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NUCLEAR THERMAL ROCKET WORKSHOP REFERENCE SYSTEM -ROVER/NERVA-

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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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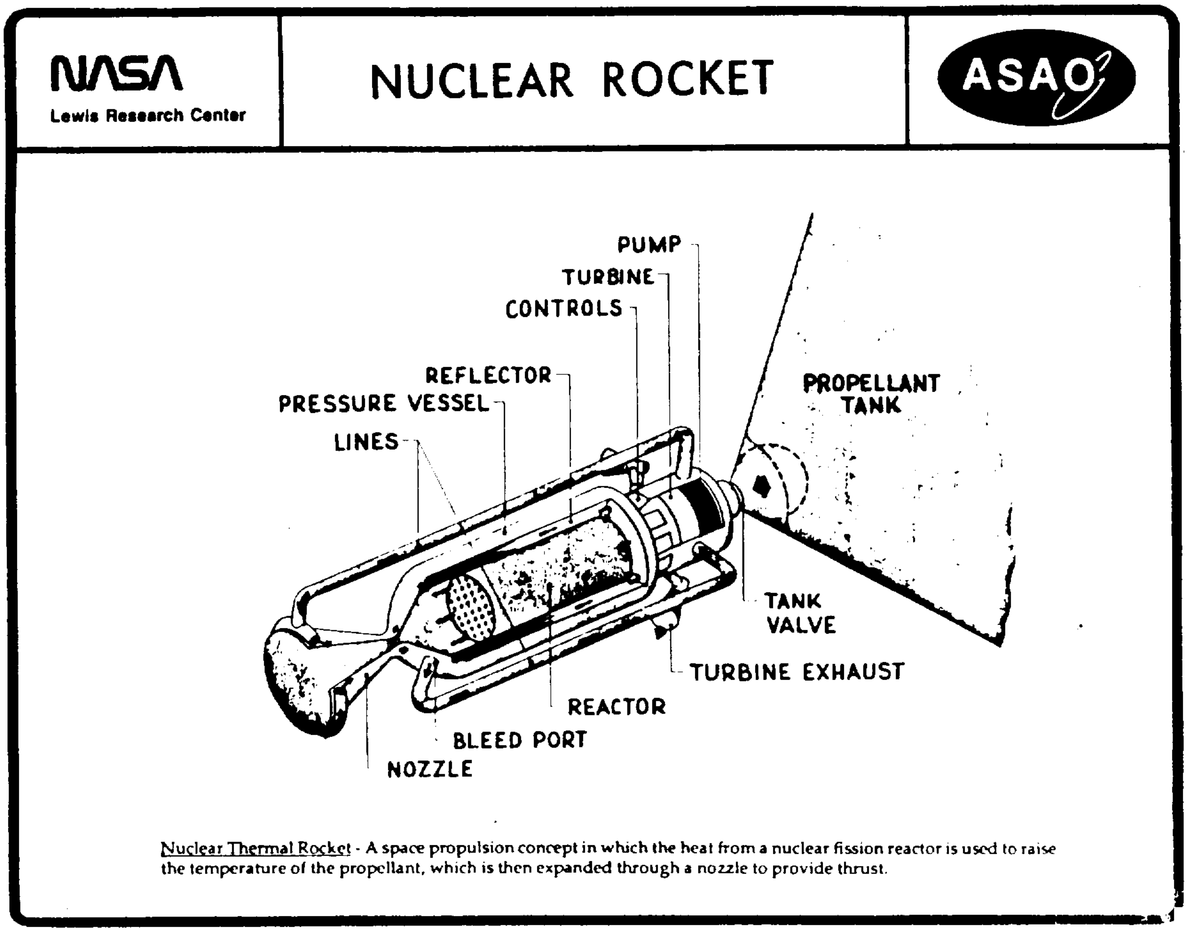

Figure 1

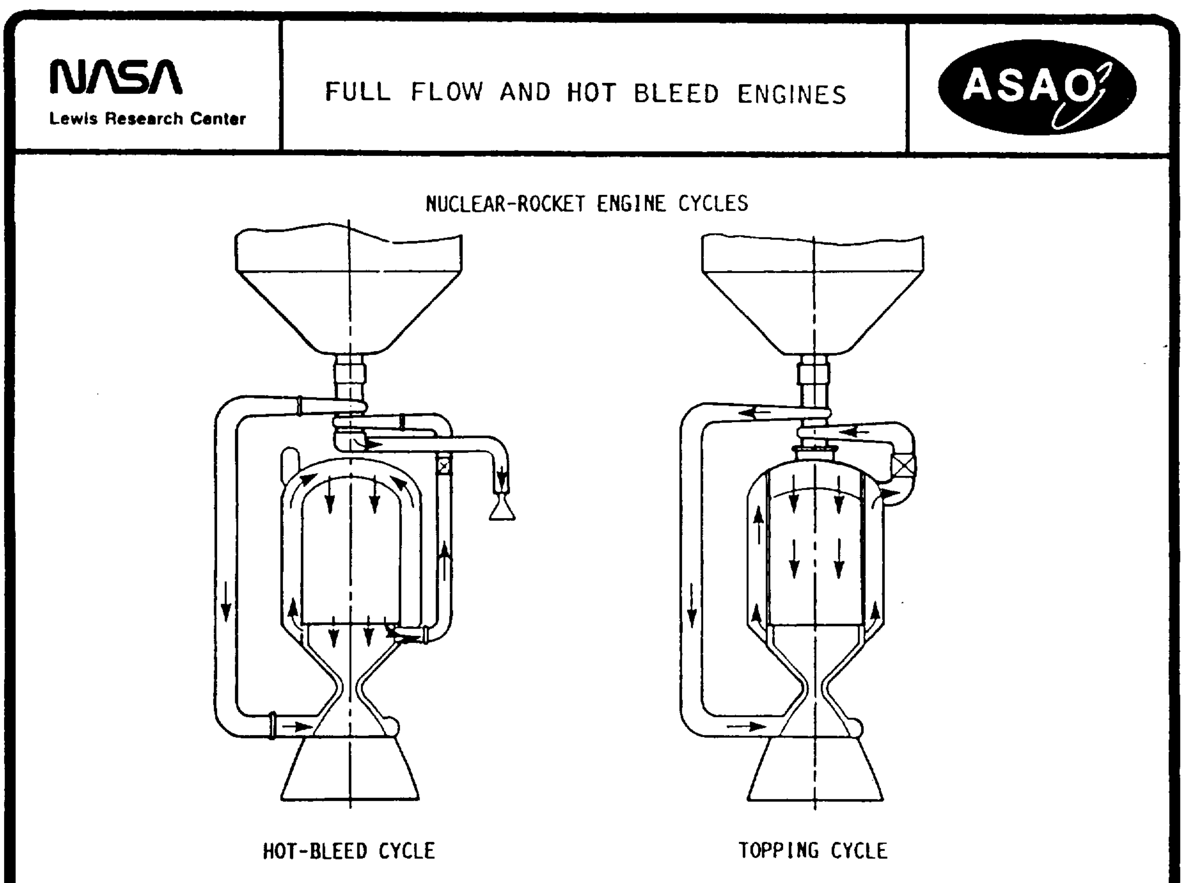

Figure 2

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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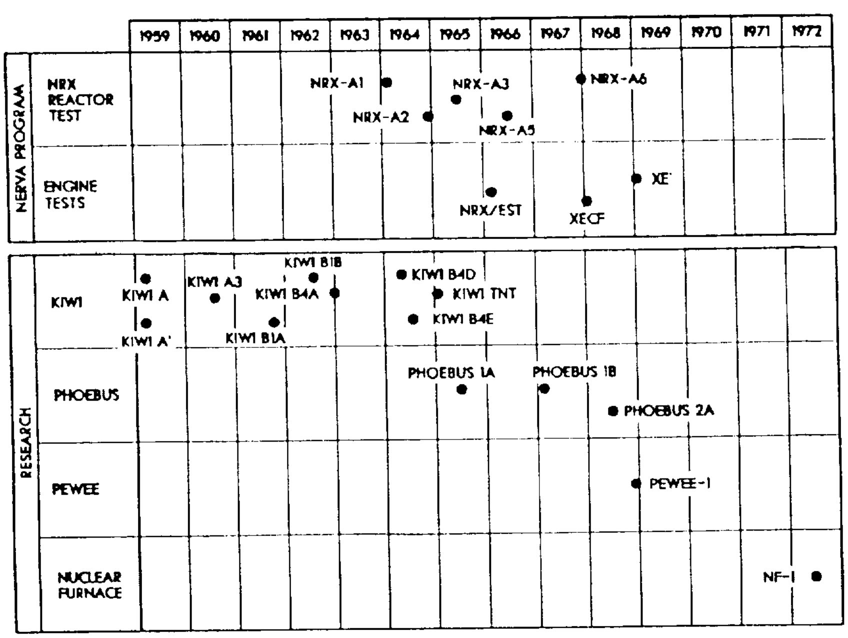
CHRONOLOGY OF MAJOR NUCLEAR ROCKET REACTOR TESTS

ASAO



REF: LANL

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Figure 4

REACTORS TESTED IN ROVER PROGRAM (LANL)

