, lb/sec	110	81
Pump disch press., psia	1381	1544
Turbine flowrate, % pump	11 (Cold Bleed)	50 (Expander)
Turbine inlet temp, °R	600	1000
Turbine inlet press., psia	531	1412
Turbine pressure ratio	10.4	1.25
Reactor inlet pressure, psia	938	1130
Reactor power, MW	1455	1645
Reactor core flowrate, lb/sec	94.4	81
Nozzle chamber temp, °R	4039	4860
Nozzle chamber presure, psia	735	1000
Nozzle exit dia., ft.	--	13.6
Nozzle expansion area ratio	12	500
Specific impulse, vac, sec	--	923

Figure 4

Growth Capability
Composite Fuel Elements

	75K NTR	100K NTR (Uprated)	100K NTR (New Nozzle)
Pump flowrate, lb/sec	81	108	=
Pump discharge press., Psia	1544	2059	1755
Turbine flowrate, % pump	50	67	43
Turbine inlet temp, °R	1000	1000	=
Turbine inlet press., psia	1412	1883	1539
Turbine pressure ratio	1.25	1.25	=
Reactor inlet press., psia	1130	1506	1231
Reactor power, MW	1645	2194	=
Reactor core flowrate, lb/sec	81	108	108
Nozzle Chamber temp, °R	4860	4860	4860
Nozzle chamber press., psia	1000	1333	1000
Nozzle exit diameter, Ft	13.6	13.6	15.7
Nozzle Expansion ratio	500	500	500
Specific impulse - vac, sec	923	922	923

Figure 5

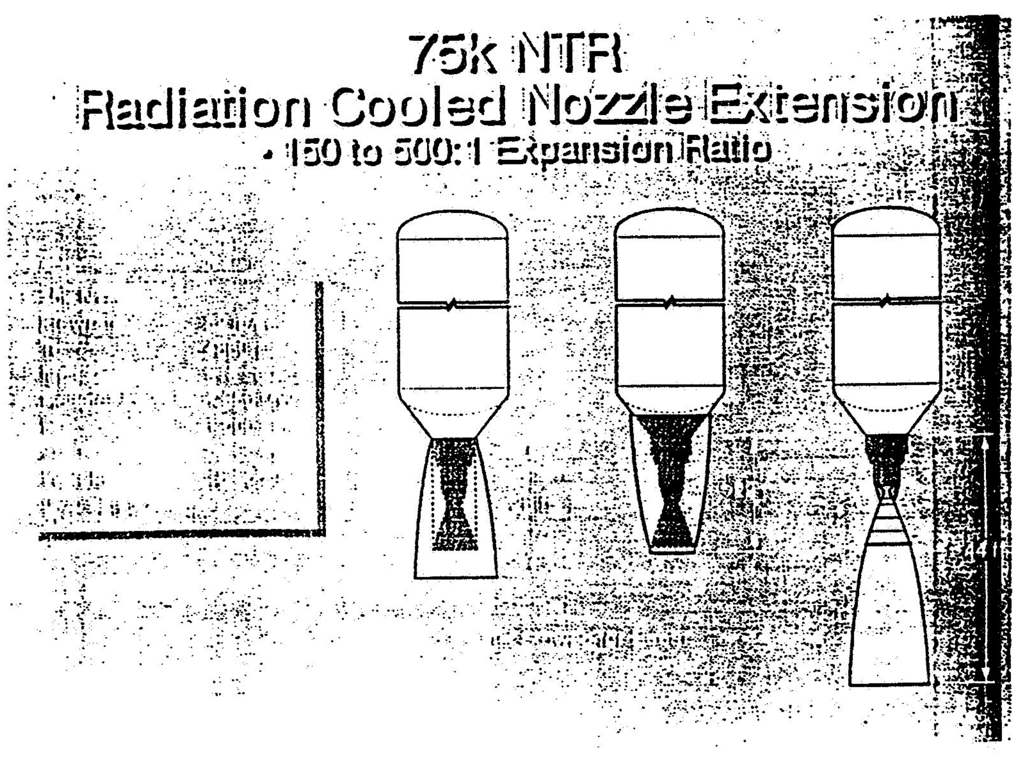


Figure 6

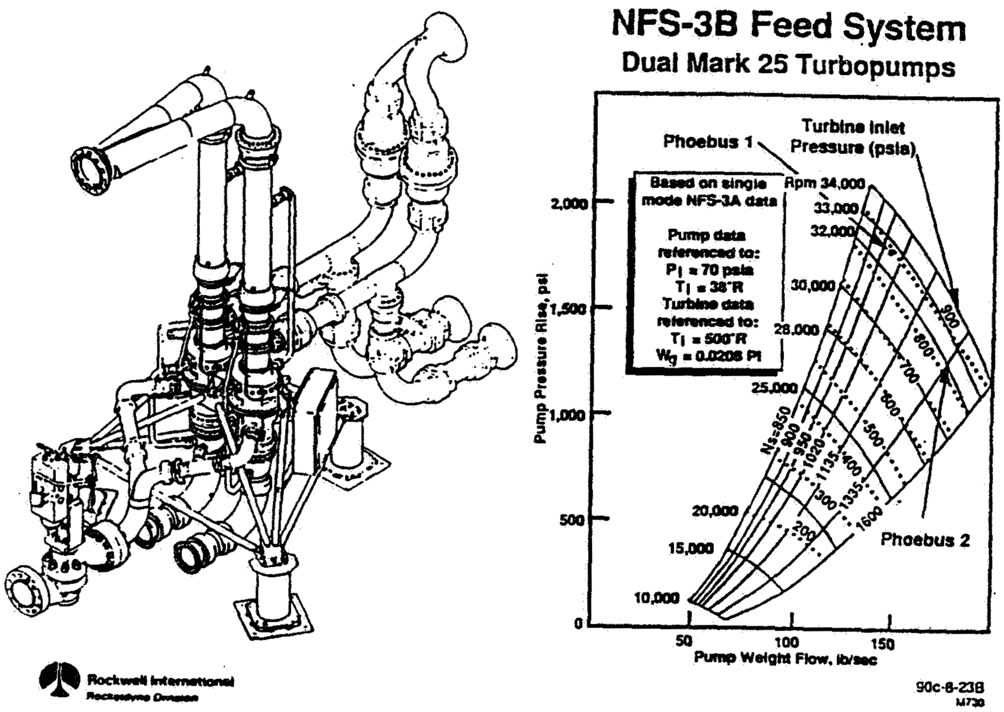


Figure 7

NTR Integrated Pneumatic-Fluidics Control System

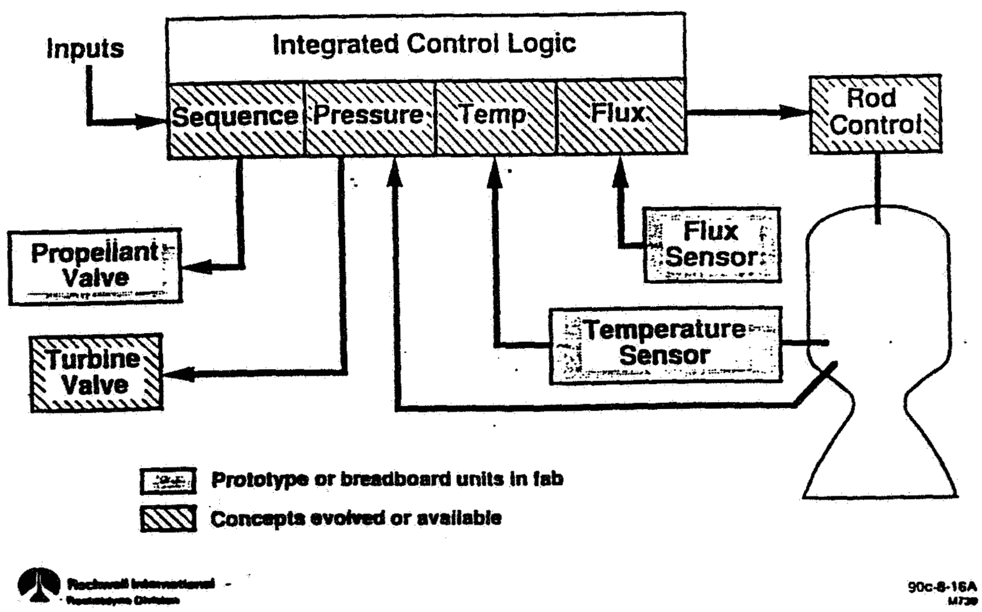


Figure 8

NUCLEAR FUELS STATUS

Michael Kania
Oak Ridge National Laboratory

This morning I am going to talk to you about coated particle fuel performance from a modular High Temperature Gas Reactor (HTGR). The experimental results I am going to talk about came from the Oak Ridge National Laboratory, in cooperation with some of our foreign partners. The talk is directed to the results from the HTGR program, the commercial program and the HGR programs, so the temperature range is much lower.

First of all, (see Figure 1), I would like to speak about the fuel particle concept: The functional requirements, the performance limiting mechanisms and the temperature range we are looking at. All of these give you an up-to-date view of what our fuel performance is at normal operating conditions (when temperatures are less than 1250 C), the results we can expect at the accident conditions (testing temperatures greater than 1250 C, up to 2500 C), and techniques for performance characterization.

The HTGR, or the gas cooled program, fuel particle provides two specific functions (see Figure 2). One is a source of fissile material. The other is the primary containment system for fission products. The fuel source is either a dense oxide or carbide, an oxicarbide, uranium, thorium, plutonium, or a mix of two. We looked at the actual oxicarbon compound, as well as a solid state mixture.

The containment is a dense ceramic coating formed by Chemical Vapor Deposition (CVD) and deposition in fluidized bed coatings (see Figure 3). It has two primary coatings of BISO (see Figure 2). In our program,.we looked at two pyrocarbon coatings. We also looked at some silicon pyrocarbon mixtures with the outer layer. The reference design is a TRISO coating (Figure 2), four layer design: two pyrocarbons followed by silicon carbide, in some cases, and an outer carbon-carbon layer.

Figure 3 gives a quick view of what I am talking about here. It is a Scanning Electron Microscope (SEM) photo of a particle we purposely broke. The fuel kernel is surrounded by four layers; the buffer layer, the inner PyC silicon barrier and an outer PyC.

In order to crack the particle, we used a small micrometer device. It was an intentional break of an underradiated fuel. We wanted to look at the coatings.

For the modular HTGR, heavy reliance is placed on the coated particle as a containment concept to prevent fission product release (see Figure 4). Fission products are kept at the site of their origin. Particle/containment performance can be continually monitored in the reactor. The high quality fuel is a requirement. On the average we require a quality level of 6 equivalent failed particle, per 100,000. We distribute the containment system over the entire population of the fuel, rather than a few barriers.

I indicated earlier that the UCO kernel is our reference: a solid gel mixture derived of UO_2 (see Figure 5). It has an enrichment of 20 percent. Normal operating conditions are a temperature range from 750 to 1250 degrees C. I think anything beyond that is considered accident temperatures. Burnup is 26 percent FIMA. Fast fluence is less than 5, and we are talking about fairly low power levels. We are talking about 150 milliwatts for these designs. We have testing at higher power levels, and higher orders of magnitude show that UCO fuel is superior to UO_2 and UC_2 under similar operating conditions. That's why it was selected as a reference.

After a number of years of in-reactor testing, we have identified the basic or dominant fuel performance limiting mechanisms for our fuel (see Figure 6). They are pressure vessel failures; meaning the internal pressures exceed the strength. The silicon carbide layer had massive failures, and we had a lot of fission product release. We also had silicon carbide coating failure. This did not necessarily cause massive releases but it did contribute to synergistic effects.

The dominant mechanics of kernel migration and carbon transport, in the presence of a thermal gradient, results ultimately in kernel/silicon carbide contact and layer degradation. In this mechanism, fission products migrate through the silicon carbide layer and interact. Again, this results in layer degradation. Consequently, pressure vessel is not of standard requirement.

Thermal dissociation is the decomposition of the silicon-carbide layer resulting in loss of coating integrity. It is active above 1600 degrees C for various periods of time. Finally, we have fast neutron damage, causing differential expansion/contraction of the pyrocarbon layer, cracks in the layer and a complete loss of coating integrity.

The thermal migration data has been around for a number of years. We have looked at different fuel kernel designs with respect to the envelope that we could allow. We found with HTGR that if the UO_2 fuels fall within the envelope at temperatures 1200-1300 degrees C, we can not use that design for HTGR applications (see Figure 6). UC_2 falls somewhat below but fairly close to oxide. Basically, what we ultimately came up with in a kernel design was the UC-O concept. This concept eliminates the thermal migration problem as well as the fission product release problem from the UC_2 fuels.

Data about thermal decomposition was attained in the accident testing program. At 2500 degrees C, the silicon carbide coating disappears. It's primarily a carbon, grain coating. Interestingly enough, this particle has not failed catastrophically. Instead it has expanded. The coating is still visually intact; fission products have been lost to a fairly large degree. There was a burn up of about 3 percent and loss of about 25 percent of the cesium. Fission gasses will be lost at somewhat less than the 25 percent level.

We demonstrated in the German program that we can fabricate fuel with very high levels of quality. They produced hundreds of kilograms of material using production scale facilities. The U.S. did it with prototypic modular scale facilities.

In-reactor fuel failure levels have been demonstrated less than 10^{-4}. In fact, the level is about $3x10^{-5}$ with very high confidence, and that is based on fission gas release data. Temperatures go up to 1200 degrees C, 12 percent FIMA and relatively high fast fluence.

Accident conditions for the German program are temperatures that temperatures range between 1250 and 2500 degrees C. The U.S. program uses any condition that causes the fuel to be less than 1600 degrees C (see Figure 8).

In accident simulation test, at 1600 degrees C for periods up to 500 hours, no significant fission product release was observed. This is based primarily upon the German data. From that we can show, with very high confidence, the induced failure levels of 10_{-5} range. At 1800 degrees C and above, we do find some significant amounts of metallic fission products being released after short periods (Short period are in hours not minutes). In a ramp test where we took fuel to 2500 degree C in 50 hours, no detectable fission products were released.

Figure 9 shows ramp heating data. You can see a plot of the fraction release krypton 58 as a function of heating time. At 1600 degrees C we see that these levels are 10^{-6} level compared to what near 10_{4} would be for a single particle failure. 1800 degrees C we see that after periods of some 50 hours or so we start seeing degradation of the fuel. Basically this is a diffusion of fission products through the silicon carbide. At 2100 degrees C you can see failure rapidly occurs.

The ramp test in Figure 10 with German data shows that you are not seeing serious silicon carbide degradation until you get to 2100 degrees C. Then you get a fairly rapid rise. This data has been used quite a few times. I think it is wrong to conclude that you can run in at 1900 degrees C or 2000 degrees for a long time. That is not true. These are only ramp test data. They need to be compared to the isothermal data, at least for our concept.

I am trying to put together a comparison of the performance attributes from the HTGR to NTP for what I knew prior to coming to this meeting (see Figure 11). I could change that quite a bit after yesterday, so it's good education for me. We have an UCO in the U.S. and UO_2 in Germany. Our coating is basically a silicon carbide TRISO design. NTP looks at zirconium carbide fuel form. The HTGR concepts is that of a machine graphite prismatic block or a sphere.

Enrichments may be the same. For the civilian program, we have about 20 percent. For the NPR program, we have fully enriched material. The Germans have 8 to 10 percent, and they have a large database on fully enriched material. This appears appropriate for NTP.

The big difference is the power produced per particle. The HTGR is a low power per particle. The power density in the core is very low. We are looking at 100 to 150 milliwatts in NTP, a watt or two per particle, maybe even beyond burnup. NTP may be, with respect to an open cycle, a tenth of a percent as indicated.

With respect to fuel quality, we demand a very high quality. I think that someone is going to say the same thing with respect to this application here. I heard things about dumping fission products out the back but that's not something to be decided by me.

Let me finalize this with what is available to us for characterization of fuel performance and fission products (see Figure 12). We have a full range of testing irradiation available to us. We also have hyper or thermal spectrum. We can achieve up to 5 watts of power per particle while maintaining in-reactor surveillance. With a full range of Post Irradiation Evaluation (PIE) capabilities, we can look at the physical metal and fission gas retention on a particle basis. We have high temperature PIE and furnaces that will go up to the HTGR program's 2000 degrees C limit. The furnaces are probably about 2800 degrees C. And we have capabilities for modeling fuel particle behavior and fission product transfer.

