NUCLEAR ROCKETS BASED ON GRAPHITE REACTOR TECHNOLOGY
Their Applications and Development Status

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Presentation for
Second AGARD Lecture Series on
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I. INTRODUCTION

The nuclear rocket offers a step increase in specific impulse over that obtainable from high energy chemical rocket systems. This propulsion advancement improves our ability to accomplish high energy space missions. Accordingly, the nuclear rocket will be the principal propulsion system for manned planetary exploration and will also be useful in extended lunar exploration and unmanned solar-system missions.

The current effort in the United States is directed at establishing the technology of nuclear rocket systems and evaluating their real performance and operating characteristics in advance of the establishment of firm mission specifications. Such an advanced technology development approach will assure that the system can be relied upon when future space missions are identified and established as firm objectives.

The major effort of the nuclear rocket program in the United States is devoted to propulsion systems which use reactors based upon graphite technology. However, a small but still significant part of our resources

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is devoted to experimental and analytical investigations of the feasibility of alternate and more-advanced nuclear rocket propulsion concepts. These alternate concepts include primarily refractory metal reactors and secondarily, molten metal reactors, fluidized beds, and gas core reactors. These various concepts require differing levels of technological development and advancement and offer different potential performance levels. Accordingly, funding levels for each of these concepts differ markedly but their total is substantially below the total effort devoted to graphite systems.

Solid core nuclear rockets, with special emphasis on the graphite reactor systems, are undoubtedly in the most advanced state of development of any of the nuclear propulsion systems. Their performance potential has already been demonstrated by prototype reactor tests. Longer time tests aimed at establishing performance limits are planned during the next several years. Engineering and technology development work is well underway. Solid core nuclear rockets, therefore, will be the first advanced propulsion systems developed for mission use and we can assess their applicability and availability with greater assurance than is the case for any other nuclear propulsion system.

The earliest potential application of nuclear rocket propulsion would be as a third stage in the Saturn V vehicle. When substituted for the chemical-rocket S-IVB stage, a nuclear stage would increase lunar landed payload by 35 to 65 percent. Such payload increments could greatly enhance a program of extended lunar exploration, providing the capability of manned direct landings and a lower number of launches to support a given level of lunar activity. The same vehicle, when used for unmanned missions to the planets and other solar-system destinations, would increase payloads by 45 to 80 percent, depending on the destination. This gain in performance might be valuable in early phases of the planetary program when the objective is to obtain engineering data for the subsequent design of manned spacecraft. In any case, the nuclear stage offers the potential of increasing the performance and, therefore, extending the utility and useful life of Saturn V.

The primary mission for nuclear-rocket propulsion in the long-range space program is manned exploration of the near planets, principally Mars. An adequate job of surface exploration, implying substantial weights for scientific tasks and crew accommodations, and including growth margin for extended exploration, is a very difficult mission. In such circumstances the performance advantages of nuclear rockets are particularly beneficial. Compared to chemical-rockets, the initial weights in Earth orbit of nuclear manned Mars spacecraft are lower by a factor of 2-3 or more. Furthermore, since a typical nuclear spacecraft weight is about two million pounds, the use of nuclear rockets may be essential to keeping the launch and orbital operations within reason. Costs of the planetary program would be greatly reduced and, in case requirements should increase markedly, the nuclear systems would have invaluable growth potential and flexibility.

Graphite reactor and engine technology is being developed in the KIWI/NERVA Projects. These efforts are being directed to ground development tests of a reactor and an engine system in the 50,000 lb. thrust class. The reactor will be capable of being used in a flight system and the engine technology can be the basis for flight engine development.

The nuclear rocket program includes programs of research and technology which go beyond the needs of our current reactor and engine projects to assure that basic data and fundamental understanding stay abreast of performance and lead the way to improved performance. These programs, known as supporting research and technology efforts, assist our present projects in the areas of engine systems analysis, feed system performance and component behavior by mapping performance under a variety of conditions. Fundamental properties of materials, the behavior of liquids and gases under extreme environmental conditions, and the development of new engineering designs and concepts provide an early indication of areas which will require emphasis in future projects as well as providing the base of technology from which these projects will be developed.

The long lead time required to provide test facilities and ground test support equipment and to carry out the work of designing and developing rocket reactors makes it necessary to select a size, power level, and design for the next generation of systems several years in advance of planned test periods. Among the factors which must be considered are the missions to be accomplished in the future space program, development and design problems and uncertainties, the state of technology currently available for these systems, and the sensitivity of reactor and engine performance to mission variations. Clustering nuclear rocket engines affords flexibility in this selection. The present KIWI/NERVA technology allows confidence in reasonable extrapolation to bigger systems.