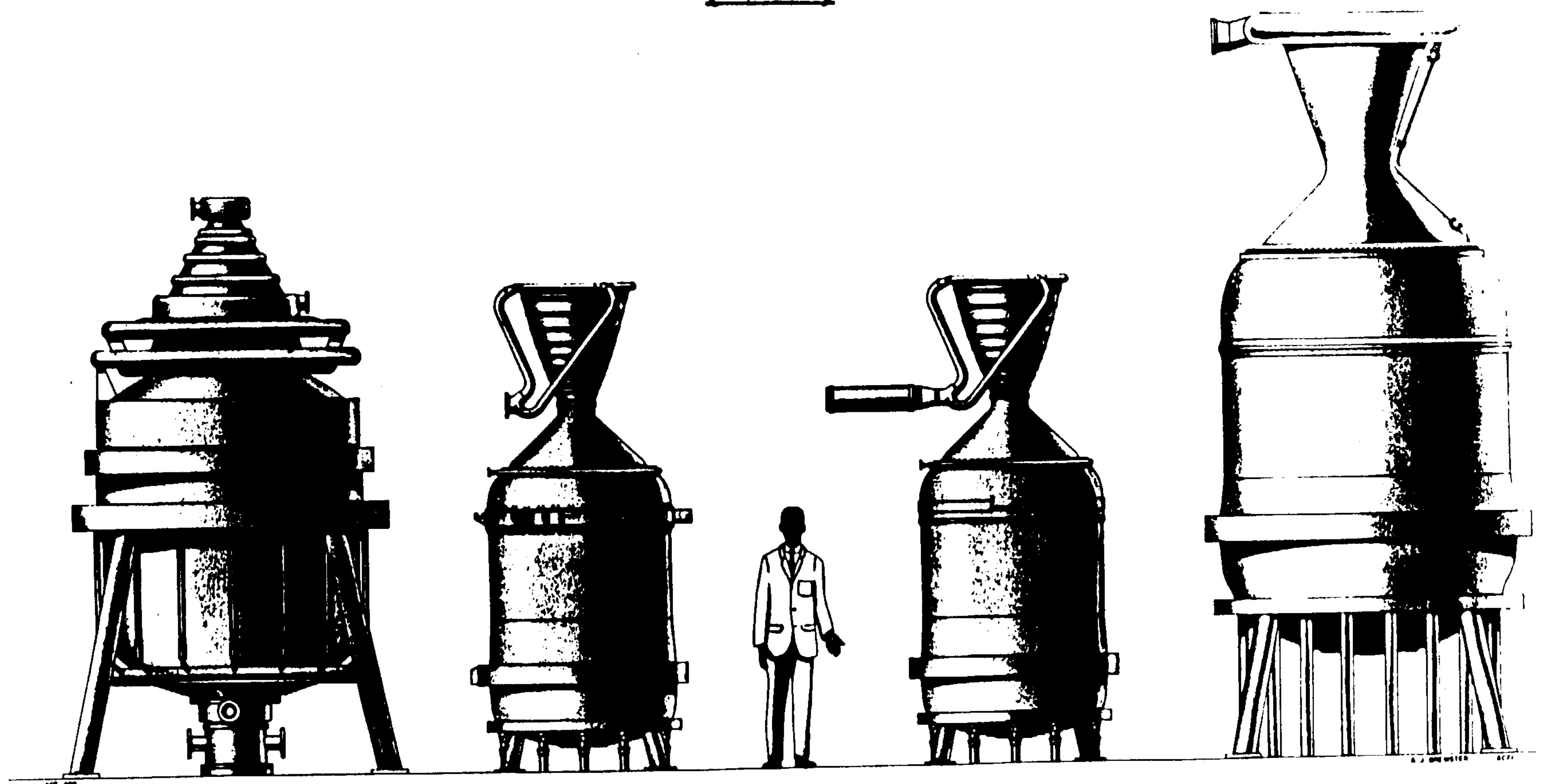

Figure 5

Phoebus 2A in Transit to Test Cell C

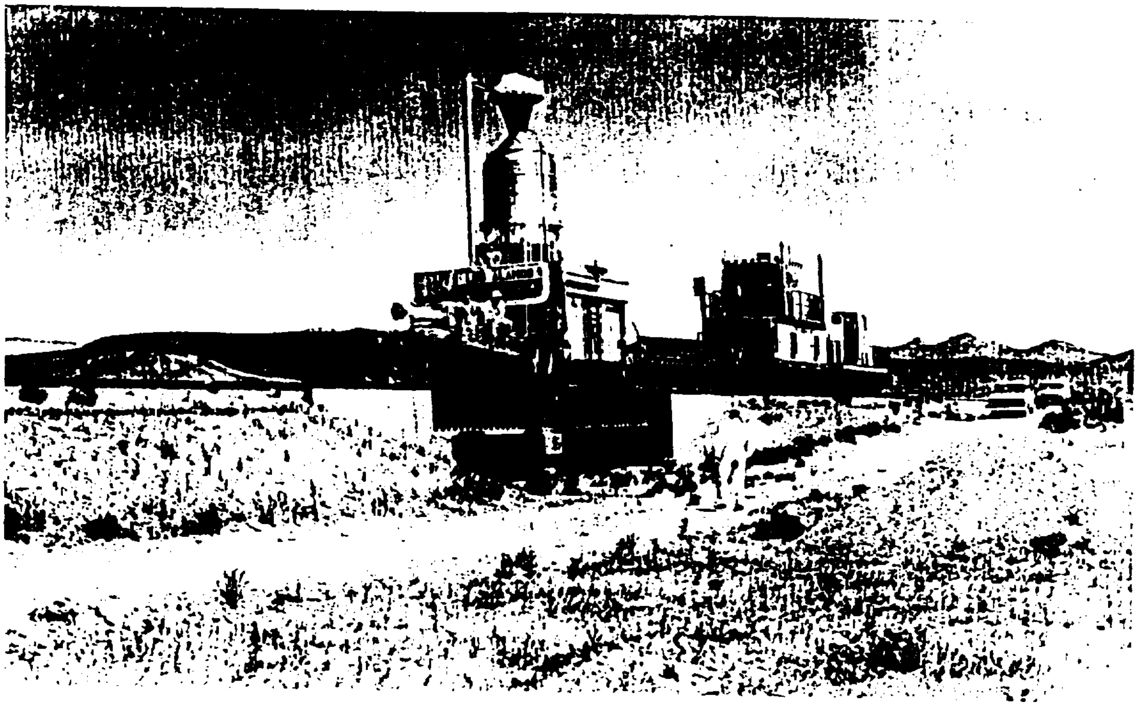

Figure 6

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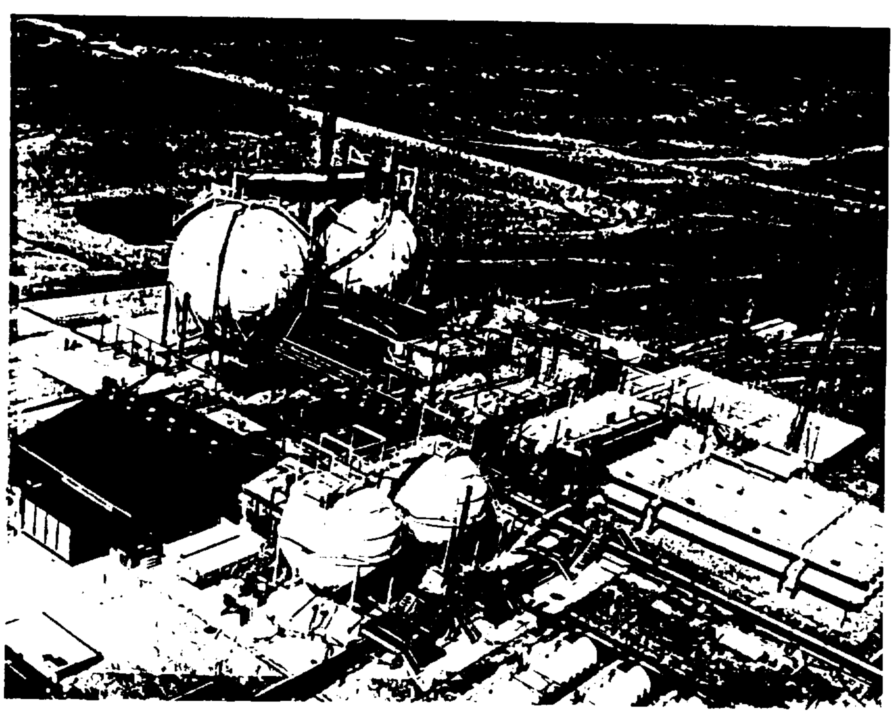
Test Cell C