Outline

* *Particle Fuel Concept*
* *Functional Requirements*
* *Performance Limiting Mechanisms*
* *Fuel Performance*
  - *Normal Operation [<1250∘C]*
  - *Accident Conditions [>1250∘C]*
* *Methods/Techniques for Characterizing Performance*

Figure 1

THE COATED PARTICLE FUEL CONCEPT PROVIDES TWO BASIC FUNCTIONS: (1) SOURCE OF FISSILE MATERIAL; AND (2) PRIMARY CONTAINMENT FOR FISSION PRODUCTS

Fuel Source:	A dense oxide, carbide, or oxi-carbide spherical kernel of uranium, thorium, plutonium, or a mixture
Examples -	ThO_2, UO_2, $(Th,U)O_2$, $(U, Pu)O_2$ ThC_2, UC, UC_2, $(Th,U)C_2$ UC_xO_y, UO_2+UC_2 mixture
Containment:	Dense ceramic coatings surrounding spherical fuel kernel formed in succession by chemical vapor deposition (CVD)
Examples -	BISO Coating, two layer design [PyC/PyC] TRISO COating, four layer design [PyC/PyC/SiC/PyC]

Figure 2

BASIC FUEL UNIT OF THE MHTGR IS THE COATED PARTICLE

SOURCE OF FISSILE FUEL MATERIAL
PRIMARY FISSION PRODUCT CONTAINMENT

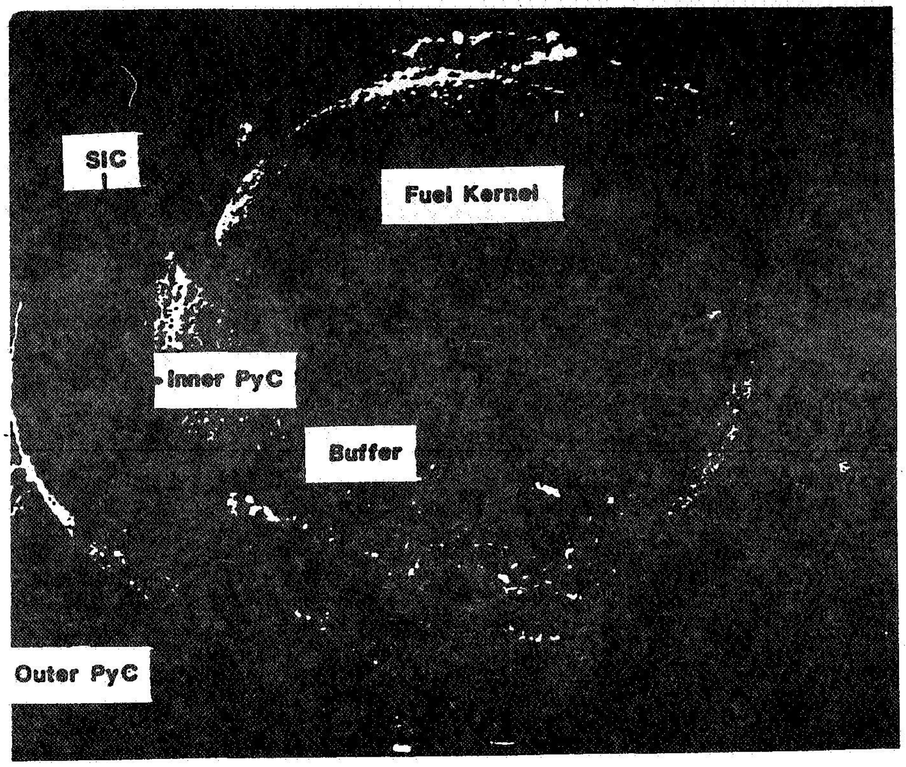


Figure 3

FOR THE MODULAR HTGR HEAVY RELIANCE IS PLACED ON THE COATED PARTICLE CONTAINMENT CONCEPT TO PREVENT FISSION PRODUCT RELEASE

- o Fission Products are kept at the site of their origin under normal and off-normal events.
- o Particle/containment performance can be continually monitored in-reactor by measuring primary circuit activity.
- o High Quality fuel fabrication by requirement - on average, quality level of < 6 equivalent failed particle per 100,000.
- o Containment system distributed over 10^{10} microspheres, rather than depending upon only a few barriers.

Figure 4

TRISO-COATED UCO SELECTED AS REFERENCE FISSILE PARTICLE FOR MODULAR HTGR BASED ON ITS FABRICABILITY AND FISSION PRODUCT RETENTION CAPABILITIES

- o UCO Kernel is a Sol-Gel mixture of UO_2 (80%) and UC_2 (20%) with a 20% enrichment.

- o Normal Operating Conditions:

Temperature (°C)	750 - 1250
Burnup (% FIMA)	≤ 26
Fast Fluence (10^{25} n/m^2)	≤ 5
Power/particle (mW)	~ 150

- o Performance superior to UO_2 and UC_2 under similar operating conditions.

Figure 5

EXTENSIVE IN-REACTOR TESTING AND POSTIRRADIATION EXAMINATION TESTS HAVE IDENTIFIED DOMINATE FUEL PARTICLE PERFORMANCE LIMITING MECHANISMS

1. Pressure Vessel Failure - SiC tensile stress induced by internal gas pressure exceeds layer strength resulting in total coating failure and massive FP release

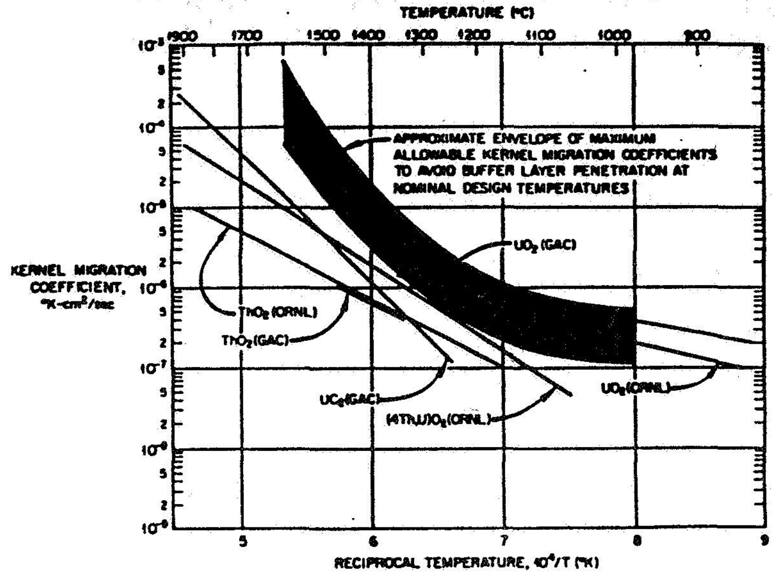
2. SiC Coating Failure [Contributes to synergistic effects]
   - o Kernel Migration - carbon transport in presence of thermal gradient results ultimately in kernel/SiC contact and layer degradation,
   - o FP Interaction - FPs released from kernel diffuse to SiC, chemically interact resulting in layer degradation,
   - o Thermal Disassociation - decomposition of SiC layer resulting in loss of coating integrity, active above 1600•C.

3. Fast Neutron Damage - differential expansion/contraction of pyrocarbon layer resulting in loss of coating integrity.

Figure 6

ORNL-DWG 74-

COMPARISON OF THERMAL STABILITY OF CANDIDATE FUELS FOR HEU PRISMATIC HTGR SHOWS UNACCEPTABLE, MARGINAL AND ACCEPTABLE PERFORMERS



Average Kernel Migration Coefficients for $(4Th,U)O_2$, UC_2, UO_2 and ThO_2

SIMILAR COMPARISONS NEEDED FOR LEU AND MEU FUELED HTGR

Figure 7

COATED PARTICLE FUEL PERFORMANCE

(Modular HTGR)

Accident Conditions [Temperatures > 1250•C]

o Modular HTGR design limits maximum fuel temperatures to < 1600•C under all conditions.

o In accident simulation tests at 1600•C for periods up to 500 h, no significant FP release was observed.

o For accident simulation tests at 1800•C and above, significant FP release (gaseous/metallic) observed after short periods.

o Ramp tests of 50 h duration to 2500•C, exhibited no detectable FP release beyond HTGR peak accident conditions.

Figure 8

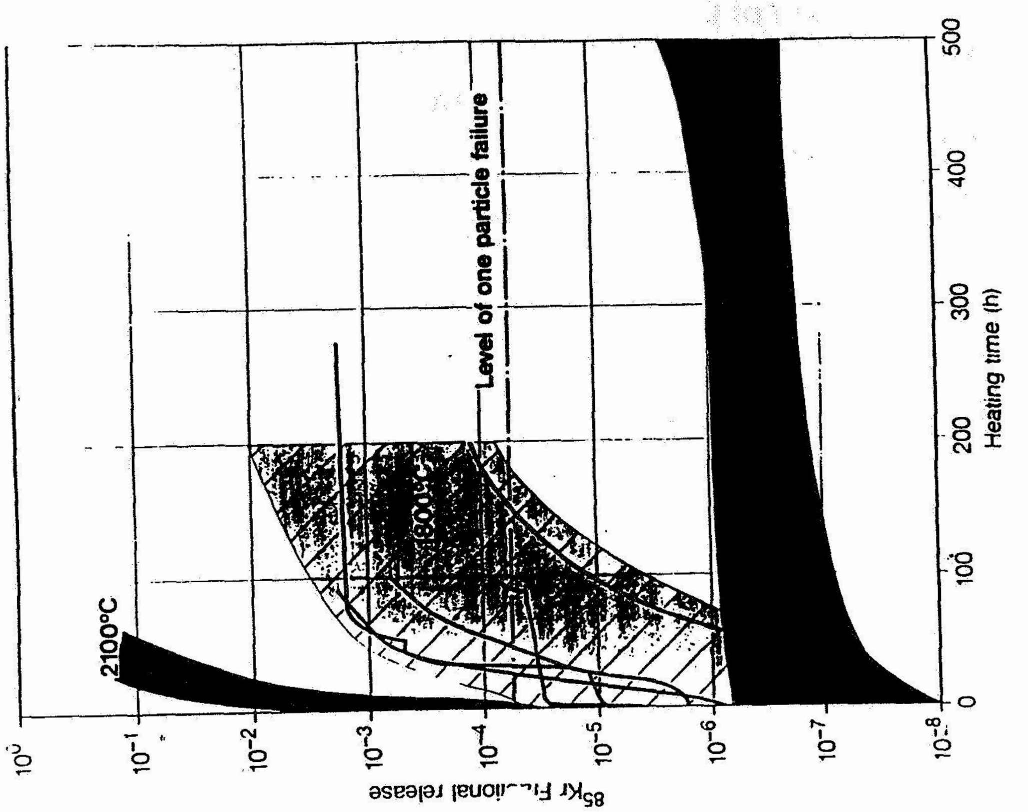


Figure 9

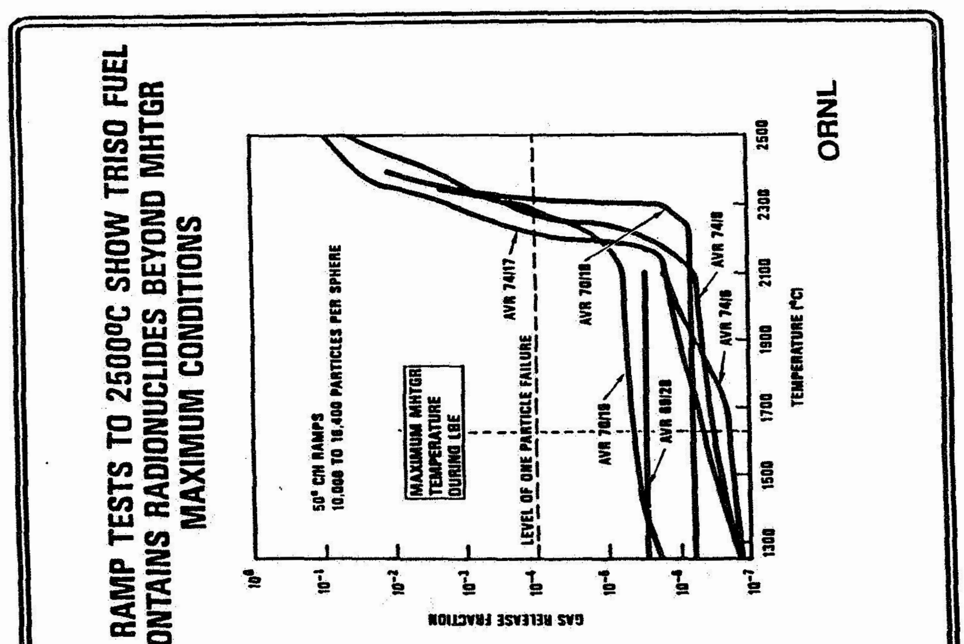


Figure 10

Comparison of Coated Particle Performance Attributes
(HTGR and NTP Concepts)

Attribute	Modular HTGR		NTP
	US	FRG	
Particle Design			
- Kernel	UCO	UO2	[UC,UC2,?]
- Coating	SiC TRISO	SiC TRISO	[ZrC TRISO]
Fuel Form	Prisimatic	Sphere	[Unbonded or Compacts]
Enrichment (% U-235) *	< 20 / >90	8 to 10	[> 90]
Power/Particle (mW)	< 150	< 100	[1000 to 2000]
Burnup [% FIMA] *	< 26 / < 75	< 12	[< 25]
Operating Temperature (°C)			
- Normal	750 to 1250	700 to 1200	[> 1250]
- Accident	< 1600	< 1600	[?]
Fuel Quality	< 6 E-5	< 6 E-5	[?]

* Civilian MHTGR / NPR-MHTGR

Figure 11

SPECIALIZED TECHNIQUES AND METHODS ARE AVAILABLE AT ORNL TO CHARACTERIZE COATED PARTICLE PERFORMANCE AND FISSION PRODUCT BEHAVIOR

o <u>Irradiation Testing</u>
- Accelerated environment
- Thermal Spectrum with Spectral Tailoring
- High Power, up to 5W/ particle
- In-Reactor Surveillance

o <u>Postirradiation Examination (PIE)</u>
- Metrology/Ceramography
- Fission Metal Retention/Particle
- Fission Gas Retention/Particle
- Electron Microscopy/Microprobe

o <u>High Temperature PIE</u>
- Remote Furnaces Temperatures up to 2000°C
- Quantitative FP Release Determination
- Post Test Characterization

o <u>Modeling/Documentation</u>
- Fuel Particle Behavior
- Fission Product Release/Transport
- Statistical Analysis
- Performance Assessments

Figure 12

N92-11114

NUCLEAR SAFETY

D. BUDEN

Nuclear safety concerns can be thought of in terms of terrestrial, unmanned space operations, manned space operations, and Moon and planetary bodies. These are overlapping in many respects; however, there are unique aspects associated with each area (Figure 1). For instance, for terrestrial operations, one must be concerned with the anti-nuclear bias and the strict laws that must be adhered to in order to protect the environment and people. For unmanned space operations, the main concerns are related to low orbits and final disposal. Manned operations add a new class of problems concerning the safety of the crew. For instance, if a nuclear propulsion unit fails on the way to Mars and the crew keeps going with no way to get home, this is not acceptable. Surface power supplies have their own unique features, but these are a subject for a different meeting.

When one discusses safety of nuclear power and propulsion, one observes overlapping and unique areas (Figure 2). Nuclear propulsion rockets have to deal with hydrogen exhausting out of a nozzle that could contain fission products or radioactive materials. Nuclear power systems need to be concerned with high burn up and fission products and actinides formed over long operating times.

It is highly desirable to have a set of generic space safety guidelines. However, such guidelines do not exist. One document on safety issued in the 1970's, OSNP-1, includes an overall safety philosophy that pretty well summarizes the U.S. safety philosophy. It states that the policy of the United States for all U.S. nuclear power sources in space is to ensure that the probability of release of radioactive material and the amounts released are such that an undue risk is not presented, considering the benefits of the mission (Figure 3). Each program, such as SP-100, includes its own version of safety requirements as part of the specifications.

General safety design requirements are given in Figure 4. In case of an accident, the reactor must be maintained subcritical if it is immersed in water or other fluids. Essentially, this relates to launch pad abort situations. Next, the reactor needs to have a significant effective negative power coefficient--unfortunately, what is meant by significant is not well defined. No credible launch accident may cause criticality relating to fires and explosions that could result in a critical reactor generating significant amounts of radiation. The reason for no reactor operation until a stable flight path is achieved is for ground personnel safety and safety during launch aborts. The reactor radiation levels are very low prior to normally planned operation in space. Flight qualification will probably include a zero power test to check the nuclear physics of the reactor, but the radiation levels will still be sufficiently low to avoid the need for special procedures around the reactor on the launch pad. Two independent shutdown systems will ensure that the reactor will shutdown when commanded. Independent decay heat removal paths are to avoid core meltdowns in case of a failure in the normal coolant path.

One important factor in preparing safety requirements is that each requirement should have an identifiable contribution to reducing safety related risk. The requirements should be generic and not specify design solutions. In other words, safety requirements should address safety issues and not particular design concepts.

Undue risk is another concern in arriving at safety requirements. There is no legal definition for this term. For some, one in a million would be considered an acceptable definition. Others would argue for some other number. Obviously, the consequences of an event enter into what we accept as undue risk. The fact that we can not quantitize the definition makes it difficult for many engineers in system design .

Terrestrial safety factors are given in Figure 5. Testing nuclear electric propulsion power plants will require at least three independent barriers to radioactive materials being released to the biosphere. Also, there will need to be an independent decay heat removal system in case the primary coolant loop fails. Additional safety controls and instrumentation will be needed to monitor ground test operations.

SP-100 flight system requirements are given in Figure 6. These are part of the SP-100 requirements document. However, the document tends to include design solutions as part of the specifications. Generic safety specifications are preferable. SP-100 provides a starting point for nuclear electric propulsion safety specifications preparation.

For manned systems (Figure 7), the safest response to an abnormal event may not be to shutdown. If a habitat power system going to or on Mars is shutdown, the crew could lose their life support equipment--not a very safe approach. We are going to have to think about how to continue operations, even at a somewhat reduced level. Reactor scram at times is an unacceptable safety action.

From past programs, we can look at lessons learned (Figure 8). Safety must start with the initiation of the design process! A systematic determination of the effects of all possible failures is needed right at the beginning of the design process. Countermeasures must be developed for significant accident situations. The cost and benefits of mitigation need to be assessed and appropriate remedies applied. Safety must be given more then lip service and must truly be given primary priority.

SP-100 has recently performed detailed safety studies through all phases: ground operations, launch, flight and disposal (Figures 9 and 10). The issues are similar to those that will need to be addressed in nuclear electric propulsion power plants. This has led to many design features (Figure 11 and 12), such as two independent shutdown systems, control rods in the core, a special in-core method of cooling the system in case primary coolant is lost, and a reentry cone around the reactor.

During ground operations (Figure 13), the key concerns are to prevent accidental criticality, avoid loss of special nuclear materials to terrorists, and ensure that radiation levels around the launch pad are sufficiently low to ensure that special precautions are not necessary for worker safety. The approaches for accomplishing safety, as given on the figure, are well known.

For launch operations (Figure 14), the key concerns are to prevent accidental nuclear criticality and to keep foreign countries from acquiring special nuclear material. For instance, if an abort occurred during launch operations, we do not want special nuclear material ending up in a foreign country and starting an international incident. Approaches exist as to how to address these concerns. Redundant neutron poisons can take care of preventing accidental criticality. In the NERVA program, we not only had the control drums, but also had wires in the core that would be extracted when the nuclear stage was separated. This provided independent redundant safety systems.

To ensure that an abort would lead to nuclear material being dispersed over water, on-board destruct devices are used. Early launch aborts will end up in the Atlantic Ocean. Later aborts have sufficient momentum to carry the satellite over an ocean where the destruct device can destroy the satellite.

In flight operations (Figure 15), the key concerns have to do with unplanned reentry into the biosphere and crew safety. Unplanned reentry can be reduced to very low probability levels by selecting the flight trajectory to always move towards a safer orbit. Interlocks can be used to shutdown the reactor if an unsafe condition is sensed. For crew safety, either redundant systems need to be supplied or means to continue to operate to bring the crew home. One must decide how much redundancy in engines and power plants are going to be required to get home safely. One concept is to use seven engines with a two engine out capability. This changes the thrust level and design complexity of the engine and drives the whole development program. This issue is important to resolve at the beginning of the systems engineering process.