Work is already underway on the Phoebus reactors, with the first tests in this project to be carried out in the smaller KIWI-size reactors so as to obtain early information on some of the technology to be used in the bigger cores.

This discussion will be devoted to nuclear rocket propulsion based upon solid core graphite reactor technology. The discussion is divided into three principal sections which will cover: Vehicle Applications; NERVA Engine including KIWI reactor work, and Nuclear Rocket Advanced Research and Technology including the Phoebus reactor work.

II. VEHICLE APPLICATIONS

INTRODUCTION

This portion of the lecture will cover the application of nuclear rockets in future space missions. We first assume that there will be advanced space missions beyond Apollo, including extended lunar exploration and manned expeditions to Mars and, possibly, Venus. The consequent desire for advanced propulsion systems requires no lengthy explanation. I will concentrate on an elaboration of mission characteristics, performance comparisons and the manner in which the nuclear rocket fits into the overall space program.

Figure 1* is a coarse roadmap of space missions--past, present and future. The three regions of space flight are arranged in columns: Earth orbit, lunar and planetary. The horizontal divisions show a progression in each region from unmanned missions (at the top) to early manned developmental flights and, finally, operational manned flights. Current authorized programs are marked by asterisks.

I do not intend to go through this chart in detail; the important thing it infers is the flow, in both time and technology, from the comparatively low-energy, low-payload missions we are doing now to more-ambitious missions in the lower righthand half of the matrix. These high-energy, high-payload missions are the primary areas of nuclear rocket application. Chief potential uses are (1) lunar logistics, (2) unmanned planetary missions with heavy payloads, and (3) manned planetary expeditions. The latter continues to be the principal justification for developing nuclear rockets.

Before discussing these application areas in more detail, a few other observations can be made from such a chart format. The integrated nature of the unmanned and manned programs for lunar and planetary exploration should be noted. Unmanned flights come first with both special scientific purposes and a vital role in preparing for manned missions. The availability of engineering data for design of manned spacecraft will certainly depend upon data obtained from unmanned probes even though manned systems will be designed to be as independent as possible of detailed lunar or planetary features. Both types of vehicles will be flown at least during the period of manned developmental flight and probably even after manned flights are operational. There is also a horizontal dependency on the chart: both the long-duration manned lunar missions and the manned planetary missions will lean heavily on human and systems experience in Earth orbit.

Although dates are not shown on the chart, the general timing of the missions is worth some comment. Anything worthy of the name lunar station or base should probably be placed in the latter half of the 1970's. Similarly, in the planetary program, manned flights to Mars or Venus are probably post-1980, and planetary operations would be correspondingly later. Many factors contribute to this estimate of time scale, but the most basic ones relate to

* Figure numbers in this section should have the prefix II, i.e. figure II-1.

systems technology and the information needed on space and planetary environments. Nuclear rocket technology is just one of many which must be established to make these future missions possible or practical. Many other areas will be more limiting than nuclear rockets in determining the pace of the future space program.

MANNED PLANETARY EXPLORATION

Performance Comparison

Many studies have led to the contention that the only reasonable form of propulsion for manned Mars landing missions is the nuclear rocket. Figure 2 is a plot of initial vehicle weight in Earth orbit vs. launch date, showing both nuclear and chemical rockets and variations due to several other factors. I will return to this figure later for some interesting comparisons; at this point I just want to give you a preview of the case that is to be made -- a glimpse of the magnitude of the advantage of nuclear rockets in manned planetary flight.

These curves are for Mars stopover missions of 420 days duration, including 40 days spent at the planet. Rocket braking into Mars orbit is assumed. The payload weights are typical of an early landing mission, and the trajectories are selected to approximately minimize initial weight in Earth orbit. Although launch opportunities are spaced roughly two years apart, the weight curves in figure 2 are drawn as continuous variations for illustrative purposes. They show the trend in weight variation through the seventeen-year cycle from one easiest year to the next.