SPACE EXPLORATION INITIATIVE OFFICE

Figure 7

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

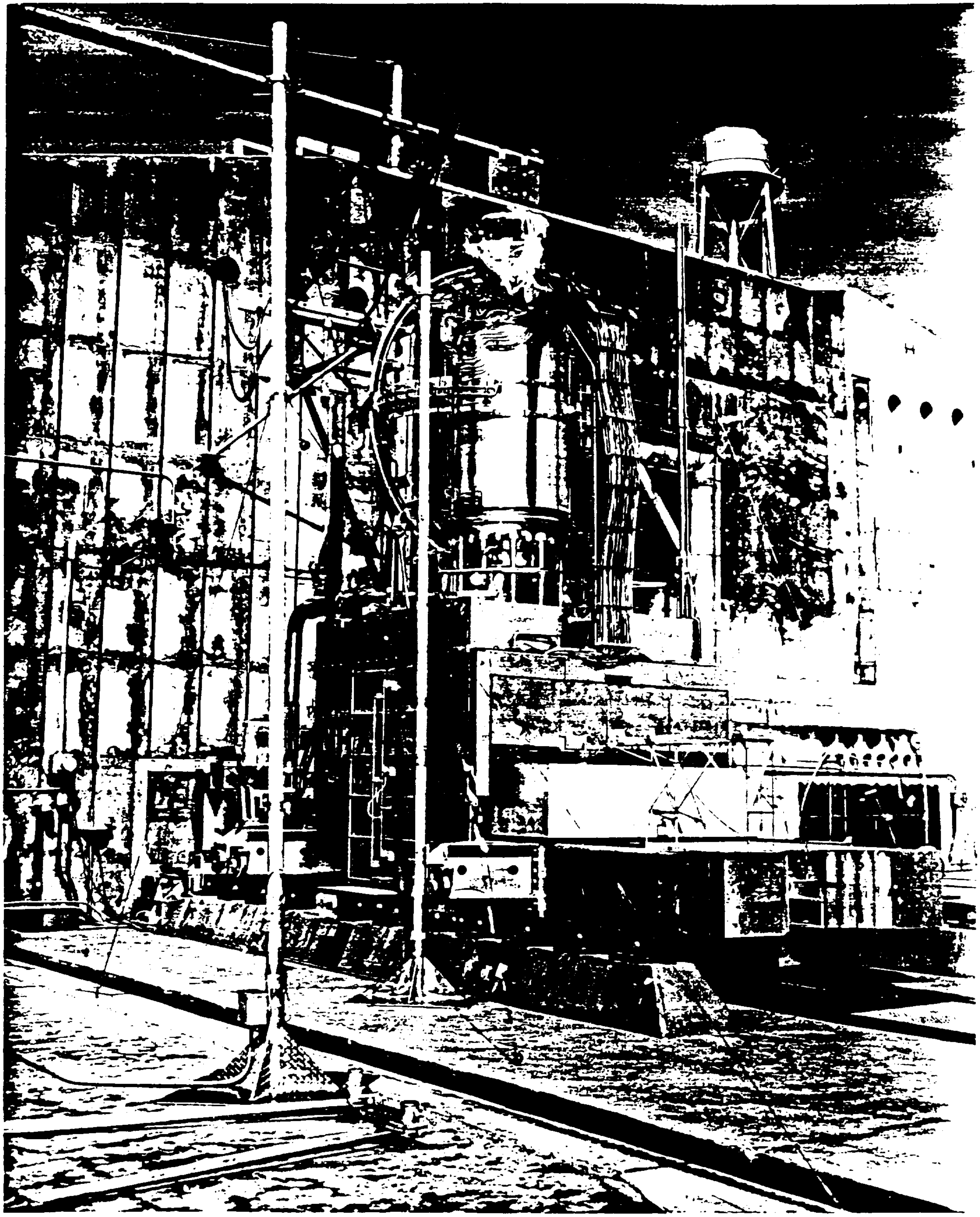


Figure 8

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

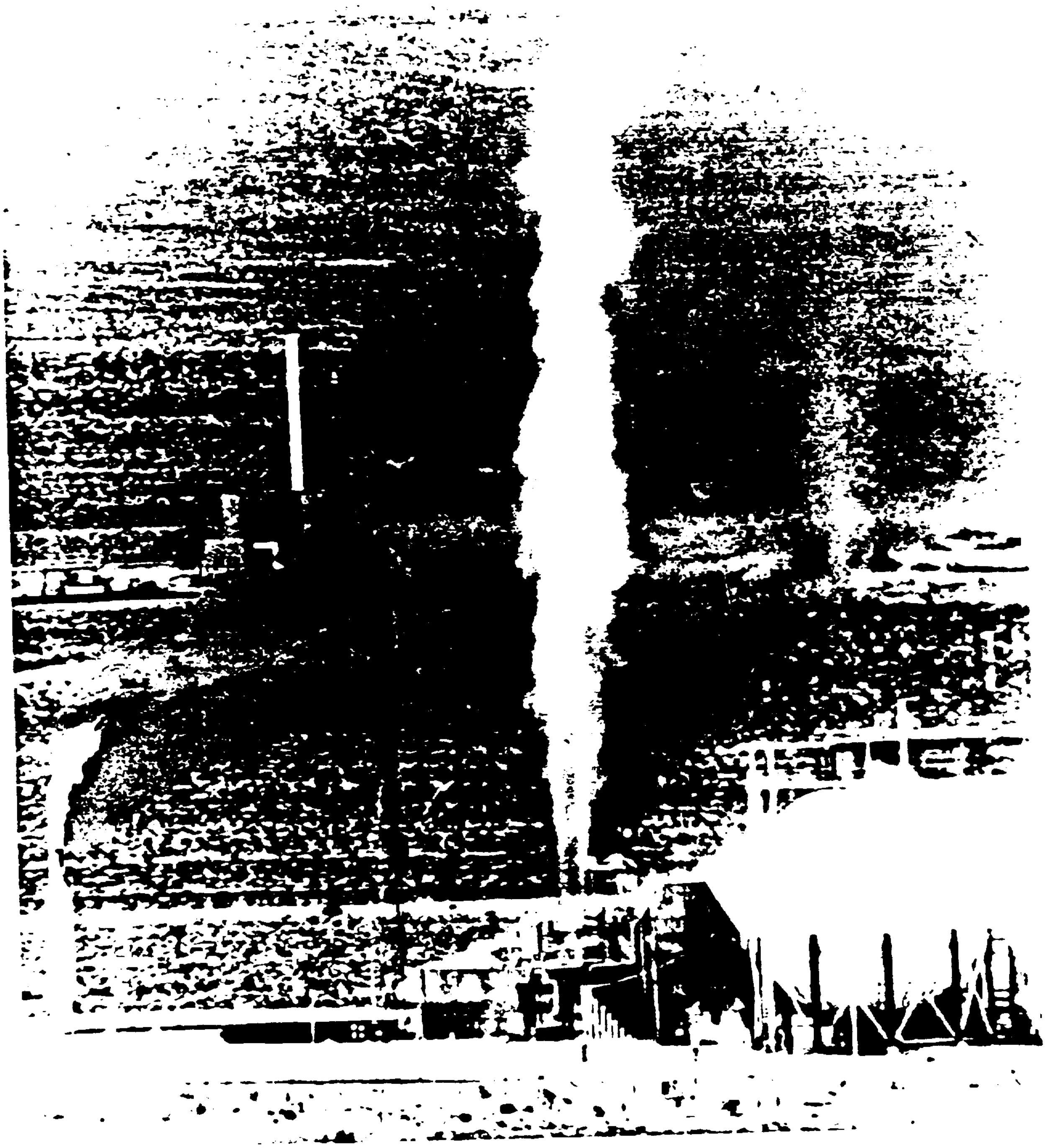


Figure 9

PROTOTYPE NERVA ENGINE - THE NRX/XE -