Final disposal (Figure 16) must be considered to avoid reentry of the reactor into the biosphere or contamination of low Earth orbit. The approach is to avoid bringing it back to low Earth orbit when feasible and to select orbits to minimize risk. Returning from Mars, a nuclear thermal rocket can be disposed of in deep space with final capture of the crew capsule by aerocapture. This way, the nuclear thermal rocket can be disposed of so that it never passes in the vicinity of the Earth.

Perceived safety (Figures 17 and 18) is an interesting subject because the public's perception of safety is not the same as actual safety. Figure 17 shows the real safety of SP-100. It is significantly safer then a transcontinental aircraft flight, diagnostic medical services, radiation therapy or lifetime natural environments. As experienced in the nuclear industry, the real and perceived safety are often very different. The nuclear industry probably has the safest record of any major industry in this country, but if you ask the average person on the street, he probably thinks it is more dangerous than driving a car. Perceived safety is an emotional issue and emotional issues are hard to deal with. However, this is something that has to be addressed early in the program. Reducing the real risk to a very low level helps in reducing perceived safety risk.

Turning to licensing, the users must know that launch approval will be granted in a timely fashion (Figures 19 and 20). A procedure is in place to accomplish this. The Interagency Nuclear Safety Review Panel performs independent safety/risk evaluations, the agency flying a payload requests permission for flight, the Office of Science and Technology Policy (OSTP) reviews the request and makes the launch decision, the Executive Office of the President makes the final decision if OSTP feels that it is appropriate.

The NERVA program design philosophy is given in Figure 21. Safety was a driving force in the flight engine design. The NERVA flight engine program and safety plan are summarized in Figures 22 and 23. They included detailed safety analyses and experiments and a requirement to be able to continuously provide 30,000 lb thrust in an emergency mode.

In summary, potential solutions exist to reduce risk to acceptable levels. Unless safety is considered from design selection and initiation, the cost of safety goes up dramatically. Not only must the safety risk be reduced to acceptable levels, it must be done in a manner that the perceived risk to the decision makers and public is acceptably low. Licensing procedures are in place and the duration of the licensing process is predictable. Users can count on approval for launch if procedures are followed and operational constraints are similar to chemical systems.

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David Buden
Nuclear Safety

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SAFETY

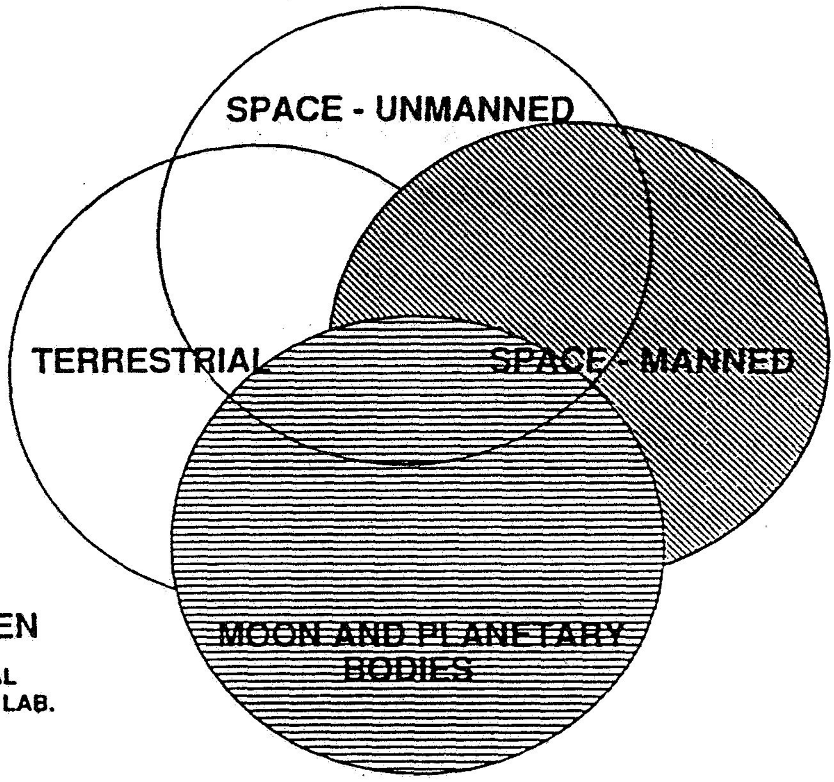

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Figure 1

SAFETY

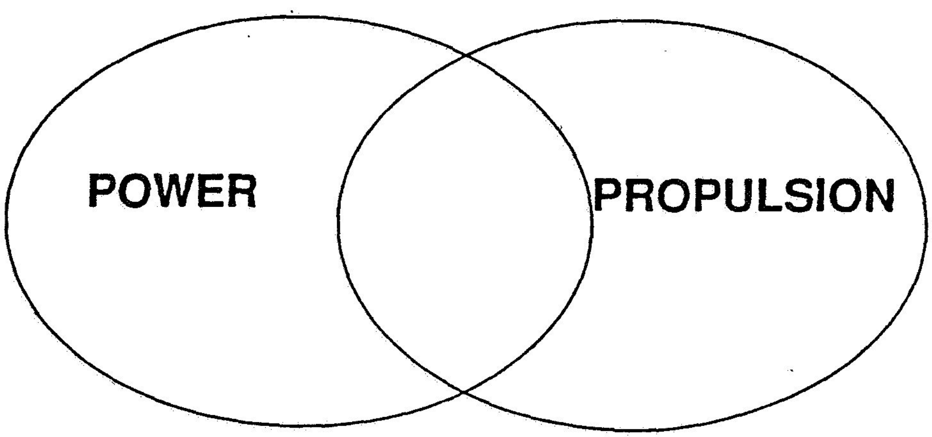


Figure 2

GENERAL SAFETY REQUIREMENTS

THE POLICY OF THE UNITED STATES FOR ALL U.S. NUCLEAR POWER SOURCES IN SPACE IS TO ENSURE THAT THE PROBABILITY OF RELEASE OF RADIOACTIVE MATERIAL AND THE AMOUNTS RELEASED ARE SUCH THAT AN UNDUE RISK IS NOT PRESENTED, CONSIDERING THE BENEFITS OF THE MISSION.

OSNP-1

Figure 3

SAFETY DESIGN REQUIREMENTS

- REACTOR DESIGNED TO REMAIN SUBCRITICAL IF IMMERSED IN WATER OR OTHER FLUIDS
- SIGNIFICANT EFFECTIVE NEGATIVE POWER COEFFICIENT OF REACTIVITY INCLUDED
- NO CREDIBLE LAUNCH ACCIDENT CAUSES CRTICALITY
- NO REACTOR OPERATION UNTIL STABLE FLIGHT PATH ACHIEVED
- TWO INDEPENDENT SHUTDOWN SYSTEMS
- INDEPENDENT DECAY HEAT REMOVAL PATH
- UNIRRADIATED FUEL POSE NO SIGNIFICANT ENVIRONMENTAL HAZARD

Figure 4

TERRESTRIAL SAFETY

- NUCLEAR ELECTRIC PROPULSION POWER PLANTS
  - THREE INDEPENDENT BARRIERS TO RADIOACTIVE MATERIAL RELEASE
  - INDEPENDENT DECAY HEAT REMOVAL SYSTEM
  - ADDITIONAL SAFETY CONTROLS AND INSTRUMENTATION
- NUCLEAR THERMAL ROCKETS
  - LOSS-OF-COOLANT FLOW SYSTEM
  - SCRUBBERS TO CLEAN EXHAUST OF RADIOACTIVE MATERIALS
  - CONTAINMENT/CONFINEMENT UNCERTAIN
  - ADDITIONAL SAFETY CONTROLS AND INSTRUMENTATION

Figure 5

SP-100 FLIGHT SYSTEM KEY SAFETY REQUIREMENTS

- MAINTAIN REACTOR SUBCRITICAL DURING ACCIDENTS AND DURING PERMANENT DISPOSAL
  - FUEL/SAFETY ROD ALIGNMENT
  - LAUNCH PAD FIRES
  - EXPLOS'ONS
  - CORE IMPACTION
- INTACT REENTRY FOR SPECIFIED INADVERTENT EVENTS
- ESSENTIALLY INTACT BURIAL FOLLOWING INADVERTENT REENTRY
- HIGH RELIABILITY FOR REACTOR SHUTDOWN
- HIGH RELIABILITY FOR SHUTDOWN HEAT REMOVAL
- RETENTION OF REACTOR STRUCTURAL INTEGRITY FOR LOSS-OF-COOLANT
- SECURE COMMUNICATIONS AND INHIBITS TO PREVENT REACTOR STARTUP PRIOR TO OPERATIONAL ORBIT
- MINIMIUM USE OF HAZARDS, CHEMICALLY TOXIC MATERIALS

ARE THE REQUIREMENTS THE SAME FOR NEP POWER PLANTS?

Figure 6

SPACE--MANNED

- CONTINUING TO OPERATE MAY BE SAFER THAN SHUTTING DOWN
- MONITORING ASSESSMENT INSTRUMENTATION

Figure 7

SAFETY APPROACH

- SYSTEMATICALLY DETERMINE THE EFFECTS OF ALL POSSIBLE FAILURES
- ADVISE COUNTERMEASURES TO PREVENT A NUCLEAR ACCIDENT
- ACCESS THE COST AND BENEFITS OF MITIGATION
- RECOMMEND APPROPRIATE REMEDIES

MUST START WITH INITIATION OF THE DESIGN PROCESS!

Figure 8

POTENTIAL MISSION ACCIDENTS AND HAZARDS

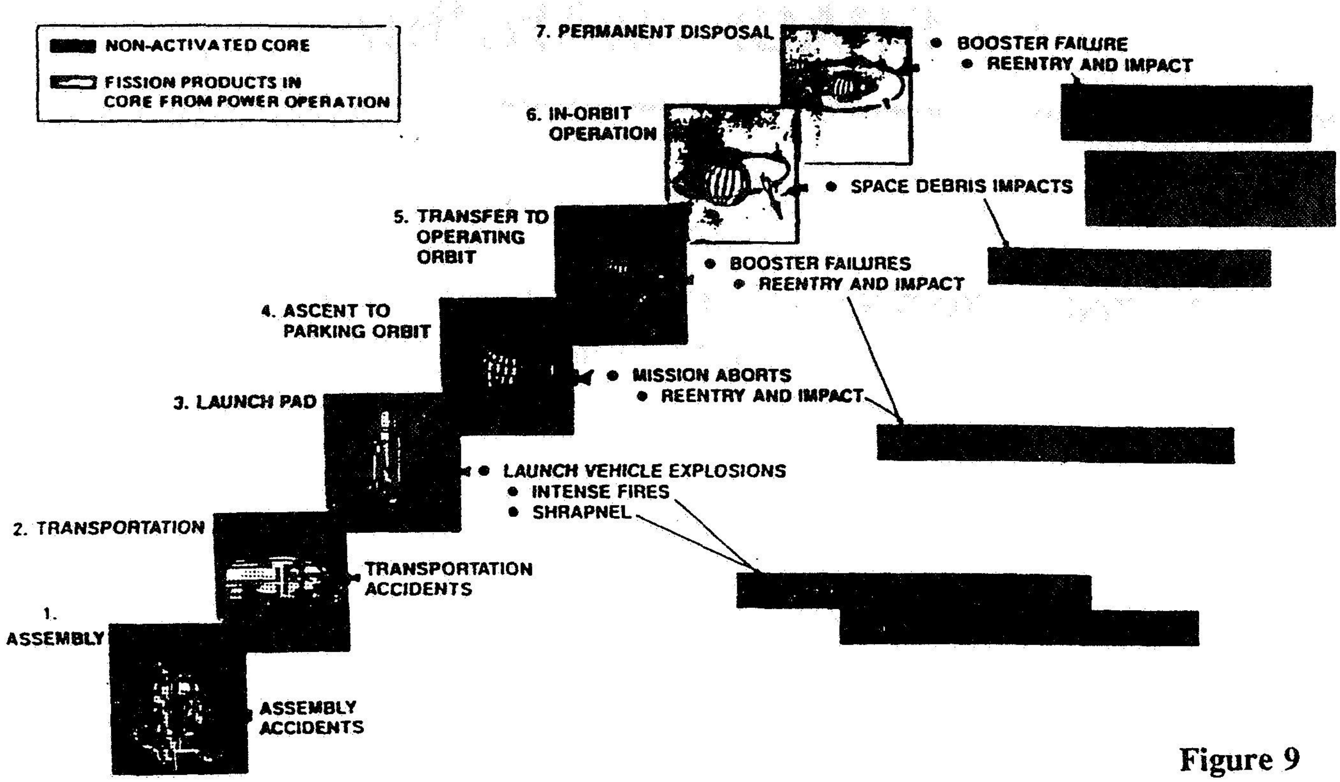


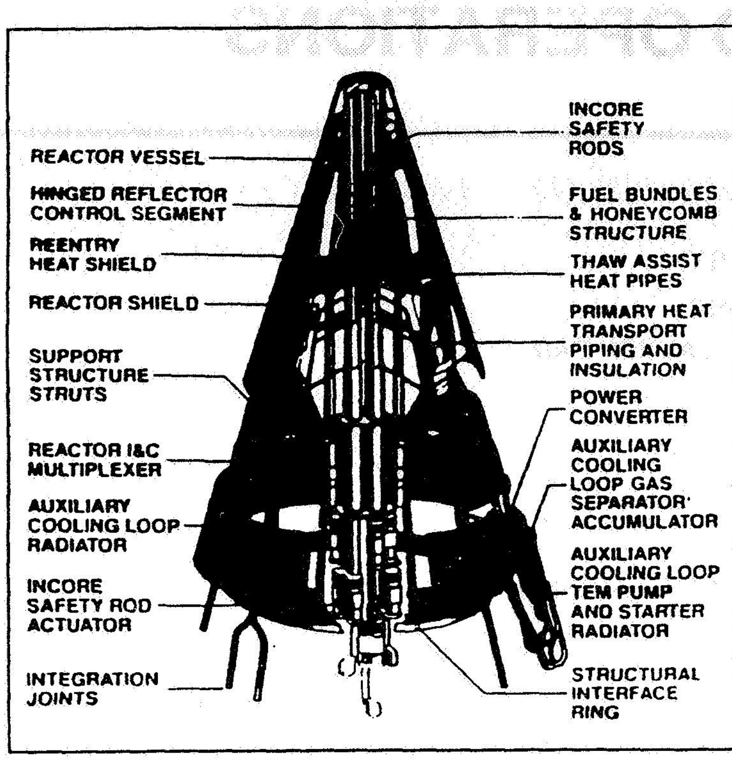
Figure 9

SAFETY CONCERNS

- GROUND
- LAUNCH
- FLIGHT
- DISPOSAL
- PERCEIVED
- LICENSING

Figure 10

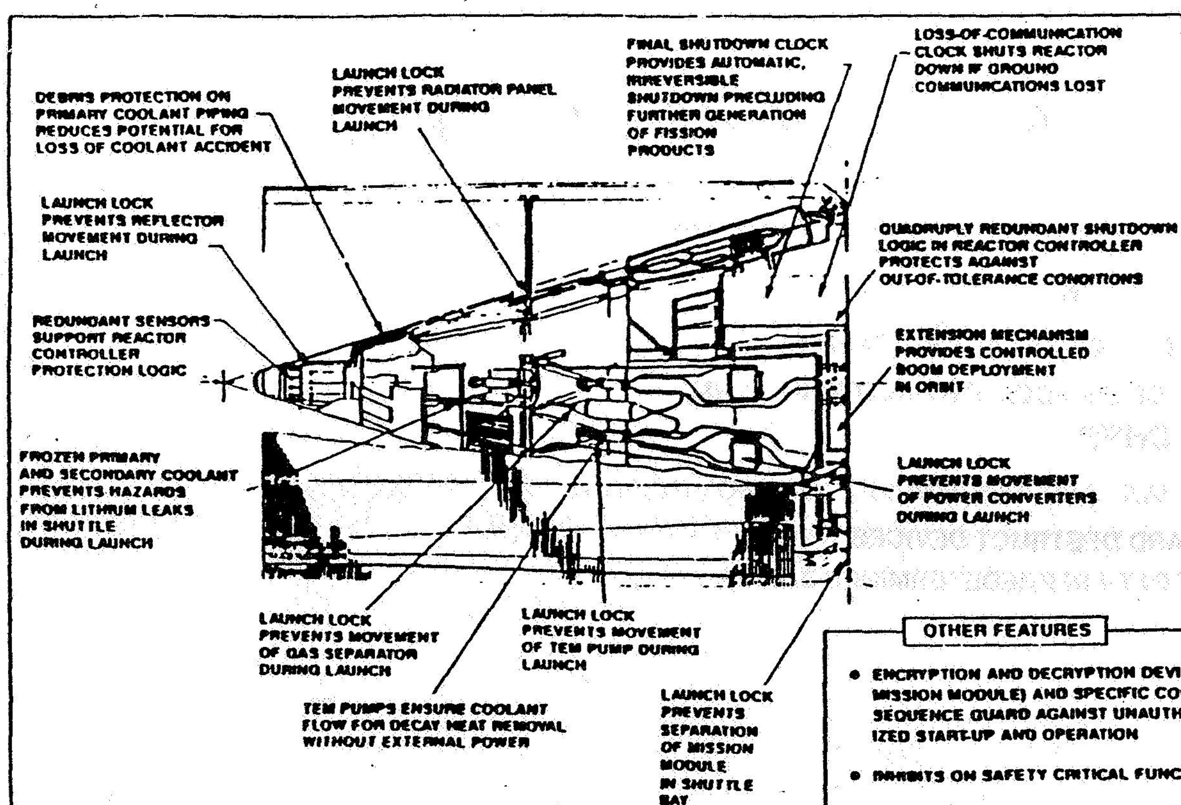
KEY SAFETY FEATURES



- Control elements automatically shut reactor down upon loss of power
- Two independent shutdown systems
- Prompt negative reactivity coefficient assures stable reactor control
- Only 4 out of 12 reflectors required for shutdown
- Fresh core at launch
- Large negative void coefficient enhances shutdown upon loss of coolant
- Control elements moved individually and in icremental amounts to prevent rapid reactivity addition
- Rhenium poison provides thermal neutron absorption for water flooding

Figure 11

KEY SAFETY FEATURES (CONT.)