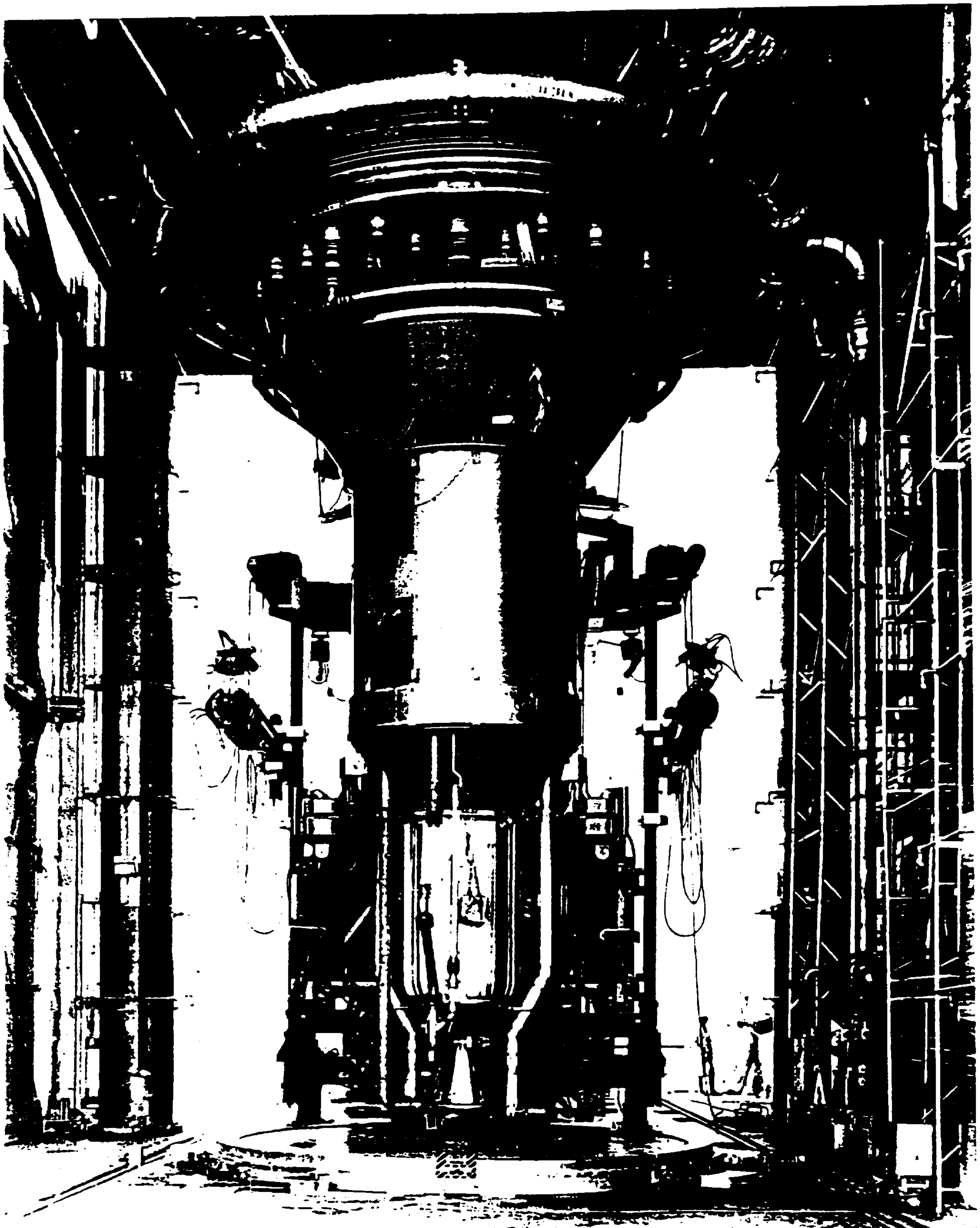


Figure 10

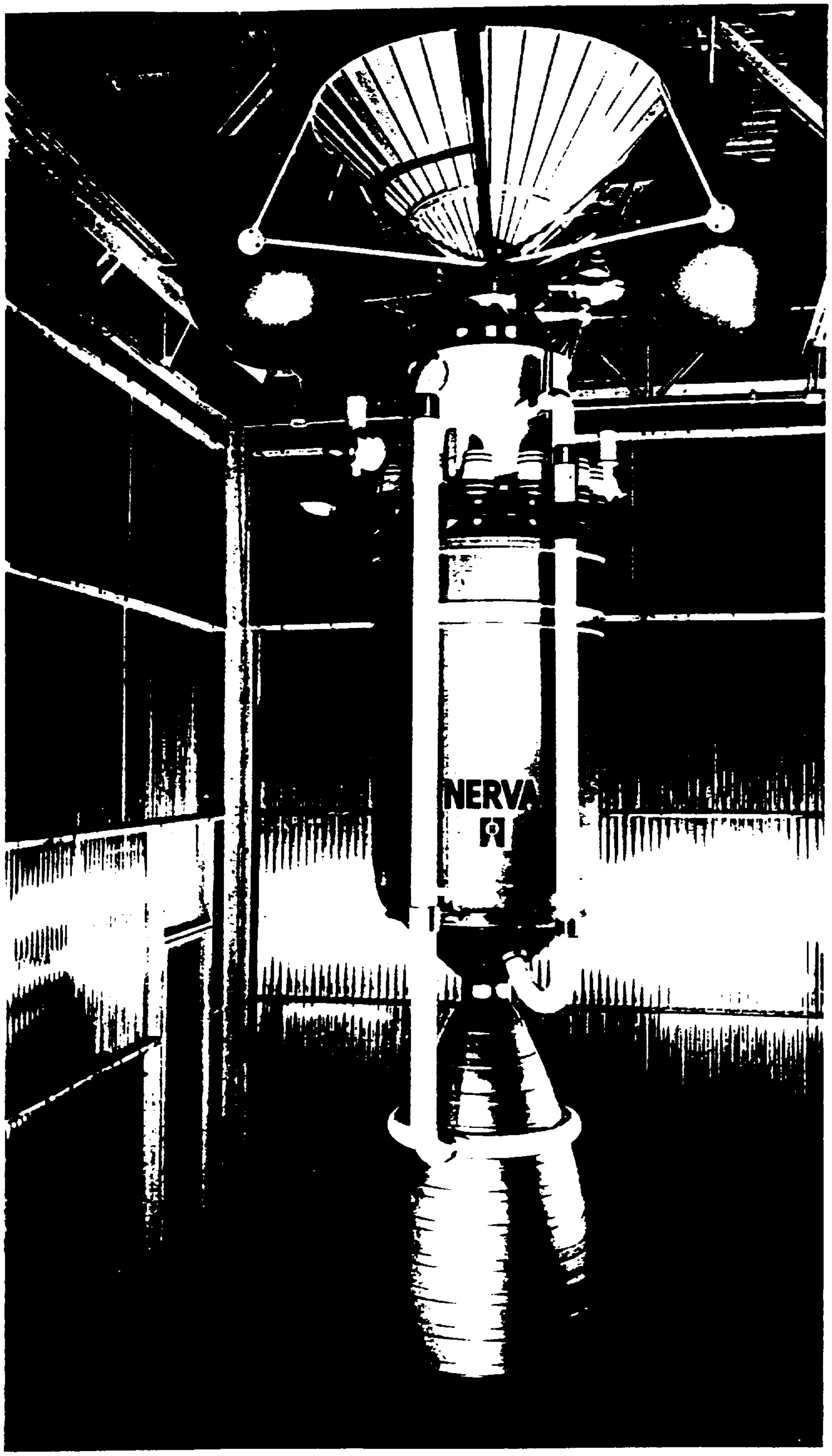


Figure 11

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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 - HARDWARE	5	HOT END SUPPORT REQUIRES 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

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

Table 1

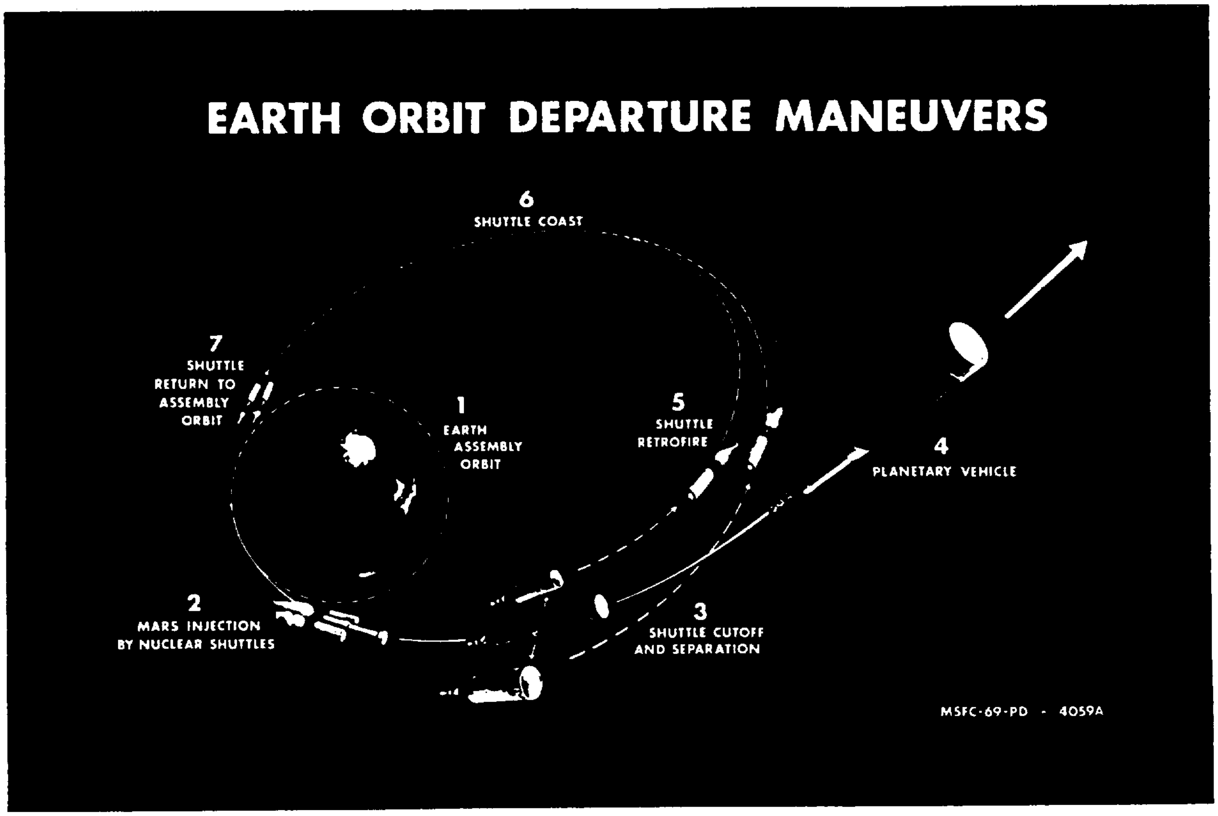

Figure 14

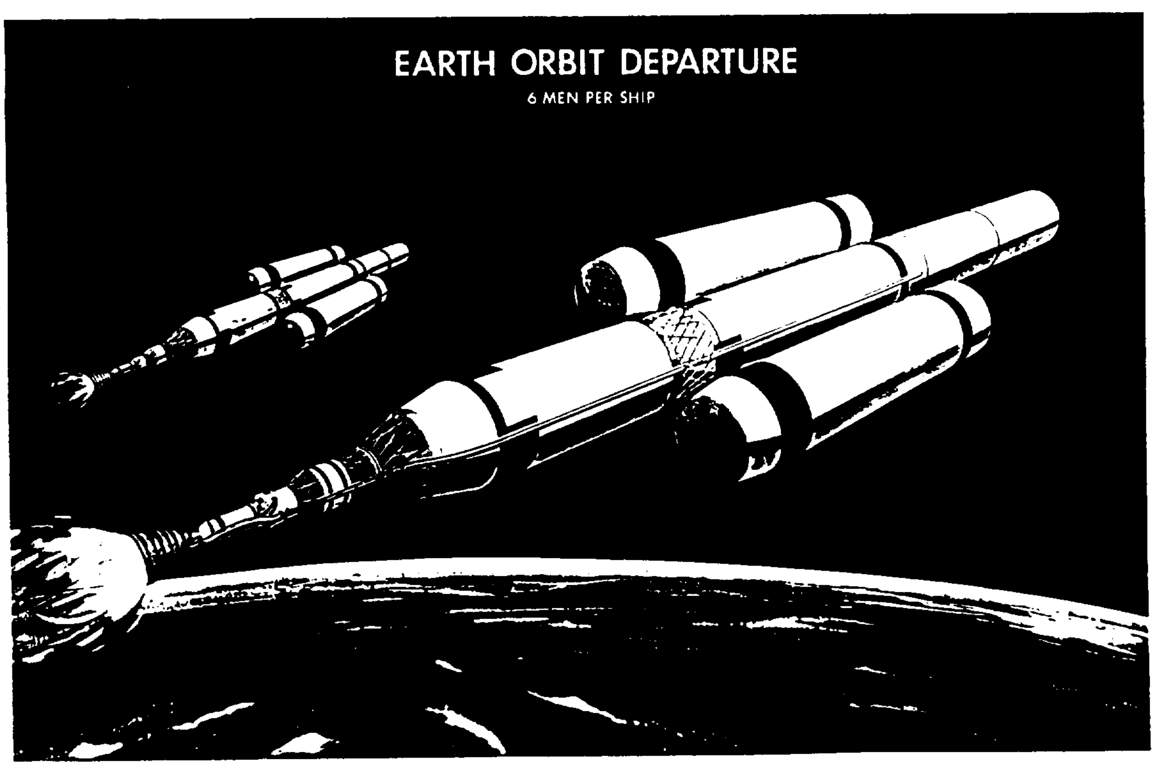

Figure 15

2016 NTR Reference Trajectory

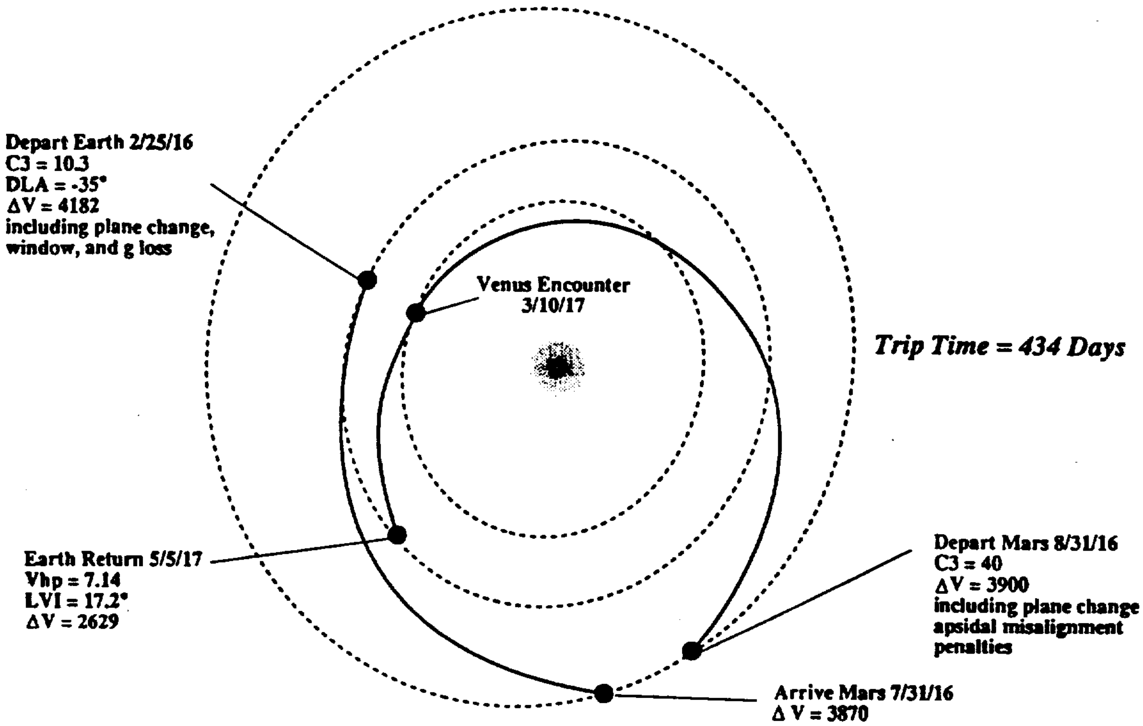

Figure 16

2016 NTR Vehicle Mission Profile

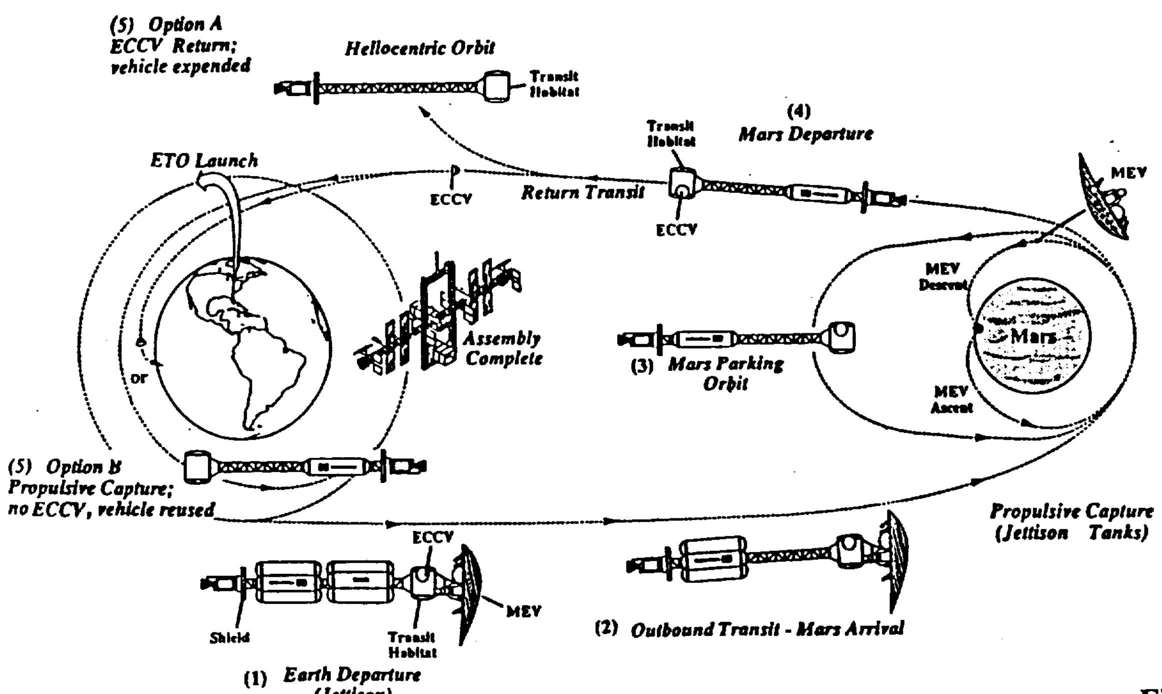

Figure 17

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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

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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)

SPACE EXPLORATION INITIATIVE OFFICE

Table 3

EARTH DEPARTURE G-LOSS

PERIGEE PROPULSION C3 =10 ISP=900

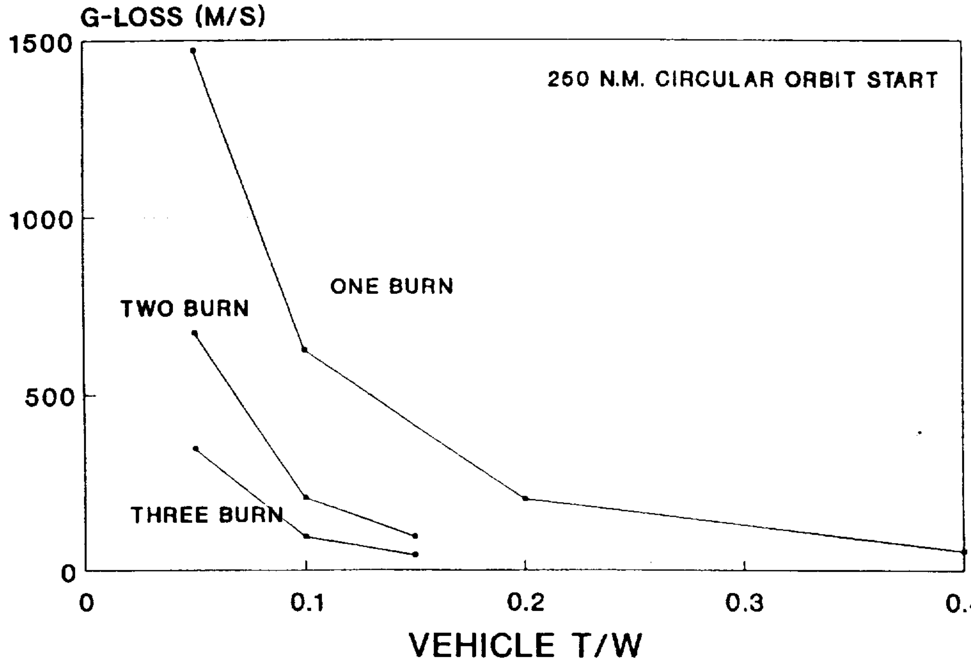

Figure 18

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)