OTHER FEATURES

- ENCRYPTION AND DECRYPTION DEVICES (IN MISSION MODULE) AND SPECIFIC COMMAND SEQUENCE GUARD AGAINST UNAUTHORIZED START-UP AND OPERATION
- INHIBITS ON SAFETY CRITICAL FUNCTIONS
- MONITORING OF INHIBITS WHILE IN SHUT

Figure 12

GROUND OPERATIONS

- KEY CONCERNS
  - PREVENT ACCIDENTAL CRITICALITY
  - AVOID LOSS OF SNM TO TERRORIST
  - WORKER CONSTRAINTS AROUND LAUNCH PAD
- APPROACHES
  - ENGINE TRANSPORT
    - CORE HEAVILY POISONED
    - WATER-TIGHT STRUCTURE
    - SHIPPING VESSEL FOR "WORST" IMPACT ACCIDENT
    - SHIPPED IN PREFERENTIAL MANNER
  - LAUNCH PAD OPERATIONS
    - KEEP RADIOACTIVE LEVELS BELOW SAFETY LIMITS
    - REDUNDANT AND INDEPENDENT NEUTRON POISONS (E.G., POISON RODS IN COOLANT CHANNELS, LOCKED DRUM SUBSYSTEM)

Figure 13

LAUNCH OPERATIONS

- KEY CONCERNS
  - PREVENT ACCIDENTAL CRITICALITY
  - AVOID FOREIGN COUNTRY ACQUIRING SNM
- APPROACHES
  - REDUNDANT AND INDEPENDENT NEUTRON POISONS
  - ON-BOARD DESTRUCT DEVICES
  - FLIGHT PATH IN PREDETERMINED ZONES

Figure 14

FLIGHT OPERATIONS

- KEY CONCERNS
  - UNPLANNED REENTRY INTO BIOSPHERE
  - RADIOLOGICAL EFFECTS ON CREW
  - FISSION PRODUCT RELEASE
  - CONTINUING OPERATIONS TO GET HOME
- APPROACHES
  - SELECT ANGLES OF THRUST TO ALWAYS MOVE TO SAFER ORBITS
    - SET ORBITS FOR SAFETY
    - INTERLOCKS
    - ENGINE DESTRUCT SYSTEM
  - REDUNDANT AND INDEPENDENT REACTOR CONTROL MODES (INCLUDING SET BACK MODES)
  - SHIELDING USING CONFIGURATION, LH2 IN TANK AND SPECIAL MATERIALS
  - ENCAPSULATED FUELS
  - REDUNDANT ENGINES/POWER PLANTS AND COMPONENTS

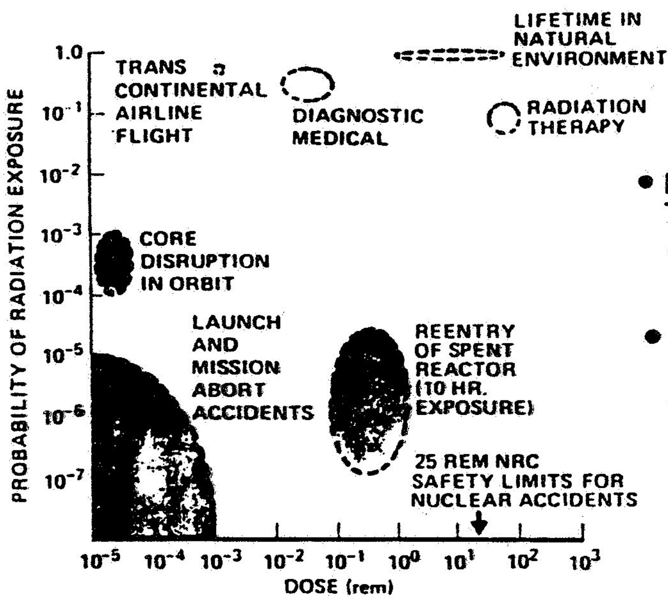
Figure 15

DISPOSAL

- KEY CONCERNS
  - REENTRY INTO THE BIOSPHERE
  - CONTAMINATION OF LOW EARTH ORBIT
- APPROACHES
  - DON'T BRING IT BACK TO LOW EARTH ORBIT
  - SELECT ORBITS TO MINIMIZE RISK

Figure 16

SP-100 RADIATION EXPOSURE vs. PROBABILITY



- PROBABILITIES OF SP-100 ACCIDENTS THAT RELEASE RADIOACTIVITY ARE LOW
- EVEN IF THE ACCIDENTS OCCUR, RADIATION EXPOSURES ARE WITHIN LIMITS

Figure 17

PERCEIVED SAFETY CONCERNS

- KEY CONCERNS
  - REAL AND PERCEIVED RISK CAN BE VERY DIFFERENT
  - EMOTIONAL ISSUE
- APPROACHES
  - REDUCE REAL RISK TO VERY LOW LEVEL
  - OPERATIONAL SCENARIOS MUST BE PLAUSIBLE AND COMPLETE (EX. DISPOSAL)
  - EDUCATION OF CONCERNED GROUPS
  - AVOID DISCUSSIONS OF PROBABILITIES (USE ANALOGIES)

Figure 18

LICENSING

- KEY CONCERN
  - TIMELY LAUNCH APPROVAL
- APPROACHES
  - CONSIDER SAFETY FROM THE START
  - WORK CLOSELY WITH IN PLACE APPROVAL PROCESS

Figure 19

SAFETY APPROVAL PROCESS

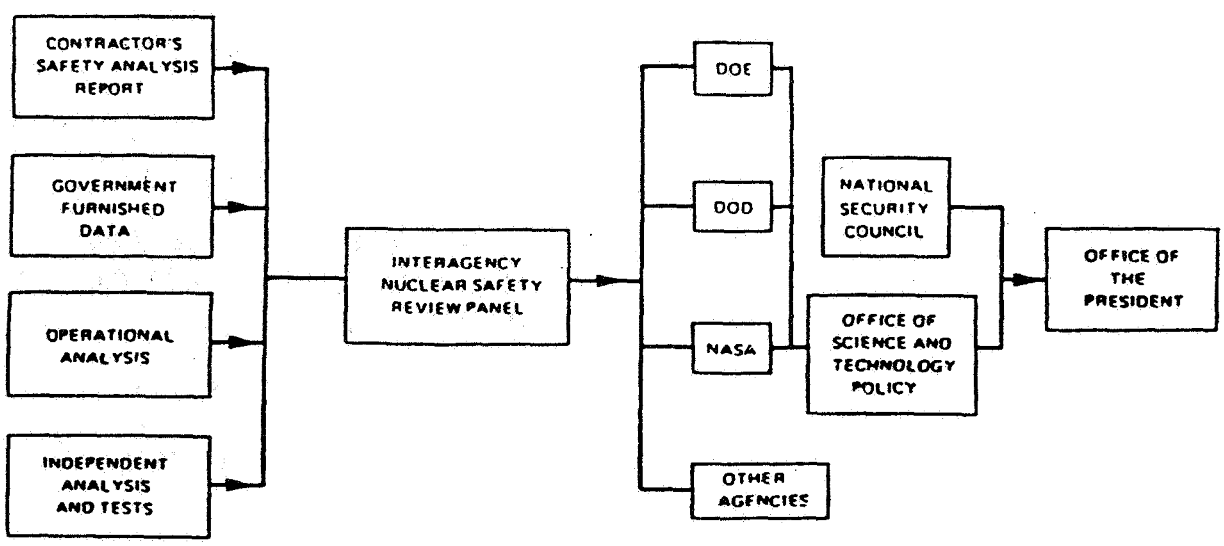


Figure 20

NERVA DESIGN PHILOSOPHY

"THE MAJOR DESIGN CRITERIA FOR THE NERVA ENGINE DEVELOPMENT PROGRAM SHALL BE RELIABILITY AND THE ACHIEVEMENT OF THE HIGHEST PROBABILITY OF MISSION SUCCESS. NEXT IN THE ORDER OF IMPORTANCE MUST BE PERFORMANCE AS MEASURED IN TERMS OF SPECIFIC IMPULSE. THEN THE ENGINE DESIGN SHOULD ATTEMPT TO KEEP THE OVERALL WEIGHT AS LOW AS POSSIBLE WITHIN THE BOUNDS ALLOWED BY FUNDS AVAILABLE FOR DEVELOPMENT. WHILE THERE ARE INTERRELATIONS BETWEEN THESE CRITERIA IN DESIGN, I CAN SEE NO BASIS FOR ALTERING THEIR ORDER OF IMPORTANCE."

MR. MILTON KLEIN (1967)

Figure 21

NERVA FLIGHT SAFETY PROGRAM

- SAFETY PLAN (S-019)
- FAULT TREE ANALYSIS PROCEDURES (S-019-002)
- FLIGHT SAFETY CONTINGENCY ANALYSIS REPORT (S-103)
- RELIABILITY ALLOCATION, ASSESSMENTS AND ANALYSIS REPORT (R202)
- SINGLE-FAILURE-POINT REPORTING, ANALYSIS, CORRECTION AND CLOSEOUT (R101 - NRP-306)

Figure 22

NERVA SAFETY PLAN

- THE MEANS FOR PREVENTING THE INADVERTENT ATTAINMENT OF REACTOR CRITICALITY THROUGH ANY CREDIBLE COMBINATION OF FAILAURES, MALFUNCTIONS, OR OPERATIONS DURING ALL GROUND, LAUNCH, FLIGHT, AND SPACE OPERATIONS.
- A DESTRUCT SYSTEM DURING LAUNCH AND ASCENT TO ASSURE SUFFICIENT DISPERSION OF THE REACTOR FULE UPON EARTH IMPACT TO PREVENT NUCLER CRITICALITY WITH THE FUEL FULLY IMMERSED IN WATER.
- THE MEANS FOR PREVENTING CREDIBLE CORE VAPORIZATION OR DISINTERGRATION OR VIOLATION OF THE THRUST-LOAD PATH TO THE PAYLOAD.
- DIAGNOSTIC INSTRUMENTATION ADEQUATE TO DETECT THE APPROACH OF A FAILURE OR AN EVENT THAT COULD INJURE THE CREW OR DAMAGE THE SPACECRAFT AND THE PROVISIONS TO PRECLUDE SUCH AN EVENT.
- THE CAPABILITY FOR REMOTE OVERRIDE OF THE ENGINE PROGRAMMER BY THE CREW AND GRUND CONTROL AS WELL AS FOR REMOTE SHUTDOWN INDEPENDENT OF THE ENGINE PROGRAM.
- AN ENGINE CONTROL SYSTEM CAPABILITY TO PRECLUDE EXCESSIVE OR DAMAGING DEVIATIONS FROM PROGRAMMED POWER AND RAMP RATES.
- PROVIDE AN EMERGENCY MODE ON THE ORDER OF 30,000 lb-thrust, 500s SPECIFIC IMPULSE AND 10^8 lb-sec TOTAL IMPULSE.

Figure 23

SUMMARY

- POTENTIAL SOLUTIONS EXIST TO REDUCE RISK TO ACCEPTABLE LEVELS
- THE COST OF SAFETY GOES UP DRAMATICALLY IF NOT CONSIDERED FROM DESIGN SELECTION AND INITIATION
- PERCEIVE SAFETY CONCERNS MUST BE ADDRESSED
- LICENSING PROCEDURES IN PLACE AND PREDICTABLE
- OPERATIONAL CONSTRAINTS ARE SIMILAR TO CHEMICAL SYSTEMS

Figure 24

N92-11115

SAFETY ISSUES

R. ROHAL
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One of the primary safety issues is that we have three organizations involved, NASA, DOD and DOE. These organizations have three sets of safety requirements that address and possibly overlap various aspects of the systems we are currently talking about. We have review processes that address each one, so I think that the significant issue is that we have to have some way or another to mold these requirements, as well as the review process, together. If we don't do this, it is going to become cumbersome and may crimp the program. I think that we within NASA are already experiencing this somewhat with regards to Space Station.

We have decided on just what the safety requirements will be for Space Station. However, these requirements are not in one single spot so that the designer can go to them and very easily find out what he has to do to be safe. As of this time, we really have not even defined a safety review process as far as Space Station is concerned.

I think that we may be able to get by for a long time on the Space Station in this current environment, but as time goes on it's going to become more and more difficult, especially for something that has a lot more public visibility such as a sizeable nuclear power source in space. This certainly will be questioned much earlier than something like Space Station. Now I would like to talk about the NASA safety review process.

The purpose of the NASA safety review process is to make sure that we preclude, as early as possible, any system hazards that can endanger the manned flight system. Today, I am going to be talking about the systems that address manned flight in a payload safety review process. However, the philosophy behind it really is NASA's philosophy in addressing how we want to treat safety with systems that interact with man. Payloads that interact with man are definitely handled this way. The shuttle system itself, the orbiter and its elements are handled in a very similar fashion.

The intent is to protect the public, its property, the environment and of course the flight hardware as well as the men associated with it. The responsibility clearly lies with the line management. It's the responsibility of the engineer to design a safe vehicle. It isn't, however, always clear what constitutes "safe." We need to clearly state what the requirements are so that the designers and engineers understand them.

Finally, I think the safety organization itself is responsible for review oversight, independent assessment, and defining and making sure that the requirements are disseminated and understood.

The types of basic hazards that we normally address on any of the payloads are:

contamination, electrical shock, explosion, radiation, and temperature extremes. With regard to any one of these particular items, there are a lot of documents which define very specifically what materials you can and can't use, what safety factors you should design and etc.. Of course, all of the hazards are appropriately documented, and either periodically reviewed, or approved both by the safety organization and the program management.

The critical thing is the way NASA defines the severity of the hazard. And as far as NASA is concerned, your design must be dual failure tolerant, you have a critical hazard that will cause a damage or failure of some space hardware or injury to personnel.

With regard to their systems that interface with man, NASA really requires designs that are dual failure tolerant. It's difficult to get around this, and is something that should be considered in our talk today. This became stronger with regards to Space Station and shuttle since the Challenger event.

The primary document that NASA uses, as far as its manned programs are concerned, is a hazard report. Essentially this report identifies the hazard, tells you what causes the hazard, tells you how to control the hazard, and tells you how you are going to verify through analysis and testing. I really stress testing because on the manned systems, you really have to have some tests supporting your claims, and your analysis on critical and catastrophic hazards. Then of course you have the appropriate approvals.

This is just the surface of what goes into a hazard report. A hazard report could be several hundred pages long. It tells the review committees how you are going to eliminate the particular hazard.

Safety analysis is just part of the verification process, and probably less important than the two system analysis or the system test. There are analyses that are accepted and address the various systems. Normally we have fault trees, FMEA's, and various calculations to show that the systems are indeed safe and reach their margins of safety.

The review process is conducted in several phases. We have an initial review and a conceptual state, (the project more or less presents the concept). They identify the operations, both from the ground standpoint and from the flight standpoint. The safety organization is there to help interpret and help the project to prepare for the Phase 0.

Phase 1 comes around right after the Preliminary Design Review. Here you start to clearly define all the hazards, and you produce your preliminary hazard report, your approach to verification, etc.

The Phase 2 review is conducted right after your critical design review. Here you have considerably more material to present, such as engineering drawings, and most analyses. You more or less define how you are going to control your hazards.

Phase 3 is the most critical. It occurs after you have done most of your testing and qualification. Here you really understand how your system is going to work, you understand the problems that your system has had, and you are able to show that you have tested and qualified the equipment to the environments that you expect to see in a particular application.

The DOE process and the DOD process are somewhat similar, but they are different. In the nuclear world, independent reviews are scheduled periodically.

In the Space Station world there is a process defined, (it's awaiting final cost approval from Dick Lures, the program director), but that process is very similar to this used by shuttle. The payload process that Bob is pointing out is a part of shuttle process. The review that he is describing, the ones that Space Station will have, are not done by direct program people. When they are reviewed, the information is provided by those in the program. The hazard reports are developed, the hazard analyses are done and then they are reviewed by people not directly involved in the program, but who do have sufficient knowledge to perform the review.

That information then gets forwarded through the independent safety and product assurance organization, up to the program director for his final concurrence or rejection. That process is in effect an independent review. They use separate engineering people, separate propulsion and electrical folks to review the work that has been done by the program engineering people.

For this particular process, a lot of the technical review is done indepedently of the program by people at Johnson. In the case of the Space Station I am not sure we defined exactly who the independent technical reviewers will be. We have not gone this far yet, have we?

I guess my final word is that I think the primary safety issue is that we really don't have a set of requirements defined for space nuclear plants that we can easily locate. I am not saying that we ought to go out and redefine requirements, but I think we need to provide some sort of a road map as to which requirements exist and where. If there are conflicts, what should we do about those conflicts? Secondly I think that we need to consider just what the review process shall be.

It's going to be pretty difficult for the designers to design easily with safety in mind if we don't do this for them. They are going to have a difficult time really understanding what the requirements are, so my recommendation is that we get the safety communities of NASA, DOE and DOD together and jointly define just what the requirements are, how to get to all the requirements, and also start to define just how we are going to do the appraisal, and evaluation of the designs and the resulting data.

I think that there are good safety organizations in all three of those organizations. I think that we have to get them together. We have to be able to identify what are the right things to do, what to do about those surface conflicts and then get them resolved. Finally, we need to set forth just how we are going to show the public that we have made sure that we have safety systems.

DISPOSAL METHODS

A. Friedlander
SAIC

I am going to discuss a number of disposal options for space nuclear reactors and the associated risks, mostly in the long term, based on probabilities of Earth reentry.

The results are based on a five year study that was conducted between 1978 and 1983 on the space disposal of high level nuclear waste. It was a study actually begun at Lewis Research Center and later transferred to Marshall. The study provided assessment of disposal options, stability of disposal or storage orbits, and assessment of the long term risks of that bad stuff coming back to Earth.

Just recently, we completed an application study of nuclear thermal rockets to the lunar outpost scenario. I suppose most of the mission results that you have heard about have to do with Mars, but we have looked at it in terms of the moon and have examined, as part of that overall study, the case of the disposal options. Therefore, I will try to configure the presentations so that it will treat both the moon and Mars because many of the options are quite similar as I will show you.

Just to put it in perspective, for the lunar NTR study we looked at various combinations of NTR (see Figure 1) starting with one burn, that is just using it for the translunar injection (TLI) and then doing everything else chemical and aero. We end with a complete four burn, where the nuclear thermal rocket was the only propulsion system starting from LEO and going back into LEO.