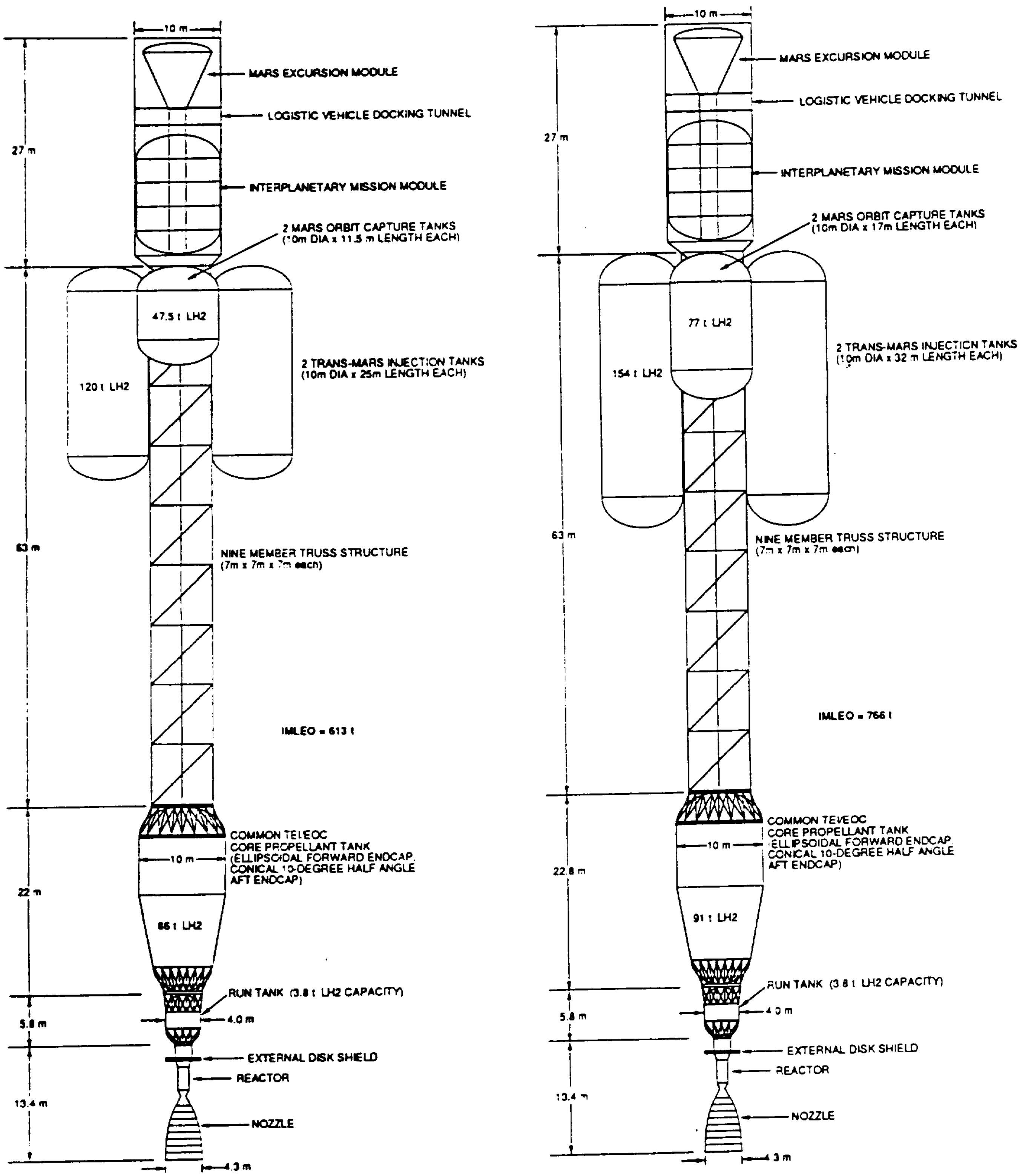

Figure 19

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

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

Table 5

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

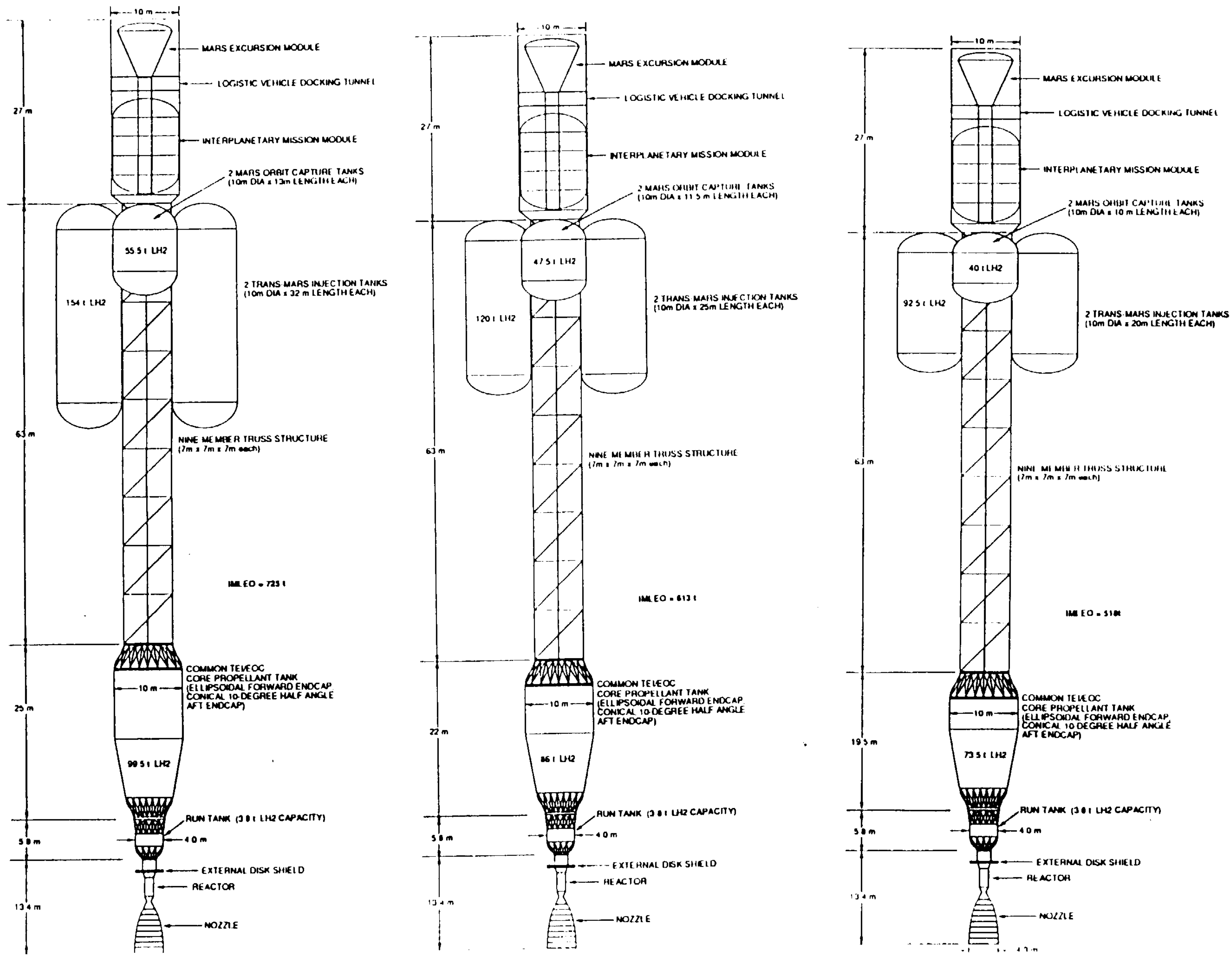

Figure 21

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

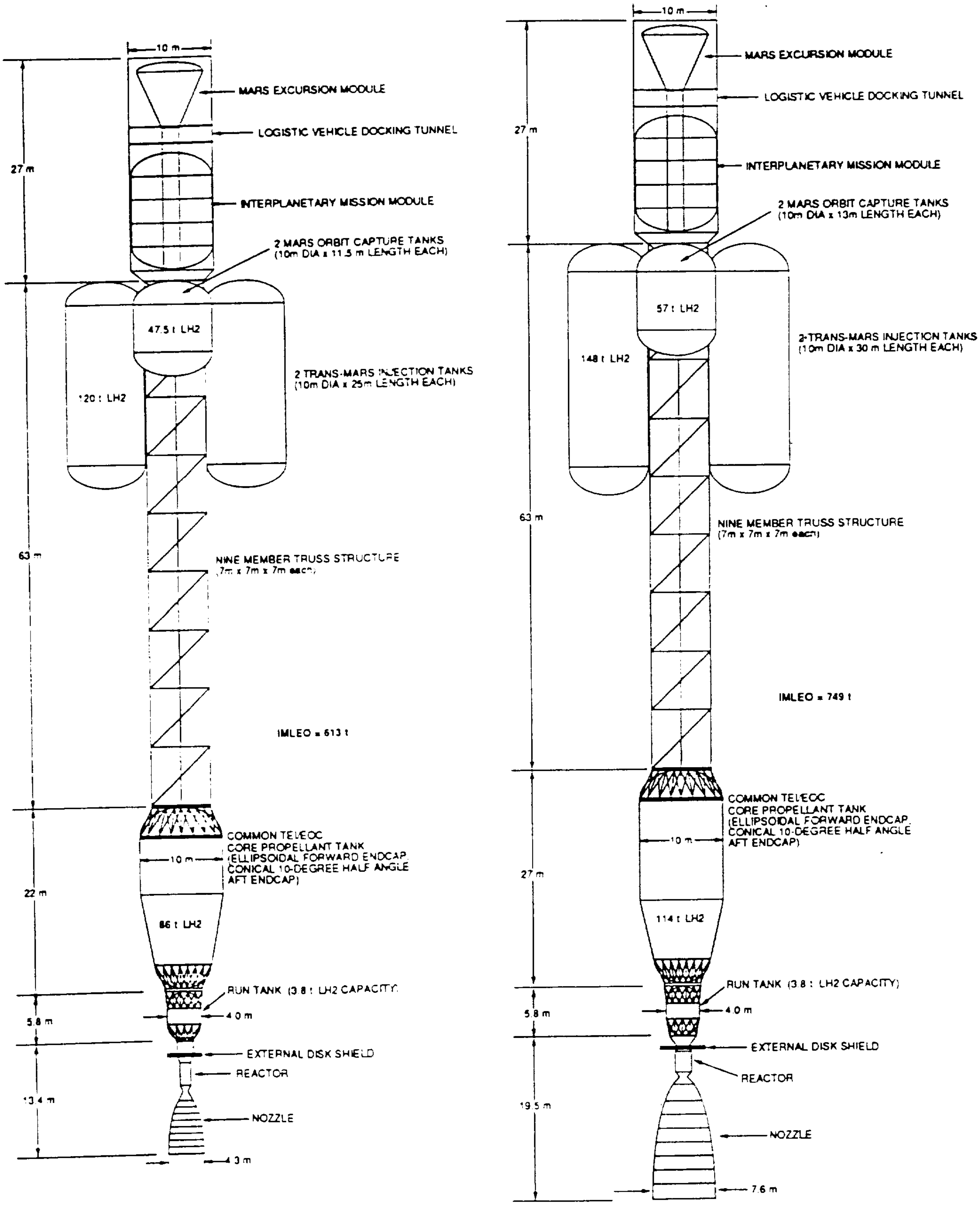

Figure 22

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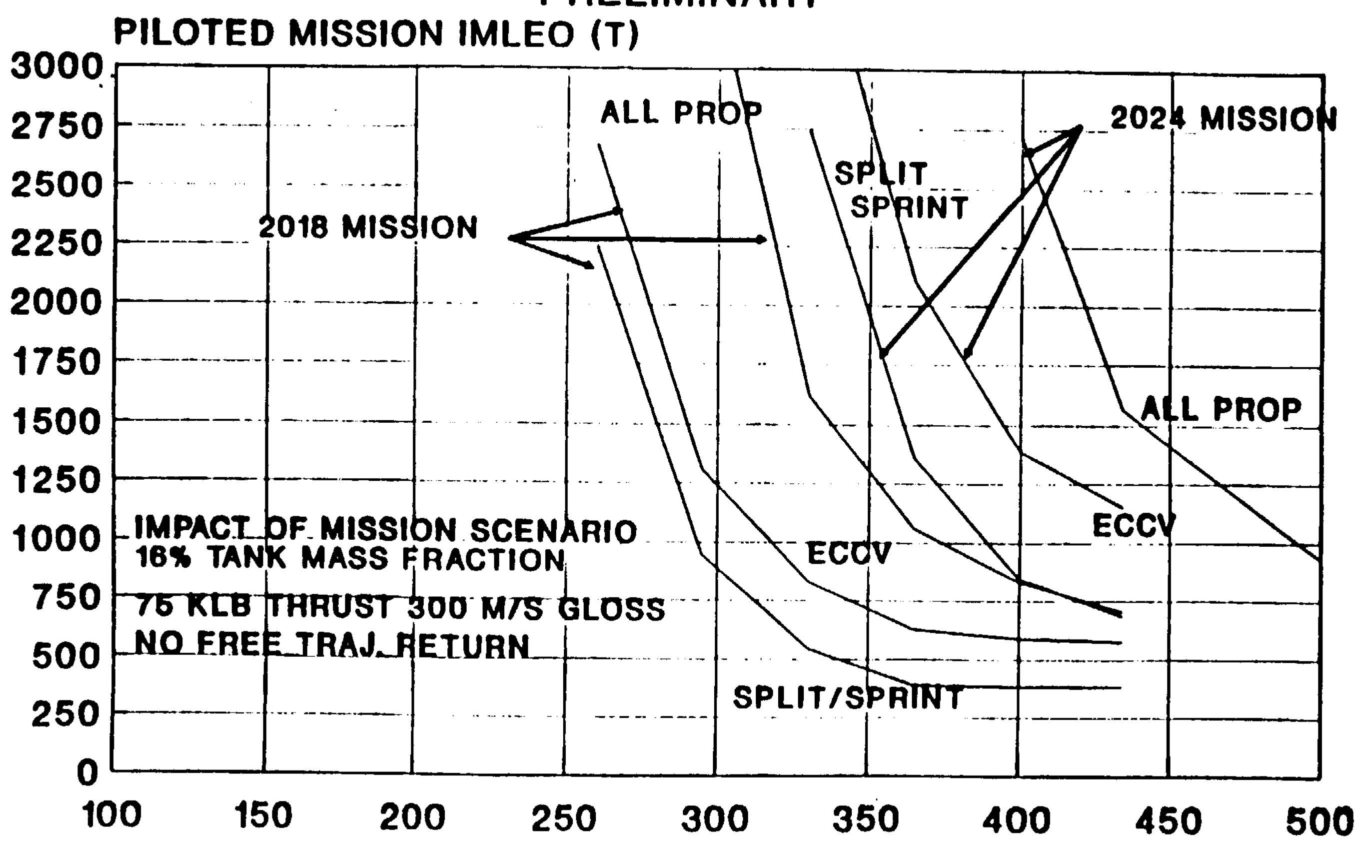
NTP ENGINE PERFORMANCE POTENTIAL
925 SEC ISP -- 4:1 ENGINE T/W
PRELIMINARY