The disposal options you have available that might work best depend very much on how the nuclear reactor is going to be used in the mission scenario. If it were only going to be used for TLI in this case, or transMars injection (TMI), then you would have a different kind of a disposal option, probably, than if it were going to be used and brought all the way back to Earth orbit, perhaps reused for several missions, but eventually disposed of in some way.

So what we mean by a spent reactor then is a device that has been operated and is radiologically active at end of life (see Figure 2). Normally the end of life would occur after normal operations and the number of reuses that it has been designed for. But, of course, end of life could also occur from a disabling accident, in which case a disposal option may be required too.

Then the question is what to do with that spent reactor to eliminate or minimize the subsequent hazard of the radioactive material coming back to Earth: being released in the biosphere.

I have listed some ten factors of consideration. If there is a need for disposal at the end-of-life, it's a fairly complex problem and needs to be considered in terms of trades (see Figure 2)

Let's look at some of the disposal options (see Figure 3 and 4).

They start with some moderate altitude Earth orbits that would be stable for some period of time, to a high earth orbit, which I called super GEO, somewhat above normal operations in GEO. If it were used for a lunar mission it could be a lunar surface delivery, including impact, which is probably not desirable, or an actual soft landing and storage on the moon.

The libration points of the Earth-moon and Earth-sun system are a possibility for disposal. If you are going to Mars you could leave it in Mars orbit. There are also libration points in the Mars system. Or we could put it into an Earth elliptical orbit, which does have a long-term risk of reentry, which I will describe. You could put it into a solar orbit that is stable for very long periods of time, which could apply either to the moon or Mars missions. Or, you could send it out of the solar system altogether, but the Delta V to escape the solar system is so high it would have a serious impact on mission performance.

To give a flavor of the kind of work that was done for the lunar application, we examined all of the cases shown in Figure 3 an 4. We looked at the situation of a disposal from a particular orbit state to another orbit state. We then calculated the disposal Delta V that would be required at the end-of-life. We found it varies quite a bit.

The lowest Delta V disposal was lunar gravity assist as applied to the NTR 1-burn case. You could deflect the trajectory to the trailing edge of the moon, take a lunar swing by and inject into a heliocentric orbit. A possible disposal solution for the NTR 4-burn case is a 1000 km circular orbit about Earth for a Delta V cost of about 300 meters per second.

The highlighted disposal options are the ones we actually examined in detail. We made comparisons against the nominal mission performance, and tried to determine what the disposal actually cost in terms of mass penalty.

In the case of the full NTR burn for lunar applications we examined two options. One was to put it into a heliocentric Earth-crossing orbit after coming back to LEO at end-of-life, or raise that orbit to a thousand kilometer altitude Earth orbit.

Now I am going to talk about reentry risk (see Figure 5). For example, a propulsion system failure might occur during injection prior to actually escaping the Earth. If we start in an orbit that has a high eccentricity with crossing of the lunar orbit distance, then you would have a mean reentry lifetime of 200 to 700 years. Lunar collision would occur with a much smaller lifetime, on the order of 50 years.

A VOICE: Are you talking about reentry into earth or an encounter with the moon?

MR. FRIEDLANDER: This combines both types of events. In other words, this was not a disposal orbit but an orbit that resulted from some kind of a failure that had a perigee close to Earth and crossed the lunar orbit. Subsequently, this "stay body" would either reenter Earth's atmosphere, collide with the Moon, or be ejected from the Earth-Moon system.

Now, what I want to talk about are some disposal options, including the stable solar orbit, the heliocentric planet-crossing orbit, and then the moderate-altitude Earth orbit.

Figure 6 is a plot of heliocentric planetary distances. It shows the maximum and minimum extent of Earth, Mars and Venus. It turns out there are two stable zones not too far from Earth. One of them is between Earth and Mars, 1.17 to 1.19 AU circular orbit. The other is between Earth and Venus. If you can get it into that circular orbit it's going to stay there for a very long period of time - at least a million years.

To give you an example of what happens to that orbit, Figure 7 is a time history over a million years of an orbit which was initially at 0.86 AU circular, between Earth and Venus. It doesn't stay circular at all because of the mutual perturbation of the planets and Earth. You can see the Venus aphelion and the Earth perihelion changing quite a bit with time.

But, though it doesn't stay circular, the disposal orbit is stable to at least a million years and probably much longer. That is to say, it does not become a planet-crossing orbit.

In fact, this was the nominal disposal destination selected for space nuclear waste after consideration of all the possibilities.

Now let's look at a situation of an Earth-crossing orbit where the orbit starts out initially with a perihelion of .85 AU and a aphelion at 1, so it left Earth on the way toward a stable circular orbit. But let's suppose that the circularization burn at .85 failed and we are left in an Earth-crossing orbit.

Figure 8 shows the results of the Monte Carlo statistical analysis. Initially the orbit only crosses the Earth orbit, but because of the gravitational effects over the long term, it actually begins to cross all the planets out to Jupiter and could be eliminated by collision in various ways or by solar system ejection caused by Jupiter gravity perturbations.

In 54 percent of the cases it will eventually come back to Earth reentry. However, the mean time for that to occur is 26 million years, which is a rather long time. There is also a substantial probability of Venus collision, and once the object begins to cross Jupiter, at least ten percent of the time it will be ejected from the solar system altogether.

But the dominant event is an Earth collision and what is shown here is probability as a function of time for the various collision events.

So, for example, even though the mean lifetime greater than over 20 million years, at one million years the probability of Earth collision is 17 percent.

Figure 9 shows those results along with the sensitivity to orbit perihelion distance and inclination. For each of these cases the mean time to reentry is quite long, but there is a finite and not insignificant probability of Earth reentry occurring over shorter time periods.

A VOICE: Right now we are looking at an Earth reentry time of a "nuclear safe orbit" of 300 years. You are several orders of magnitude beyond that even in your worst case.

MR. FRIEDLANDER: That's quite true. This would be a very favorable result unless some particular design and analysis of the fission products showed that you really needed to provide nuclear safety for many, many thousands of years.

A VOICE: That's going to be a function of the safety groups to determine what is the minimum time we can have for reentry of any nuclear system in Earth orbit.

What I am trying to show is that, in the long term, we are talking about probabilities which might be quite acceptable. In fact, from my point of view, an Earth-crossing orbit is a fairly acceptable disposal place.

MR. FRIEDLANDER: You can get about a three and a half fold reduction in collision if you went ten degrees out of the elliptic plane. However, it's very costly to get ten degrees out of the elliptic plane.

If you are talking about disposal, you really want to put is someplace and be done with it. You don't want to be monitoring it for thousands of years.

A VOICE: I might want to reuse the materials.

MR. FRIEDLANDER: You might. In fact that was one of the considerations when it came to looking at space disposal of high level nuclear waste. Some people said they might want to use it in ten thousand years, so some people wanted to put it into Earth orbit. But that high level waste is bad stuff compared to a reactor.

Let's talk about the disposal in a moderate altitude Earth orbit. Consider a lunar or

Mars application that has been used for four burns and comes back to LEO. At that point, the easiest thing to do is add a very small Delta V is raise it up in altitude. Figure 10 shows the orbit lifetime against atmospheric drag and reentry.

If you place it at a thousand kilometers this would give you a lifetime against Earth reentry of 24 hundred years.

So if the safety time requirements are on the order of a few thousand years, you could put it into a moderate altitude circular orbit about Earth.

Figure 11 shows results from a recent paper by Chobotov and Wolfe in the Journal of Astronomical Sciences, January- March of this year. It's probably the latest update of a summary of the meteoroid and debris flux impact per year per square meter as a function of particle diameter.

This is the natural or man-made environment that an object put into a moderate altitude disposal orbit would face.

If you have a collision with a meteorite it's at about 20 kilometers per second impact speed. A collision with space debris tends to be around ten kilometer per second impact speed.

Even though the debris flux is low, it's getting worse and worse, and some people talk about trying to sweep some of that debris out. But, there is debris out there which could certainly do damage.

In future work, I would think that we might want to do a preliminary trade study of the disposal options for Mars applications to get a handle on what the impact on the nominal mission performance would be.

There are also short-time reentry risks. These would come about as a result of failure or accident environments. In this case, quantitative information about risk could not come out of the long-term statistical analysis that I have described. A different type of analysis would have to be performed.

Both short-term and long-term risks were examined in the previous studies of space disposal of nuclear waste. We looked at the reentry probability and the radioactive element inventory as a function of time. This was quite important for nuclear waste. I am not sure how important it is for the reactor operation but it is something that might need to be done. Eventually one would want to do an overall risk benefit assessment of disposal options.

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NTR CONCEPT OPTIONS

SPECIFIC APPLICATION OF NTR CONTROLS THE TRADE SPACE: 4 CASES

"Handle"	Propulsion Application: Trans Lunar Inject	Lunar Orbit Capture	Trans Earth Inject	Earth Return Capture	LTV/NTR Separation	Application
1-BURN	NTR	CHEM	(CHEM)	(AERO)	After TLI	· NTR to CHEM Transition · Easy disposal
2-BURN	NTR	NTR	(CHEM)	(AERO)	Post-Capture at Moon	· Expendable LTV · One-way Cargo · Low-Risk NTR Use
3-BURN	NTR	NTR	NTR	AERO	After TEI	· Reduce Risk to Crew on Return
4-BURN	NTR	NTR	NTR	NTR	After Earth Orbit Capture	· All-Nuclear LTV

( ) = ONLY IF THE MISSION IS A ROUND TRIP

NOTE: "Handle" refers only to number of major LTV burns this mission

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Figure 1

END-OF-LIFE DISPOSAL OF SPACE NUCLEAR REACTORS

- DEFINITION OF SPENT REACTOR

  DEVICE HAS BEEN OPERATED AND IS RADIOLOGICALLY ACTIVE AT END-OF-LIFE WHICH OCCURS EITHER AT TERMINATION OF NORMAL OPERATION OR AS A RESULT OF A DISABLING ACCIDENT EVENT

- QUESTION OF FOCUS

  WHAT TO DO WITH SPENT REACTOR TO ELIMINATE OR MINIMIZE SUBSEQUENT HAZARD OF RADIOACTIVE MATERIAL RELEASE TO BIOSPHERE?

- FACTORS OF CONSIDERATION

  1. DISPOSAL DESTINATION
  2. PROPULSION SYSTEM REQUIREMENTS AND COST
  3. OPERATIONAL COMPLEXITY, RELIABILITY AND COST
  4. FAILURE MODES AND ACCIDENT ENVIRONMENTS
  5. PAYLOAD RESPONSE TO ENVIRONMENT
  6. ORBIT EVOLUTION CONSEQUENCES OF FAILED ORBITS
  7. PAYLOAD MONITORING
  8. RETRIEVABILITY/RESCUE MISSION CAPABILITY
  9. REENTRY PROBABILITY - SHORT VS LONG TERM
  10. RADIOACTIVE RELEASE RISK TO BIOSPHERE

Figure 2

NTR DISPOSAL OPTIONS

DISPOSAL FROM	DISPOSAL TO	ΔV (m/s)	COMMENTS
Post-TLI Separation Trajectory (1 Burn)	Lunar Gravity Assist (LGA) to Heliocentric Earth-crossing orbit	30	• C3 = 2 [illegible] • Mean lifetime to reentry depends on r,a [illegible]
"	LGA to E-M L1 Halo	580	• Must control for long-term orbit stability
"	LGA to E-M L2 Halo	330	"
"	LGAs to E-Sun L1 Halo	54	"
Post-TLI (1 Burn) or Post-TEI (3 Burn) trajectory	Perigee kick to final heliocentric orbit	194	• Orbit at 1x1.15 or 0.88x1 AU • Reentry risk = f(rp,a) i=2 deg
"	Capture to h=1,000 km (1 Burn - free return)	2955 155	• Capture incl. 20 m/s nav. • Circularize at h = 1,000 km
"	Raise orbit altitude to "Super-GEO"	710 1456	• Capture incl. 20 m/s nav. • Circularize at h = 36,287 km
"	Solar circular orbit at .85 or 1.19 AU, i=2 deg	200 1250	• Orbit stability > 1,000,000 yrs [illegible]
"	Solar system escape	5678	• C_3 = 152

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Figure 3

NTR DISPOSAL OPTIONS (CONTINUED)

DISPOSAL FROM	DISPOSAL TO	ΔV (m/s)	COMMENTS
300 km circular lunar orbit (2 Burn)	Lunar surface delivery	2000	• 2-burn program for controlled landing on lunar surface
"	E-M L1 halo orbit E-M L2 halo orbit E-Sun L1 halo	1150 775 850	• 2-burn sequences
Full NTR Propulsion (4 Burn)	Heliocentric Earth-crossing orbit	3267	• Orbit is 1x1.15 or 0.88x1 AU i = 2 deg; 3200 m/s w/LGA
"	Solar circular orbit at .85 or 1.19 AU; i=2 deg	3300 1250	• 2 burns to circularize
"	Capture to h=1,000 km	313	• Earth orbit [illegible]
"	Raise orbit altitude to "Super-GEO"	3859	• 2 burns and 20 m/s nav. h = 36,287 km
"	Solar system escape	8751	• C3 = 152
"	Refurbish for Use on Robotic Mission	varies	• SEI Mars Robot Explorer • Outer Solar System Mission

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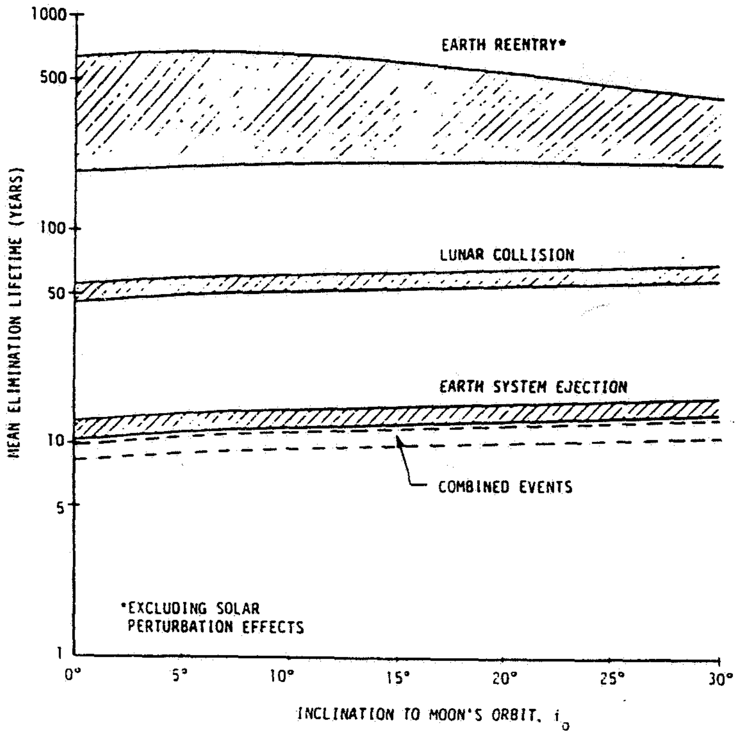
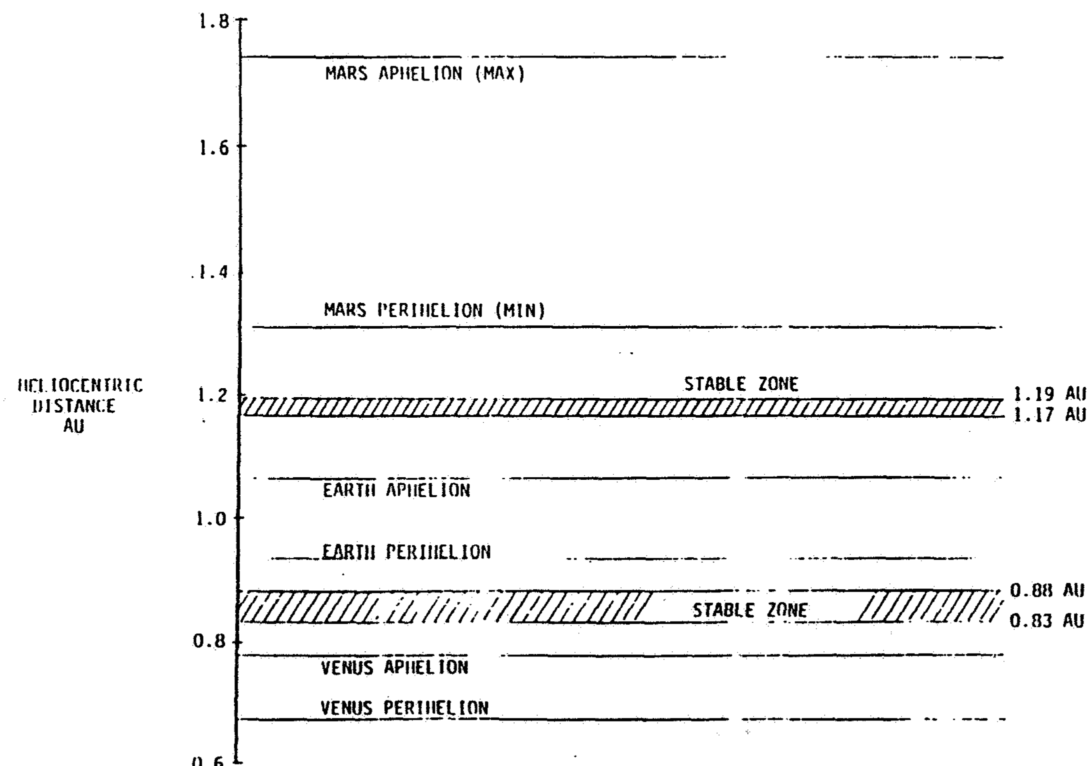
Figure 4



PROBABILISTIC EFFECT OF LUNAR ENCOUNTERS ON ELIMINATION OF STRAY BODY IN LUNAR CROSSING ORBIT (OPIK'S THEORY)

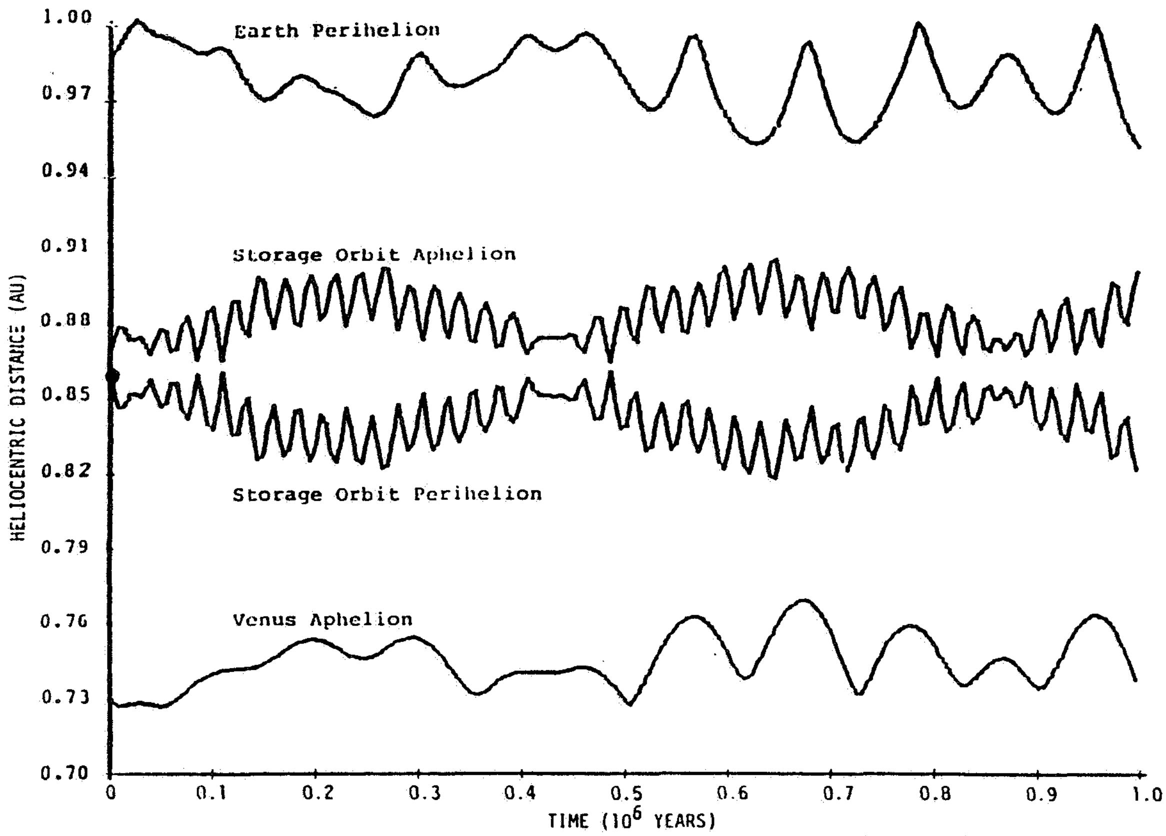
$e_J = 0.965; \quad 1.02 \leq Q_0 \leq 1.10$ LUNAR DISTANCE

Figure 5



STABILITY BOUNDS OF INITIALLY CIRCULAR ORBITS BETWEEN THE PLANETS

Figure 6



ORBITAL VARIATIONS OF AN INITIALLY CIRCULAR ORBIT AT a = 0.86 AU

Figure 7

PLANETARY ENCOUNTER PROBABILITY ANALYSIS – COMPUTER PRINTOUT

MONTE-CARLO STATISTICAL SUMMARY

NUMBER OF CASES = 500
MEAN LIFETIME =.438E+08 YEARS

EVENT		NUMBER	FREQUENCY	MEAN TIME	MEAN U
COLLISION WITH PLANET					
	MERCUR	9	0.0180	.536E+08	0.507
	VENUS	162	0.3240	.310E+08	0.264
	EARTH	270	0.5400	.263E+08	0.189 (V_e = 12.6 km/sec)
	MARS	8	0.0160	.578E+09	0.325
	JUPITE	1	0.0020	.524E+08	0.619
	SATURN	0	0.0000	.000E+00	0.647
	URANUS	0	0.0000	.000E+00	0.000
	NEPTUN	0	0.0000	.000E+00	0.000
	PLUTO	0	0.0000	.000E+00	0.000
0 SOLAR IMPACT		0	0.0000	.000E+00	
SOLAR SYSTEM EJECTION		50	0.1000	.925E+08	

0 MEAN ELEMENTS AT ELIMINATION
A= 2.041 E= 0.279 I= 7.65

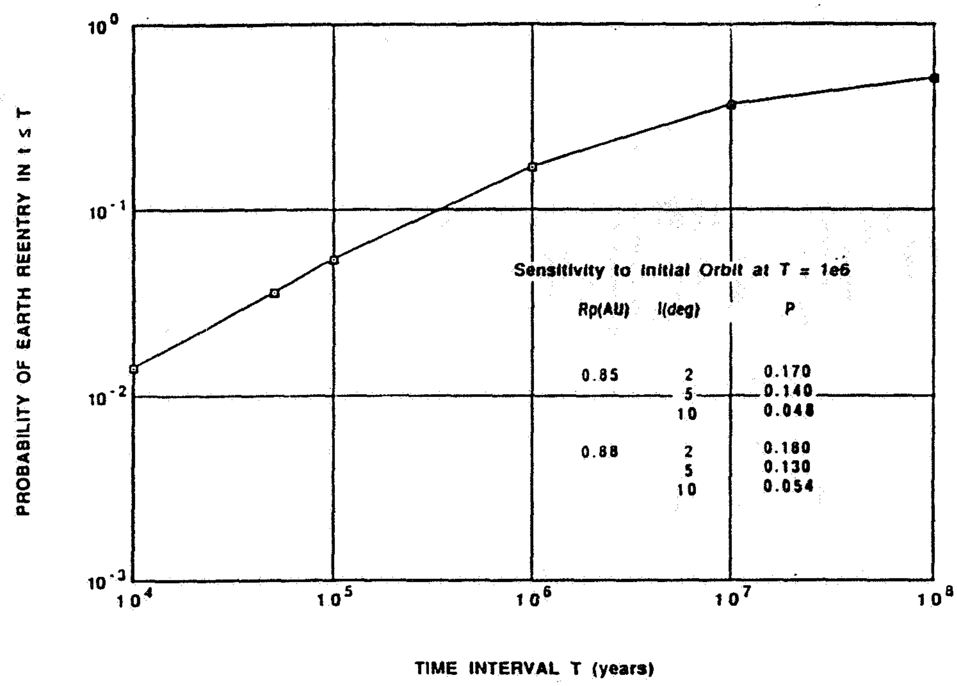
0

INITIAL ORBIT CONDITIONS TIME= 0.00
PERIHELION(AU) = 0.850 APHELION(AU) = 1.000
INCLINATION(DEG) = 2.000 PERIOD(YRS) = 0.890

0EVENT PROBABILITIES

0FOR TIME INTERVAL =	0.10E+05	0.50E+05	0.10E+06	0.10E+07	0.10E+08	0.10E+09
COLLISION WITH PLANET						
MERCUR	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.20E-02	0.16E-01
VENUS	0.00E+00	0.20E-02	0.20E-02	0.30E-01	0.17E+00	0.31E+00
EARTH	0.14E-01	0.36E-01	.54E-01	0.17E+00	0.37E+00	0.51E+00
MARS	0.00E+00	0.0	.00E+00	0.00E+00	0.20E-02	0.40E-02
JUPITE	0.00E+00	0.0	.00E+00	0.00E+00	0.00E+00	0.20E-02

Figure 8



Long-Term Probability of Earth Reentry
Initial Orbit: 0.85 x 1.0 AU, I = 2 deg

Figure 9

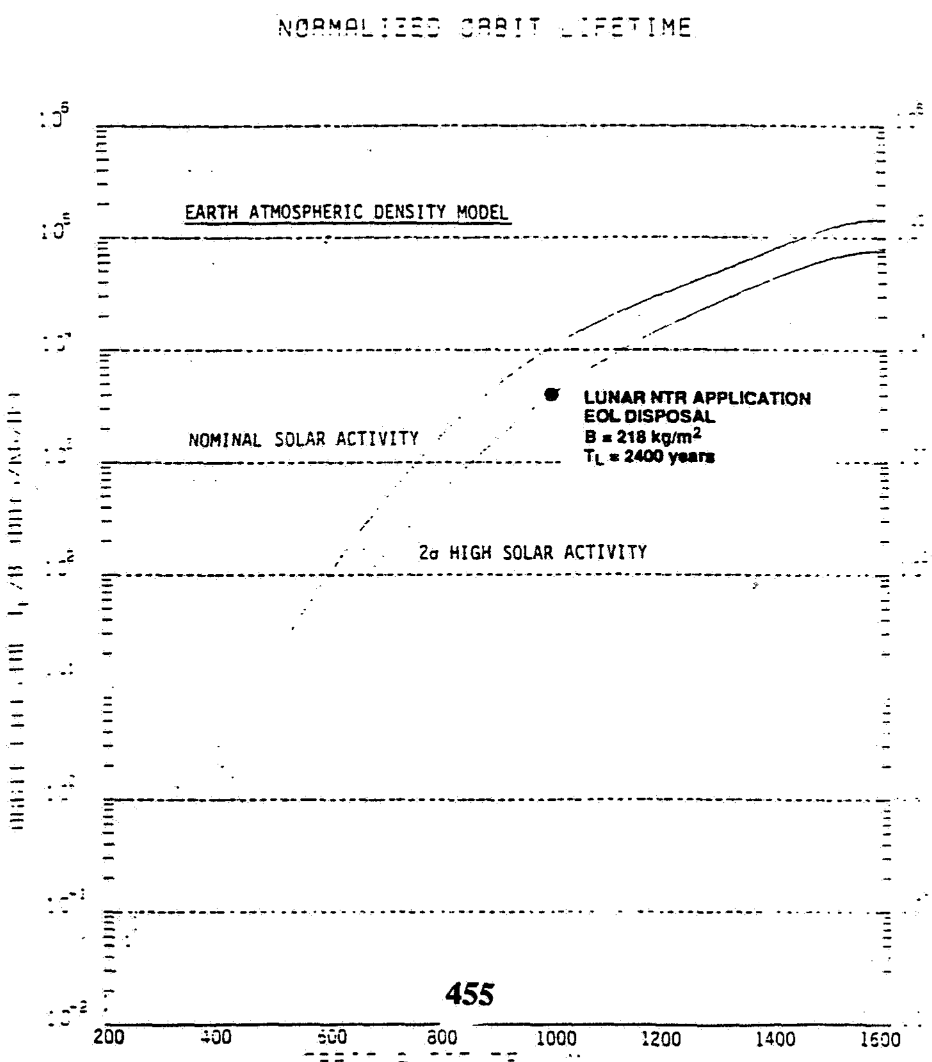


Figure 10

fragmentation events is the migration of the fragments as a function of time, resulting in a larger volume of debris through which threatened satellites may have to pass. For instance, the fragmentation of an object in a 60 deg inclination orbit produces a torus of orbits at 60 deg inclination. With time, the orbits of the fragments remain at 60 deg inclination but cross the equator at random points. The fragmentation cloud effects are therefore expected to be of great significance to the permanent manned occupation of low Earth orbit as well as to the safe use of the remainder of near-Earth space.

The total mass of man-made debris in orbit is currently on the order of three million kilograms at altitudes below 2000 km. This compares with about 200 kg of meteoroid mass in particles of about 0.1 mm diameter. Man-made debris is much larger, however, ranging from millimeters to meters. Also, the relative velocity at encounter is on the order of 10 km/s for space debris, compared to 20 km/s for meteoroids. Primary sources of man-made debris were more than 100 United States and Soviet spacecraft explosions in orbit, possible collisions among debris objects, and the erosion of spacecraft surfaces due to oxygen, ultraviolet, and thermal radiation effects. Solid rocket motor firing also contributes to particles in the 0.0001-to-0.01-mm range [3].

In addition to the above sources of man-made debris, there have been numerous intentional breakups of spacecraft in orbit. Among these were a number of Soviet antisatellite tests at low altitudes, and United States intercepts of spacecraft in orbit, such as the breakup of the Solar Wind satellite in 1985 and a Delta-180 stage in 1986.

On November 13, 1986, an Ariane third stage exploded into more than 460 trackable objects 10 cm in diameter and larger. This suggests the presence of several thousand or more small particles that cannot be tracked by ground-based radars. Recent orbital debris measurements are summarized in Fig. 3 [3].

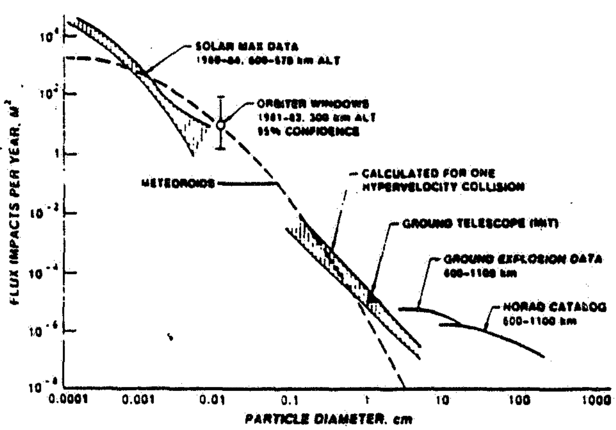


FIG. 3. Recent Orbital Debris Measurements Compared to Meteoroid Flux

Figure 11

WORKSHOP FEEDBACK

Mission Analysis Panel
Tim Wickenheiser

This is the mission analysis panel review. Figure 1 lists the members of the mission analysis panel. We felt that the proposers did a great job in presenting their concepts given the time to prepare (Figure 2).

Compared to the previous NEP conference, we feel that the CFP's at this Workshop provided a better focus on the mission aspects. They didn't answer all of our issues, but I think within the definitions of their concepts, they did the best they could. In general, what we found was that all concepts that were presented exceeded the baseline mission, with the exception of a couple of the concepts which were 1960's versions that had just not updated their technology (Figure 3). Many of them didn't address issues such as reliability, safety, lifetime operations, but they did provide appropriate level based on the preliminary concept definitions. One thing I want to stress is that these qualities are important to NASA and if you begin seriously considering these concepts, they must be addressed early in the concept.

Overall, what we found is that if you looked at capability versus uncertainty, the panel felt that the systems with the best concept definition had the lowest capability and as you improve technology the capability goes up as does the uncertainty in the ability to achieve that capability. The solid core systems produced Isp's of 900 to 1000, and they were basically a function of temperature (Figure 4). That tells us that to improve solid core technology, you work on higher temperature fuels. There was some variation in the solid core concepts, primarily in thrust-to-weight. While thrust-to-weight offers you some performance advantage, its greater benefits are in terms of operational issues. (i.e. assemblies didn't require multiple perigee burns)

Basically, mission benefits increased as you went from solid core to liquid core to gas core. There were three interesting concepts that came out of the workshop that (Figure 5) didn't quite fit into the baseline mission. First of all, the low pressure concept. There was a lot of controversy as to whether they could really get the benefit out of dissociation/recombination. Low pressure dissociation provides benefits to a lot of the engine concepts. One of the key issues is finding out whether the benefits really exist or not.

The hybrid systems didn't really fit well into the mission scenario we gave them. There are a lot of issues with hybrid systems' reliability. Does it decrease the Isp of the initial NTR system? If so, how much? These issues really need to be understood before we can really understand how much benefit, if any, the hybrid systems give you.

can really understand how much benefit, if any, the hybrid systems give you.

The problem we had with NIMF was that it's not really an Earth to Mars propulsion system as presented. It was more of a hopper on Mars to give you additional science. But I think it's a very interesting and very positive thing. The panel didn't know how to judge it in the criteria we have been given. I think this needs to be given to MASE and evaluated as part of an overall mission scenario.

So what do we think is next (Figure 6)? The first thing we feel is necessary is to get the propulsion people and the power people and the vehicle people and mission people together and do an integrated study to find out what the real mission benefits of these are. The operational and redundancy issues are very important to selecting the system. You need to understand those fairly early.

We recommend investigating dissociation/recombination; it offers tremendous advantages if it really exists.

The other issue with the solid core NTR was the higher temperature. We need to find out if that higher temperature affects the reliability of your system. You need to find out if the higher Isp (or a higher temperature) reduces reliability.

Finally, all the concepts basically exceeded the performance requirement, but the real major reductions in trip time come about when you go to gas core (Figure 7).

I can't overstress reliability, safety, lifetime issues. Those are going to be very important.

Facility requirements are going to be a key driver in selecting technologies, particularly with dissociation. You need to make sure the facilities are compatible with whatever system you have. We had a wide range of systems being proposed. How do you build your facility or define your facilities early enough to be able to accommodate the uncertainty in the selection of the technology?

And finally, I guess this is more to my fellow NASA people, I think we need to be careful in how we sell nuclear propulsion. There is a danger if we just go off and say, "it reduces the trip time in half, therefore it's great." If trip time no longer becomes important, if there is some political decision that says it's okay to take two years, then we may not have a program. We need to understand and preserve all the benefits of nuclear propulsion.

NASA – Lewis Research Center

Technology Review Panels: (Both NEP and NTP Workshops)

Mission Analysis:

WICKENHEISER,TIM	NASA	LERC	PANEL CHAIR
SAWYER,BUZZ	NASA	HQ/QS	CREW SAFETY
DANDINI,VINCE	DOE	SNL	NUCLEAR SAFETY
PERKINS,DAVE	DOD	AFAL	PROPULSION SYSTEMS
COOMES,ED	DOE	PNL	POWER SYSTEMS
AUSTIN,GENE	NASA	MSFC	
EVANS, DALLAS	NASA	JSC	LUNAR/MARS EXPLORATION
GEORGE,JEFF	NASA	LERC	NEP SYSTEMS ENGRG
GILLAND,JIM	NASA	LERC	NEP STUDIES, THRUSTER TECH.
HACK,KURT	NASA	LERC	TRAJECTORY ANALYSIS
SAUER,CARL	NASA	JPL	TRAJECTORY ANALYSIS
SUMRALL,PHIL	NASA	MSFC	
STANCATI,MIKE	SAIC	ILL	EXEC.SEC.