FOR SPLIT/SPRINT PILOTED VEHICLE HAS RETURN PROPS

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Figure 23

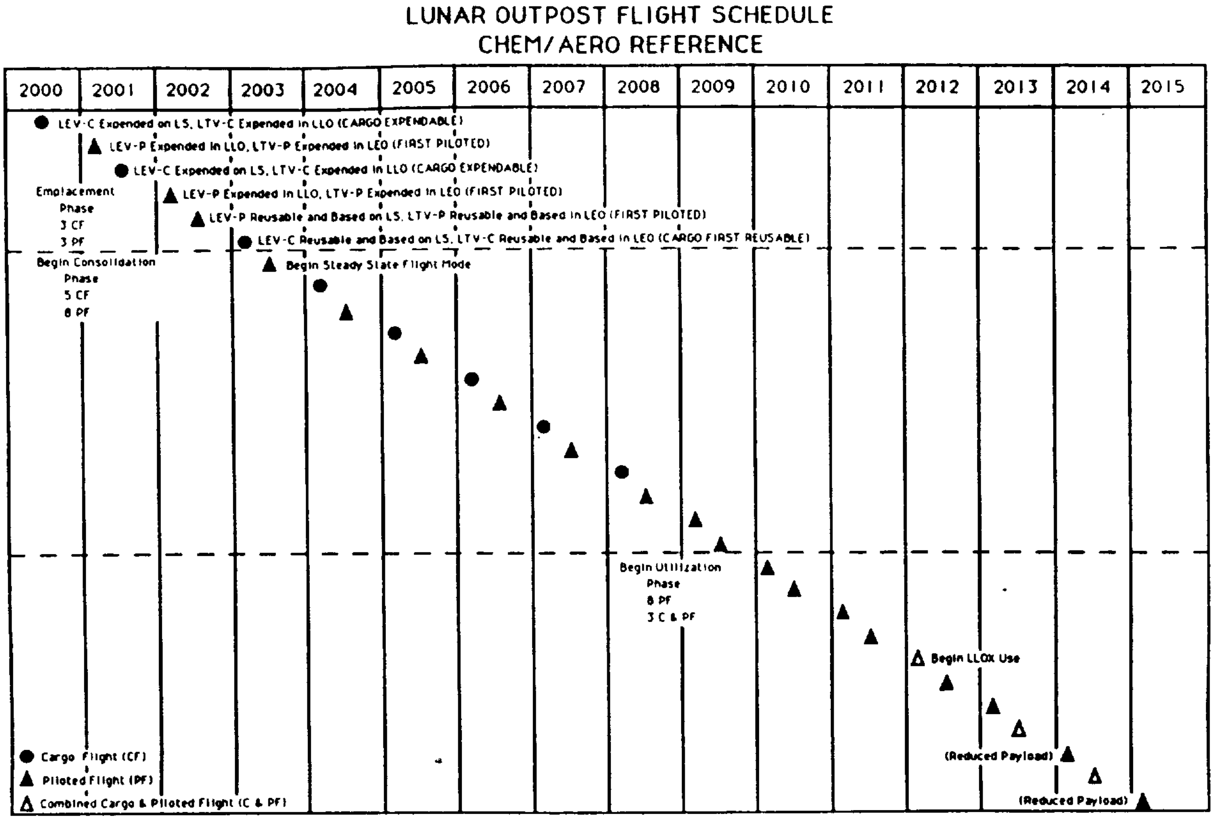

Figure 24

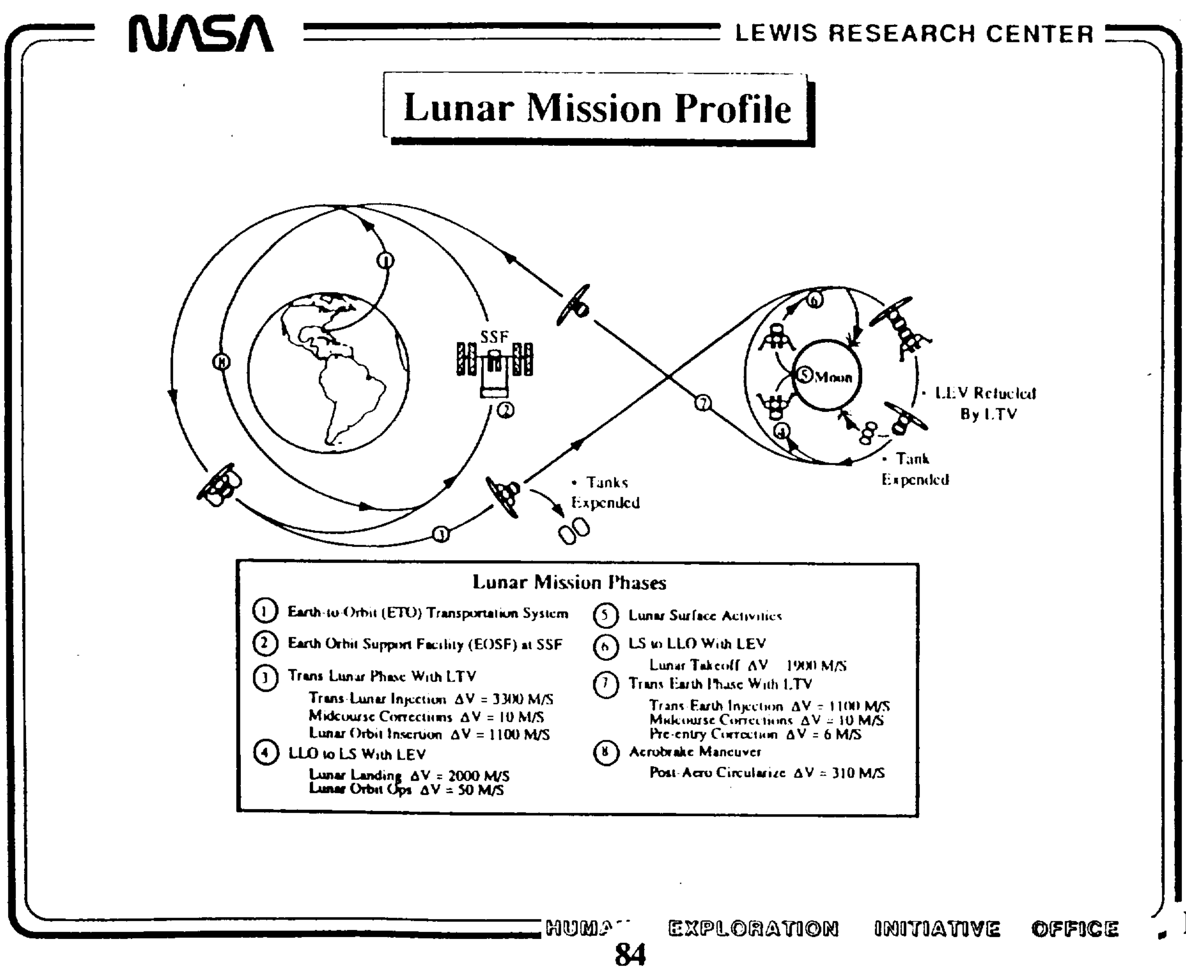

Figure 25

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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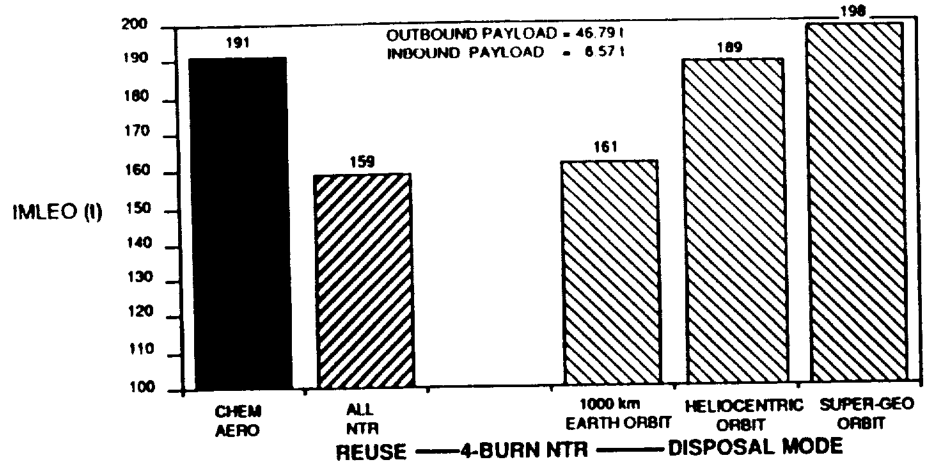
Figure 27

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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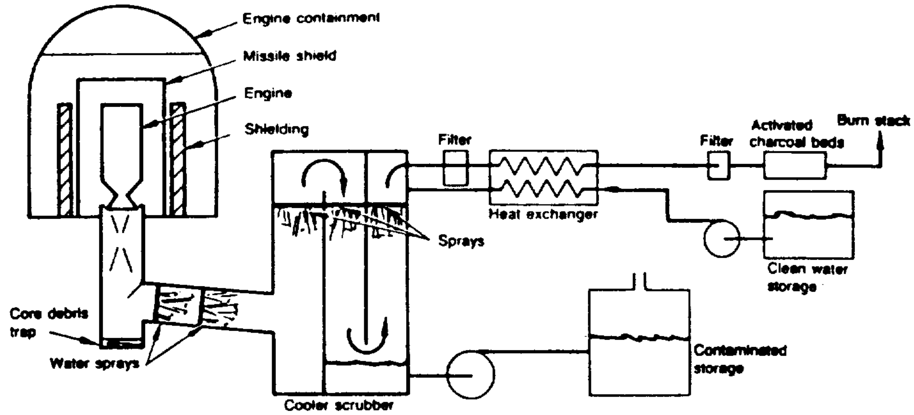
EFFLUENT TREATMENT SYSTEMS

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

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Figure 33

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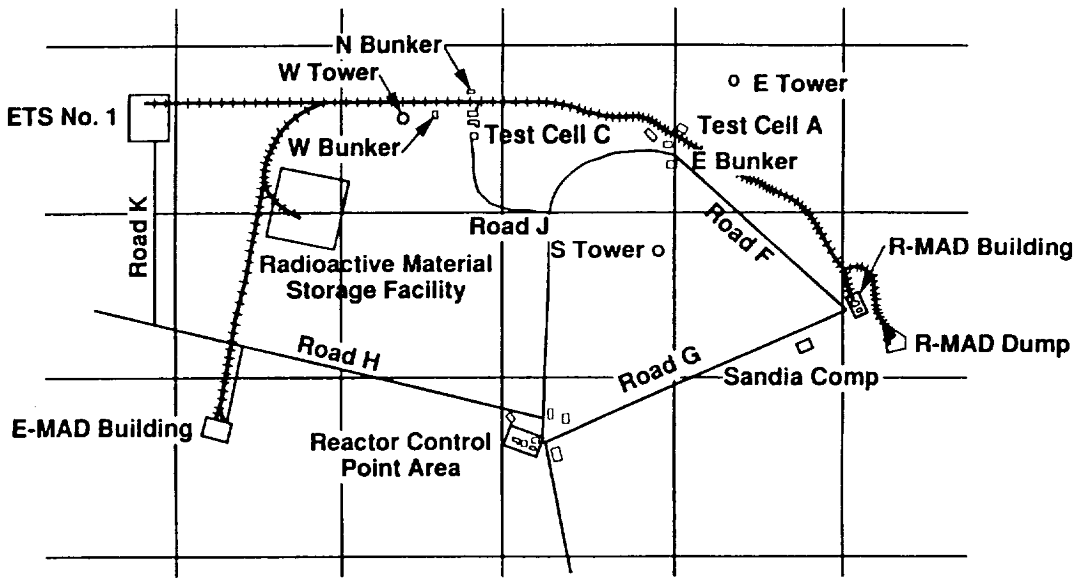
Nuclear Rocket Development Station

Site 400, Nevada Test Site



SPACE EXPLORATION INITIATIVE OFFICE

Figure 34

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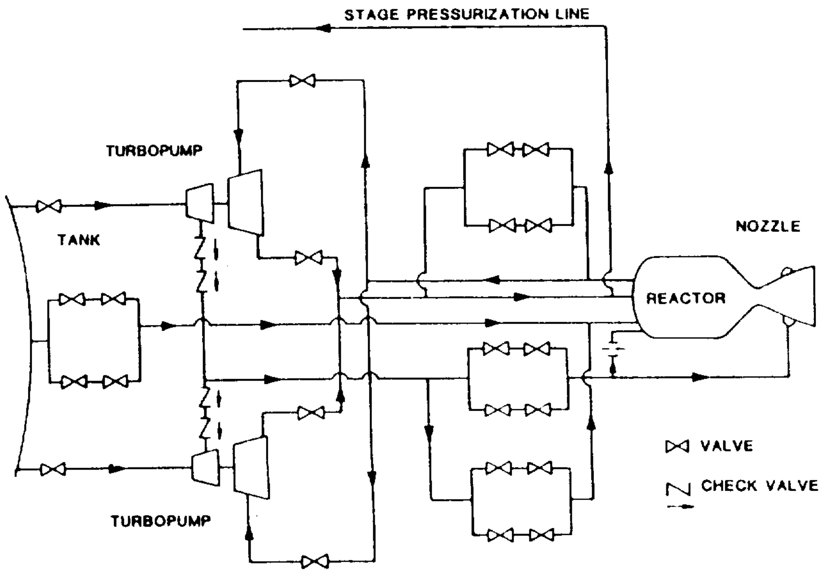
TESTING AT THE NEVADA TEST SITE (NTS)

- 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

SPACE EXPLORATION INITIATIVE OFFICE

Figure 35

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

Figure 36

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.

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}

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*Dose point on axial midplane 100 feet from core centerline

REF. B. SCHNITZLER (INEL)

Figure 38

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 ENABLING FOR MARS MISSIONS AND CAN BE ENHANCING 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

Figure 39