NUCLEAR PROPULSION PROJECT

Figure 1

CFP Presentation to Panel

- o CFP's did a great job, given time to prepare, resources
- o More focus on Mission Analysis then Pasadena
  "I'm not a mission analyst, but..."
- o Did not answer all of our issues
  Reference: the list of questions

Figure 2

Concept Capability vs. Baseline

o All concepts met baseline mission requirements; majority excluded baseline performance.

o Many did not address:

- Safety
- Reliability
- Lifetime
- Operations

May have been appropriate for level of concept definitions

o The Panel's Rating

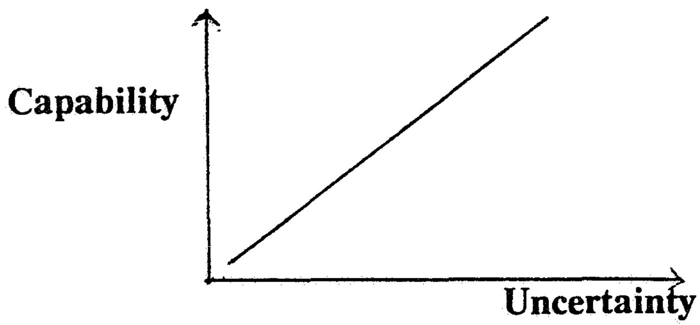


Figure 3

Specific Comments

o Many concepts (solid core) @ Isp 9000-1,000 s.
where Isp = f(temperature)

o Solid core concepts varied in F/W. Higher F/W offers performance and/or mission operations advantages.

o The "Phase Change"

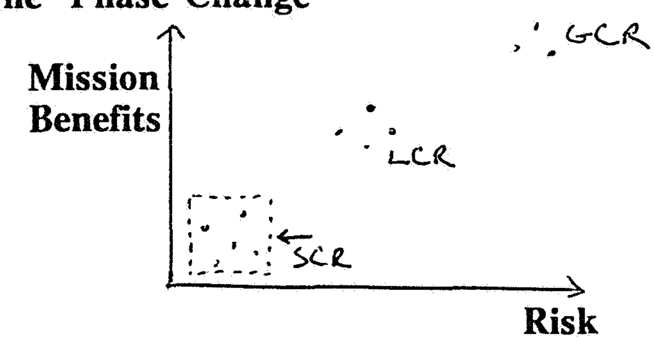


Figure 4

Interesting Concepts

- o Low Pressure

- o Hybrid Systems

- o NIMF

Figure 5

What's Next?

- o Need integration study of:
  Mission - Vehicle - Propulsion - Power System

- o Recommended investigating effects of
  dissociation/recombination

- o Since high temp drives high Isp, technology should be investigated

-but-

Must include implications on reliability

Figure 6

Summary

o All concepts met, and most exceeded, performance requirements of baseline.

o Safety, Reliability, Lifetime, and Operations issues should be addressed early.

o Facility requirements.

o Care must be taken in selling Nuclear Propulsion on a single criterion (INLEO). Support must be broadly based, and built on many factors.

Figure 7

WORKSHOP FEEDBACK

Propulsion Panel and
Reactor Technology Panel
Ned Hannum

The first point I would like to make is that this is the end of the workshop remarks, but certainly not a summary. There wasn't time to summarize these remarks into any intelligent or comprehensive summary. The most important general comment is that as propulsion people, we certainly don't have any problem with saying that nuclear thermal propulsion will work and can get the job done (Figure 1). It can get it done with reasonable funding and certainly within reasonable time periods.

Our panel had a little bit of difficulty knowing where reactors quit and propulsion begins. I'll summarize the results of both the Propulsion Panel and the Reactor Panel.

We are really short on systems-level information at this point in time, except for a couple of systems. NERVA certainly has addressed systems-level issues. However, a couple other systems have been looked at in considerable detail and are still short of propulsion system level understanding. We need to figure out ways to go out and get that information.

We need to provide substantial design margins. I recognize this is a paradoxical comment because, at the same time, we are out looking for every second of Isp that we can find. But in the end, we have to come up with a propulsion system that has a great deal of design margin. One of the examples that I feel comfortable with is the RL-10. The RL-10 has been an important part of our transportation system over the years and will continue to be. I think one of the things that makes it so is that it was originally designed with a great deal of design margin. When the target changed, when things had to be done a little differently, there was room to accommodate those changes within the basic concept of the RL-10.

Our industry is certainly blessed with a fine cadre of interested, creative, and competent people. It's important to note that cadre is spread over industry, universities, and government.

I think the most important comment on Figure 2 is that there are a lot of generic technologies that can be worked on. We do not have to zero in tomorrow afternoon on a concept in order to know what to do here. There are a lot of things that we can proceed with intelligently at whatever funding level is there. Now, we all want more funding. We all know that if the pace is too slow, we miss the milestones, and the program is jeopardized because we can't produce. There is a lot of generic work to be done. Here is a list to start with:

Nozzle cooling versus Isp or versus temperature. As we think of some of these very high Isp systems, we certainly are taking on some nozzle issues.

There is a whole field of high temperature, hydrogen compatible, neutronically compatible materials. I don't think we can sit back and say, "As soon as you materials guys come up with unobtainium, we will use it." We have to work with the materials we have. However, there is certainly room for advances in both basic structural materials and coatings.

I don't think dissociation/recombination is a very expensive one to look at, but we need to know about that.

Control, operability, reliability, redundancy. I think redundancy should evolve, rather than be dictated.

There is certainly a great deal of work that can be done on radiation hardening of electronics. We need to look at valves, pumps, lines, flanges and instrumentation and see if we can make them hard, when we have to cool them, or put them behind a shield.

In addition to the generic studies, there is a long list of concept-specific technologies, too. Of course, fuels and controls are on that list.

Another thing on which we must comment is facilities (Figure 3). The major point I would like to make here is that we can start now. Facilities are fairly generic. I think we can start defining facilities and moving out. We do not have to wait to the point of knowing what concept we are going to test. Eventually, there will have to be some customizing of the facilities to make them concept-specific, but I really feel we can begin moving on facilities.

These facilities are going to require a significant fraction of our resources. It's going to take some pretty gutsy program manager to spend significant monies to get brick and mortar instead of technology for it. We are going to have to make those investments. One of the reasons it's so important is that these facilities are the pacing item for the systems tests, and these systems tests develop the bright and shiny products that we have to produce. We have to have bright and shiny products popping up fairly often throughout the program. At least those products have to pop up every time that the membership of our space committees turns over, and every time the administration turns over. The other thing that paces the facilities is the approval process. There is no reason why we can't start that right away. Certainly we are committed to full-scale ground tests and unmanned flight tests of the system.

We need to define our targets of opportunities and figures of merit (Figure 4). The right thing to do is very dependent on what you want to do, when you want to do it, how much you are willing to pay to do it, and how versatile you want to be.

MASE made a list, which isn't complete. Are we going to optimize mass to low Earth orbit? Are we looking for the highest Isp, mission versatility? Are we trying to find a program that fits some cost profile that we think might be out there? Are we looking for launch date/availability? (what if we want a flight demonstration prior to 2005, before some decisions are made about how we are going to go to Mars). I presume if I were the program manager for manned mission to Mars, I certainly wouldn't select a propulsion system that didn't have a lot of data on the table.

Do we want to optimize the schedules or do we want to optimize compatibility with the infrastructure? Or, do we want to recommend that the infrastructure be customized to help us maximize our capability of delivering payload?

It's going to be difficult to make selection without a somewhat clearer picture of what it is we are trying to optimize.

I said we were short on propulsion systems information. When I realized this, I tried to list some of these issues (Figure 5). As I did this I discovered that I don't have a clear understanding of what those issues are either.

These are the questions for clustering -- is it a good thing to do, does it add reliability, does it reduce reliability, do you cluster, do you have the same number of engines as you do pumps or is there a different number of engines and pumps, are there more pumps than engines or engines than pumps?

How important is size? We had one concept which was very loyal to the size constraints, and we could see how those size constraints really caused a lot of ramifications in that concept. We have to decide how loyal we have to be to size constraints.

Other issues include redundancy and disposal modes. I didn't hear anybody talk much about disposal. Similarly I didn't hear what our Earth launch constraints are. Man rating: the list goes on. There are a lot of systems issues that we really haven't dealt with, and they are probably going to be rather important to our designs.

To summarize, again the most important thing to say is nuclear thermal propulsion can work (Figure 6). It can work in some of our lifetimes. It can work with budgets that are affordable. We have the intelligence, we know how it can happen.

The technology work cannot get ahead of the safety work. We have to work the safety issues in parallel. When our nation asks, "Do you know they are thinking about nuclear propulsion?" we better be able to say, "Yes, and here is what we are doing to make it safe."

The final point is we need to provide space transportation, not just a propulsion system, to get men on Mars in the year 2016. We need to avoid getting trapped into a focused target. It's bound to change many times while we are working on it. Another way to say that is that we can do parametrics.

We better have some intermediate targets prior to man on Mars, and I think those targets need to be spaced every five or six years. I think it's very difficult to establish a program that's 35 years long or even 25 years long.

The reactor group found a lot of commonality with the other groups, but one thing that we noted is the reemphasis of what we found at Pasadena; that we still need to pursue areas of fuel development, coatings, and high temperature materials (Figure 7). I, too, think there is enough commonality in these areas that we can start that now and really accomplish something.

Now, for this particular workshop we found the additional need for high temperature hydrogen effects (Figure 8). At the NEP Workshop we were primarily concerned with some of the other coolants such as helium and xenon. But hydrogen brings in some special problems that need to be looked at.

We think there is a lot of advantage to starting immediately to look into hydrogen physics and chemistry regarding the dissociation and recombination of hydrogen, the material in addition to interaction of the hydrogen chemistry and the kinetics.

In the fuels area, we need to investigate the basic fundamentals of fuel behavior as well as the chemistry and kinetics of various compounds. We also need to get samples of existing materials and plug them into reactors.

We also need to get some basic policies and guidelines as soon as we can (Figure 9). One particular one we picked out is fission product release. There were several concepts (especially vapor core, liquid fuel or foil reactor) where the design puts fission products or fuel out the rear end. Now, is that going to be allowed? Some policy decision needs to be made because that is going to guide the concept decisions and design. Even with the contained fuels, should we not have any release except maybe in emergency circumstances? We need a recommendation as soon as possible (Figure 10). We need to get a consistent set of requirements in the areas of mission, technology, safety, and policy.

In summary there are a lot of concepts that we could probably start tomorrow. There are lots of opportunities for optimization and improvement, but at least the basic concepts are there. It is very encouraging.

General Comments

- o NTP can work
  - "Reasonable funding"
  - Options available
- o Short on systems level information
- o Need to provide substantial design margins
- o Industry blessed with cadre of interested/competent people
  - Industry, Government, Universities

Figure 1

Technologies

- o Generic:
  - Nozzle Cooling vs. Isp
  - High Temp., Hydrogen Compatible/Neutronic Compatible Materials
  - Dissociation/Recombination
  - Control
  - Operability
  - Reliability/Redundancy
  - Radiation-Hardened Feed Systems
- o Concept Specific:
  - Fuels
  - Control

Figure 2

Facilities

- o We Can Start Generic Facilities Now!
  - Requires Significant Fraction of Resources
  - Passing Item for Systems Tests ("Bright & Shinny Products")
  - Facility Paced by Approval Process
- o Committed to "Full Scale Ground Tests"
- o Committed to Unmanned Flight Tests

Figure 3

Define Targets of Opportunity/FOM's

- o What are we Trying to Optimize?
  - IMLEO
  - Isp
  - Trip Time
  - Mission Versatility
  - Cost/Cost Profile
  - Launch Date/Availability
  - Compatibility with Space Infrastructure

Figure 4

Propulsion System Definition/Critical Issues?

- o Issues

    - Clustering
    - Size
    - Redundancy
    - Disposal Mode(s)
    - Earth Launch Constraints
    - Man-Rating

Figure 5

Summary

- o NTP CAN WORK!
- o Technology Work Cannot Get Out Ahead of Safety Work.
- o Need to Provide Space Transportation

    - Not Get Trapped into Focussed Target That Will Move/Change Many Times-Do Parametrics!

Figure 6

Reactor Technology

- o Re-emphasis on Need for:
  - Fuel Development
  - Coatings
  - Materials
- o With Additional Effects of:
  - High Temp H_2

Figure 7

Reactor Technology

- o Fundamentals on:
  - H_2 Physics of Disassoc/Recomb
    Material Interaction
    Kinetics
  - Fuels

Figure 8

Reactor Technology

o Concept Require Policy/Guideline Decision Criteria on Fission Product Release -

- Vapor Core
- Liquid Fuel
- Foil Reactor
- Contained Fuels

Figure 9

Reactor Technology

o Recommendation:

- ASAP

o Development of a Set of Requirements which is Consistent Between:

- Mission
- Technology
- Safety
- Policy

Figure 10

WORKSHOP FEEDBACK

Advanced Development Panel
Steve Howe

We had a number of wonderful presentations and we wanted to point out a couple (Figure 1). Dave Buden talked about safety on reactors, and Bob Rohal also addressed safety. Even though we weren't the safety panel, we wanted to get across the idea that it is an integral part of facilities and testing and has to be included in a fundamental way.

Darrell Baldwin gave us a brief rundown on facility and testing issues. I want to quickly run through the concepts that have been developed by Dick Bole and Don Hansen at Los Alamos in cooperation with Rocketdyne. We have four different ways of treating effluents, if you will.

The basic idea on the first one is you are ejecting the exhaust products into a large holding tank that captures all the effluents. You valve it off when it reaches pressure, and you scrub it at a low rate so it's a closed volume containment.

The second concept was flaring of the scrubbed hydrogen. Hydrogen comes into a cooled pipe that has a sump, and you trap the fission products and flare the gas into a flame holder. The whole point here is to give you a feeling for what magnitude of facility is being considered to do a safe test of the integrated test facility.

The third concept, is the same thing except you are injecting liquid oxygen in and actually combusting the hydrogen and oxygen prior to the scrubbing. This gives you a little smaller facility downstream because of the hydrogen volume but gives us a lot of energy coming out.

The last concept condenses all the effluents. I'm not sure what the fundamental differences are between trying to cool it down, and condensing it in line.

Our intention during the presentation was to look for discriminators of each concept.

Integral test facility impacts include physical size and geometry for the large reactor needed for the foil concept (Figure 2). It also probably releases a higher inventory of fission products. A uranium accident scenario has to be treated carefully for liquid cores.

Gaseous core has a high fission products inventory.

In a dual concept, if you require a dual-mode reactor to test both operations in a single facility, it may result in a more complex and expensive facility. Is dual mode more expensive than doing two facilities (one to do electric and one to do thermal)?

As far as cost, we essentially concluded there was no concept presented that had an estimated cost that was less expensive than the baseline. The committee says $2 to 5 billion, which has been the number that has floated around the country for many months and we didn't see we could impact that.

We want to absolutely require a nuclear furnace or core driving type system for fuel development, and in all concepts we see fuel development issues to be a major component that must be pursued.

I want to make some comments about safety. Our feeling was that you must develop a safety-oriented mindset in the designers right at the beginning. They have to have that incorporated in their thinking from the start.

Following on the safety a little more, there was some concern that if we were going to make this a national program, we would encounter safety criteria or procedures defined by three different agencies (Figure 3). We must have some type of integrated safety review process among the players.

The nuclear furnace core driver should highly flexible so that it could test fuel elements or components of as many concepts as possible.

On the other hand the integral test facility may require multiple test stands, so you might consider less flexibility in the initial test stands. Then, as you evolve your concept, you may build different test stands.

Then we came up with the high temperature concepts. Do you need a full nozzle test facility or can you simply demonstrate a heat flux density at the throat and not have to do a full nozzle evacuated chamber?

The clustering issue came up dramatically. I think that the pro side of the clustering issue is that if you can make smaller engines that are easier to test, demonstrate life and relax some of the reliability problems, you will have to deal with the safety panel because you cannot have multiple engines.

On the negative side of that, if you are going to have multiple engines, do you have to do a full cluster test? That would make the facility big and tough, so that is an issue that has to be addressed very early by some appropriate decision making body.

The dual mode should also be addressed very early because these are big impacts in the test facility.

It's our recommendation to start the paperwork as soon as possible on both the ITF and the driver core facility, so you can hit the pavement running (Figure 4).

We also felt you should begin basic research and development to try to answer some of the basic questions in the concepts you have heard. There are some critical experiments that can be done on laboratory scale, and they should be started as soon as possible.

One thing that we also want to recommend that oftentimes is left in the dust is instrumentation and diagnostics. The point here is that you want to know what's really happening in your test facility. You also want to have some real-time adaptive controls on these reactors so you better have real-time instrumentation to feed back into the controls.

At 3000 degrees Kelvin, the temperature measurements are tough. There is a whole major effort that must be conducted for instrumentation and diagnostics.

As a final point, I contend that the second year of the second term of a president is an optimum opportunity to do something a little risky. And so I am saying that the year 2002 is a very large window, if you wanted to do an unmanned demonstration of nuclear propulsion to an outer planet system.

SUMMARY

o Safety Talks

- D. Buden
- R. Rohal

o System Testing Issues

- D. Baldwin

o Tried to I.D. Discriminators of Concepts Based on Impact of Integrated Test Facility (ITF)

Figure 1

"CONCLUSIONS?"

o ITF Impacts

- Foil - Size, Fission Products
- Liquid - Uranium Accidents
- Gas - Fission Products
- Dual Mode - Complexity

o Cost

- No Concept < Baseline
- Some Concepts = Baseline
- Poll Results - $2-5B

o Absolutely Require NF/Driver

- For Fuel Development/Testline

o Require Fuel Development in all Cases

o Must Develop a Safety Oriented Mindset From the Beginning and Work with Designers Early on.

Figure 2

CONCLUSIONS (Cont'd)

- o Need a 3 Agency, Integrated Safety Review Process
- o NF/DRIVERS Needs Hi Flexibility
- o ITF May Require Multi-Test Stands, i.e. Evolution With Concept
- o Nozzle Test Facility?
- o Clustering Should be Assessed Early
  - PRO: Smaller, Reliability
  - CON: Cluster Test?
- o Dual Mode Should be Assessed Early

Figure 3

Recommendations

- o Start Paper Work on ITF and NF/Driver ASAP
- o Begin Basic R & D on Fuel Development ASAP
- o Support Strong Effort in Instrumentation and Diagnostics

Figure 4

WORKSHOP FEEDBACK

Safety Panel Summary
Charles Sawyer

As other people have already commented, we need some policy decisions regarding safety (Figure 1). There need to be joint agency policies dealing with safety.

We would like to see some policy (Figure 2) on the kind of protection we need for orbital or space debris and meteorites. Where do you shield when you get on the lunar surface or the Mars surface? How do you protect the astronauts as they get out of the vehicle and move all around while the reactor continues to work?

And as somebody else pointed out, we need an integrated review process -- initially one that helps at the beginning looking at the concepts and initial designs, helping to feed back information to the designers, so it gets related and then integrated into the continuing design.

We have already commented about the three agencies involved. Obviously before you come to a joint safety review panel or safety review board, each of the developers and contractors is going to have to his own procedure and process ready. One of three agency panels has said "Okay, we are going to press on. We know that we have to go through the instrument process." Part of it is in parallel. We have to get approval from the president as well.

Let's get the requirements up to the developers (Figure 3) and be prepared to help them interpret what those requirements mean. As the designers progress, some of those requirements are probably going to evolve and begin changing.

From the testing standpoint I want to talk about the documentation (Figure 4). First, determine the kind of testing you need to do and information do you expect to get from it. Then, you need to save the results, so that when you begin looking at trade-offs, you know what you have in your past testing.

The whole question of how much full scale and subscale testing you do and how much you can use simulation and analysis will also have to be reviewed. I think that it's important that developers anticipate doing full scale testing when it's a reasonable thing to do. If it gets to the point where you can do simulation and analysis, you save time and money, but to try to do it the other way will have significant impacts on the program.

Finally, the margins for the higher fuels and materials involved have already been mentioned. We need to be sure that we don't press temperatures to the point where you are melting the containers that are holding the reactors.

You need verification techniques so that you know what it will withstand as you are on your way to Mars. As things go wrong, you should be able to trace them and know that particular items can take whatever additional loads they need to pick up because of failures we have had as we have gone along.

Finally, in the public perception, what are the real versus the perceived hazards (Figure 5)? Those of the us in the safety community can certainly help put some of that together. We can also help both management and public affairs folks begin educational programs to try to make it a little bit easier. We can suggest some answers and some explanations to the questions that we know are going to be there.

Safety Panel

- o Requirements
- o Communications
- o Testing
- o Public Perceptions

Figure 1

Safety Panel

- o Requirements
  - Policy - Joint Agency
    - o Fission Products
    - o Confinement/Containment
    - o Reentry/Impact Response
    - o Orbital/Space Debris
    - o Shielding
  - Review Process - NASA/DOE/DOD/Others?

Figure 2

Safety Panel

- o Communications
  - Get Requirements to Developers
  - Be Prepared to Help Interpret

Figure 3

Safety Panel

- o Testing
  - Documentation - Plans Thru Results
  - Full Scale/Sub-Scale/Simulation/Analysis
  - Margins
  - Verification

Figure 4

Safety Panel

- o **Public Perception**
  - **Real vs. Perceived Hazards**
  - **Education/Public Awareness**
  - **Prepare Answers/Explanations**

Figure 5

WORKSHOP FEEDBACK

Concluding Remarks
Greg Reck

There have been a tremendous number of things occurring lately. At Albuquerque earlier this year, I reviewed what has happened over the past year or two regarding policy, the establishment of the Space Council, and endorsement by the President of the Space Exploration Initiative. It's been a very exciting period of time. I think we see more and more change on a daily basis. Certainly with all the activities and speeches by the president and vice-president, Congress, members and executive secretaries of Space Council, it is very clear that this is a high priority to the administration. I think they have done everything they can up to this point.

Now that we have an endorsement, the next step is to figure out the right way to get the support we need. Of course we have been looking at the alternatives, and the study that we at NASA conducted last fall was a part of that. We called it a 90-day "study," but really it was a summary of a lot of activities that had been underway for a number of years in the Office of Exploration. One of the most important things that came out of that was critical technology. There was a long list of technology needs that were identified. There were seven technologies listed, and three of those were directly related to your efforts and your activities. Nuclear power, nuclear propulsion and radiation protection were on that list.

For the first time in a very long time, there was recognition at NASA that nuclear systems had a key role to play in the future. For many years, we have argued and lobbied hard for electric propulsion and other elements associated with nuclear space systems but it was falling on deaf ears. Now there is some recognition of the significance and benefits that could be derived from nuclear systems.

I think some of the comments earlier today on systems aspects are crucial. We have to make sure that we continue to highlight and identify the benefits of these systems, whether they be initial mass or trip time. Both of those are certainly key driving factors that all of the mission studies up to this point have utilized in trying to select technologies. I think that will continue to be the case.

My perception is that trip time is probably the lead motivation for pursuing the nuclear systems. The promise of a much higher Isp that might get you there faster to minimize the risks associated with radiation exposure, zero gravity, isolation and long periods of time in threatening environments are all big factors.

But right behind that, of course, is initial mass. Initial mass directly affects cost, which is one of the biggest hurdles to cover in trying to make this thing work. For the first time, within the last year, there has been a national level recognition of the fact that we need to determine what the architecture is going to be.

For the past couple of weeks in Washington, Tom Stafford has been setting up his office, bringing in the kind of people that he expects he will need, collecting information and sifting through information, trying to identify all the architectures that represent reasonable ways of doing the job. And I think over the next few months he is going to be contacting a lot of you, looking for ideas. The time line for that is four to six months.

I heard a lot of discussion here about requirements. We need to know just what the real requirements are. We have to know how many days we are going to be there, and what the trip time requirements are. That information is not going to be available in the very near term. We are going to have to work hard for a couple years to sort through all of the possibilities and all of the options.

I am not sure that any one system or any one approach is going to turn out to be the winner. I think it may be a combination of chemical, nuclear and aerobrake. All of those have to be traded off against each other. I think that we have to look at the longer term.

Remember we were able to get to the Moon and able to stay during the day but we couldn't stay overnight. I think what we need to do this time around is put the kind of systems in space that we can use over and over again economically. That's a very important factor to pay attention to if we are thinking about a longer term program.

With regard to what is happening in Congress right now, the first round wasn't very good. The House Appropriations Committee identified not only the new monies that we had requested just for the exploration activity and zeroed those, but also went back into ongoing programs that we had in place for several years. We had identified efforts, ongoing activities and efforts that related to and supported SEI and they also targeted those for significant cuts. So, some of the activities that we have had in place for several years, which are reaching experimental phases, could be cut back or terminated if nothing changes in the budget figures.

That's very unfortunate. But there is some encouragement on the Senate side. Some of the difficulties that we are experiencing in the shuttle program and Hubble are probably not helping matters, but I think people recognize that those are shorter term problems and that they will be solved. What we are talking about in SEI is the long term.

I believe that the Senate markup has been postponed; we expected it might occur this week or next. As I understand now, they may even try to extend that, give us a little more time. I think that's encouraging. I think the administration is working hard to try to bring us a markup so we can at least come out of this sustaining the ongoing programs and providing some seed money to get the technologies and studies on track. I think it is imperative that we get those technology activities underway.

It is clear that there are some specific areas that we need to invest in now to help understand just how effective the nuclear propulsion or nuclear propulsion options are going to be. We need to find out what sort of payoffs we will get from them. Those technologies need support and we are working as hard as we can to try to ensure that they at least get a start.

The final point is that this is a national program. Head-of-agency meetings have taken place to discuss in general terms how NASA and DOE are going to work together for the program.

The same sort of meetings are taking place with the Department of Defense. I have seen at least a draft of the MOU that will be established between NASA and DOE. It's an umbrella kind of agreement that will be very broad in character and will not be specific in terms of technology. Eventually, an agreement will delineate exactly how we are going to work together. That process has started.

In some cases we have programs that have been established a number of years ago. The power area, for an example, where we have worked very effectively with the Department of Defense and Department of Energy. I would expect that those kind of programs would continue and that other mechanisms may be introduced to make sure that the collaboration is close.

Meetings like this, where we get all of the groups together and talk about what needs to be done are the best places to do it. I am really impressed by the organization. This workshop and what I also heard of the workshop in Pasadena were very impressive. It looks to me like you have dealt with all of the right issues. It's going to provide a strong board to move on out and develop the program.

I guess I am still encouraged that when we come out of the budget exercise, there is going to be at least some money available to get started on the study efforts that really need to take place.

There have been a lot of changes. We are very hopeful that we are at the front of an exercise that is going to take off in the next year or two.

A VOICE: Has there been any discussion, planning or thinking about a serious cooperative program with the Soviets.

MR. RECK: There has been a lot of discussion and right now we are waiting for guidance from the Space Council on how we approach that. Many people have suggested nuclear propulsion/power, human endurance, behavior and medical effects in space.

The Soviets have a lot of capability and a lot of experience. Certainly you can list a number of technology areas where there is potential benefit from a joint endeavor. I think that one of the motivations of the Space Exploration Initiative itself is its international character. You can really bring the world together in this kind of undertaking. The president has already directed the Space Council to study and develop a policy for international cooperation.

It's clear that there is a desire to bring other countries into the program and to truly make it an international activity, but until we really get guidance and information from the Space Council, we won't officially make those kind of overtures.

A VOICE: Along that line, how much do we know about what the Russians are doing in the nuclear area?

MR. RECK: With regard to power, the Soviets have participated in the last two out of three Space Nuclear Power Symposiums. There have been visits from senior NASA officials and I believe the other agencies have been involved in visits to the Soviet Union, to their facilities. There have been opportunities to see a lot more over the past year or so than we have in the past several decades.

In the life sciences, there are already working arrangements with the Soviets. We all know they have a lot of experience with operating for long periods of time in space. In the science area in general, there has been a lot of effort to try to set up a joint collaborative space science mission with the Soviets, where we would both fly each other's instruments in the future.

In the propulsion and power area, there hasn't been that kind of exchange. As soon as we have the endorsement of the Space Council, we will be prepared to put those kind of things in place with the Soviets, as well as the Europeans.

WORKSHOP FEEDBACK

Summary
Gary Bennett

Tom Miller said last night, "I hope I don't hear the word workshop for five more years." And someone else said, "you will hear it until we get money."

But these workshops are more than just assembling a data base. I anticipate that a number of us are going to spend time over the next year developing and promoting nuclear propulsion and we need the kind of information that you are providing us to explain the benefits of nuclear propulsion. I want to echo some of the comments that were made.

We do have to maintain a certain degree of flexibility on these concepts. I have talked to the people at Johnson Space Center and they have repeatedly told me it's going to be several years before they are in a position to define how they want to go to Mars. Nuclear is a candidate, along with chemical plus aerobrake, so we do have to maintain a certain degree of flexibility.

I also want to remind you of what we were trying to do with the workshops and where we plan to go. We put together a Steering Committee in May consisting of Lt. Colonel Lenard, Earl Wahlquist from DOE and me working with the panels. The idea was to assemble a data base. I think we have a good start.

And we have had the two workshops. Those of you on the panels are the technology review panel, and over the next month or two, we are looking to the members of the panel to put this down in written form.

The intent is that in September, the technology review panel will be briefing the Steering Committee on the results of the two workshops. Then, we hope to have a general feedback meeting later in the fall with all of you on how all of this went.

I want to again use Dick Bohl's viewgraph to echo something that Colonel Worden said in the banquet talk Tuesday night. We all need to work together on this.

There were some general things that I think we can look at. Let's assume the worst case -- if we don't get any money in 1991. I still think there are things that can be done.

You heard for example from a number of panel chairmen, and specifically from Buzz Sawyer and the safety panel, that there are issues out there on safety philosophy and safety policy. We could use that time to get together an interagency group that could work with OSTP, NRC, EPA, you name it, to come up with some general guidelines on safety.

The big question is "Are we going to be allowed to release fission products into space?" There are people in Congress and elsewhere who have strong opinions on that. We need to look at them rationally and see what the risks are, what the payoff might be to have an advanced concept that might do that. We could get no funding and still move forward on safety philosophy.

Another area is testing. We have discussed full up tests and certainly in the wake of Hubble, one could argue that full up tests are certainly going to be required. We could wrestle with the issue of deciding exactly what full up testing means. We could also decide how to build a facility for solid core, but then also have the capability to go to some advanced concept later.

One thing that came out of the Pasadena workshop is that, at least now, the reactor people know who the electric propulsion community is and also the electric propulsion people now know what reactor power sources are out there and available to them.

Hearing about all the different concepts and the different thought processes has at least been educational. I hope it has been educational to all of you, so that we can come up with systems that get us on a "level playing field."

Back to safety, I agree we have got to get it in early as several people noted. However, let's make sure that the issues are kept at a more general level. For example, I don't think we should "require" two emergency core cooling systems which will be independent, etc. That has plagued other systems, where people have developed requirements in the dark and passed them down to contractors. The contractors have done crazy things trying to live with them. I don't want to get into that mode. I would like to get some creative thinking out of people.

Another thing I think we can do over the next year is recapture our history. For example, on the nuclear light bulb, I recall Tom Latham saying there are about 160 reports out there. I don't know where they are, but if we could find them, we could at least write up a good technical summary.

Something occurred to me when several of us went out to the Nevada test site to look at the Nuclear Rocket Development Station. We ought to get somebody with a video camera to interview the guys who worked on the gas core, liquid core, and all the solid core concepts and get an oral history of why they did what they did. If we don't do that, we will lose a lot because the documents don't always explain some of the things that were done and why the plumbing went one way and not another. That's something that some charitable NASA center or DOE lab could possibly do next year, if we don't have any money.

And finally, I want to emphasize that the national space policy says we are expanding human presence and activity into the solar system; we are not stopping at Mars.

Therefore, we should build flexibility and growth capabilities into the program. We think about these nuclear concepts now because we are going to Mars. But suppose we want to go to Ganymede or we want to do something else in the 21st century, what extra capability could we put into this now that would allow us to do it at not much additional cost?

A slow ramp-up may be a blessing in disguise, if it allows us the time to think these things through to meet the requirements that are going to come out in a couple years from the MASE team. I would like to have good designs that hold up and don't require last minute changes.

WORKSHOP FEEDBACK

Closing Remarks
Tom Miller

Three years ago, in the fall of 1987, the Office of Exploration (headed by John Aaron) was established to look at what we might do to explore the Moon and Mars. Do you go to Phoebus first? The Moon first? Do you exploit in situ resources on Phoebus? How do you do that?

All of the centers were researched. Our organization at Lewis attended the first meeting out in Denver. That was an organizational meeting. We at Lewis were looking for a way to be involved with that process. So we ended up being a participant as Special Assessment Agent for Power and Propulsion. That was in October of 1987.

One of the first things we did was started digging out all of the old reports, looking at the good work on alternative systems that had been done many years ago. We started talking more seriously about nuclear systems and, in fact, sold the Office of Exploration on studying advanced systems including nuclear systems. The folks in the Office of Exploration thought that was a pretty good idea because they wanted to encourage innovation.

About a year later, in October of 1988, we went down and spoke to the Division of Power, Propulsion and Energy at NASA headquarters. We told them what we were doing, and it became clear that the organization that was interested in advanced technology wasn't really a supporter of nuclear propulsion. But I give headquarters a lot of credit because as they have seen the Space Exploration Initiative continue, they have recognized the advantages of nuclear systems. I think we are going to see a lot more emphasis on trip time and I think the nuclear systems are a way to solve that issue.

We continued to study propulsion systems, and as we know today, the nuclear thermal system with about 850 Isp is comparable with a chemical aerobrake at about 13 percent mass fraction for aerobrake. That's pretty interesting because I think we can maybe do better than that with nuclear systems. In fact, it is a challenge.

After the study review, we created the opportunity to hold these workshops. We have had approximately 350 individuals participate in these two workshops. It's been a very rewarding experience for me. I have seen a lot of bright people, who have a lot of enthusiasm for the field, and I appreciate that.

Finally, to go forward we told you when we started this process, we were going to use this information as input to a project plan and to the steering committee. We have drafted the project plan and put the draft plan in place for FY 1991. We will be reviewing the plan and updating it based on our input from this meeting. I think things are becoming clearer for us. We will also summarize that information for the steering committee and provide some feedback to this community in the fall.

Nuclear Thermal Propulsion Workshop
Cleveland, Ohio
July 10-12, 1990

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13. ABSTRACT ***(Maximum 200 words)***

The Nuclear Thermal Propulsion (NTP) Workshop, co-sponsored by NASA, DOE, and DOD, was held in Cleveland, Ohio on July 10-12, 1990. The purpose of the workshop was to review as many NTP concepts as possible, evaluate their current state-of-the-art, and discuss development requirements - to provide a database from which to develop NTP Project Plans. Most of the results and plans discussed were from earlier studies. In many cases the work was done during the ROVER/NERVA era (i.e. 1955-1972). A Concept Focal Point (CFP) was selected to represent each concept at the Workshop. The CFP was asked to describe the concept, discuss its safety and performance characteristics, technology development activities to advance the concept to Technology Readiness Level 6: (TRL-6 - full system ground testing complete), and present a "first-order" development cost and schedule for the concept. Technical Review Panels (TRP) were established with recognized national NTP experts to: provide a consistent comparison of the concepts, outline strengths/weaknesses, and provide a "first-order" ranking of the concepts compared to a NERVA reference engine system. The presentations of each of the Concept Focal Points (as were the additional presentations) were transcribed and then edited for clarification for this Proceedings. No new material has been added to the resulting papers except a bibliography for each concept. The final presentations by the Technical Review Panels, while preliminary, were left mostly unedited, so as not to change the intent or content of their presentations. The preliminary conclusions of the Panels included: (1) Technology development and safety must be pursued in parallel. (2) The highest priority for technology development is in high temperature fuels, coatings, and materials. However, the desire for high temperature (high performance) must be balanced by man-rating

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requirements - reliability, substantial design margins - and must include test verification. (3) Integrated conceptual design studies are required to properly evaluate all of the concepts proposed. (4) Other technologies to be developed include: high heat flux nozzles; controls, instrumentation, and diagnostics; and feed systems. (5) Facilities are also the highest priority and activities should be initiated now! A nuclear furnace will be required to test a number of fuel element concepts. An integrated, full-system test facility will be required, probably with multiple test cells, to verify technology readiness. (6) The cost of the technology development program is estimated to be between $2 and $5 billion. (7) Fundamental research should be conducted early to understand potential benefits of hydrogen dissociation and recombination. (8) Innovative concepts (gas core, for example) may lead to substantial performance improvements and should be included in the program. A number of issues were identified, but not resolved including: engine clustering requirements, engine size, redundancy requirements, disposal modes, Earth-to-orbit launch constraints, and "man-rating." Policies and guidelines will be required for: fission fragment release, an integrated safety process, and mission requirements. Finally, it was recognized that a proactive public perception/awareness program should be incorporated in the Project.