The lower curves are for nuclear rocket propulsion during Earth-orbit departure and during arrival and departure maneuvers at Mars. The upper family of curves shows the corresponding weight variations for chemical rockets. Both the chemical-to-nuclear weight ratios and the absolute magnitudes are important. The ratios vary from a minimum of about two in the lowest-energy years to over three in the most-difficult years for the reentry velocities assumed; the magnitudes are about 1.5 - 2.5 million pounds for nuclear rockets and 3 - 8 million for chemical. Such ratios of initial weights make the choice of nuclear rockets appear only logical; the magnitudes of the chemical-rocket vehicle weights suggest that the missions may never be done without nuclear propulsion. The weight variation from year to year is also important. The use of nuclear rockets greatly cuts down the difference between the peaks and valleys, suggesting that one basic vehicle could be made to serve all launch opportunities. We will examine the many aspects of this comparison in some detail later, after discussing trajectories and payloads.

Trajectories

There are two primary types of Mars roundtrip trajectories with which we should be familiar in order to assess the role of nuclear rockets in providing Mars-mission capability:

(1) Opposition-class (or short) trips, generally of 400-500 days duration and short (10-40 days) stopover time.

(2) Conjunction-class (or long) trips, with total mission times of 750-1000 days including staytimes of up to 300-450 days.

In addition, a variation worth special mention is the Mars roundtrip with Venus swingby, which makes use of a gravity turn at Venus on either the outbound or homebound leg of an Earth-Mars trajectory.

Figure 3 shows the geometry of a typical opposition-class trajectory. The inner circle is the orbit of the Earth about the sun; the outer ellipse is the eccentric orbit of Mars, which is also inclined to the plane of Earth's orbit nearly two degrees forming this dashed line of nodes (Ω). A spacecraft on a 500-day roundtrip would start from Earth at point 1 and follow the heavy dotted path to its rendezvous with Mars at point 2. This outbound leg is of 9-months' duration, as seen from the position of Earth at 2. (The name "opposition-class" is due to the fact that Mars is near opposition -- that is, directly opposite from the sun to an observer on Earth -- during the capture period.) After a stay in Mars orbit of a month or so, the spacecraft follows the dash-dot path back to Earth. The homeward leg is of about 6.5 months' duration, arrival at Earth being at point 4.

Note that the outbound path stays almost completely between the orbits of Earth and Mars, but the return flight must cut in close to the sun in order for the spacecraft to catch up with the Earth. The minimum heliocentric radius may be less than half an astronomical unit ($<$0.5 A.U.). Such a roundtrip trajectory is inherently a high-energy flight path. In order to keep the total mission time down to a year and a half or less, the velocity increments of the various propulsion periods must be relatively high. This figure gives a qualitative indication of this when we note that a large angle between the flight paths and the planetary orbits corresponds to a high velocity increment (ΔV). The encounters at Mars are both shown to be high-angle situations; Earth return would be another, although much of this phase may be handled aerodynamically.

Compare this geometry to that of a long or conjunction-class trip shown in figure 4. In this flight mode all trajectory sections stay outside the Earth's heliocentric orbit. Note also how small the encounter angles are. This is obviously a low-energy class of trips.

A conjunction-class roundtrip is so named because planetary conjunction occurs during the stay at Mars. That is, Mars is on the far side of the sun from the Earth and, to an observer on Earth, Mars and the sun appear to be in line. The outbound leg (from points 1 to 2) is about 10 months long. The return leg (3 to 4) is about 11 months in duration. In between, from points 2 to 3 on the Mars orbit, the spacecraft spends over an Earth year at the destination planet. Such a long stopover is required because the spacecraft must wait for the next chance to make a low-energy journey back to Earth. The lengths of both the staytime and the total mission time are the key characteristics of conjunction-class trips. They raise major technology problems and uncertainties associated with human factors and system reliability, but they also present the opportunity for extensive exploration which may be desirable later in a planetary exploration program.

The Venus-swingby variation of an opposition-class trajectory is illustrated in figure 5. In this example the spacecraft travels from Earth to Mars via Venus. There are an equal number of cases in which the Venus swingby occurs on the return leg. In figure 5 the outbound trajectory includes a 160-day Earth-Venus segment (1 to 2) which goes inside the orbit of Venus before rendezvousing with that planet. At the Venus encounter the spacecraft makes a close enough approach for the planet's gravitational field to act upon its trajectory. The perturbation, which in this case is an acceleration, permits the spacecraft to catch up with Mars, thereby making the return flight easier. The Venus-Mars trip time is about 130 days, and the staytime at Mars (3 to 4) is short (15 days in this example), as is characteristic of opposition-class roundtrips. The homebound leg (4 to 5) is of about 230 days duration, and the velocities relative to the planets are lower than they would have been without the Venus swingby. The Earth-atmosphere approach speed, for example, is reduced from 66,000 to 42,000 feet per second. Although the total mission time has been extended by a few months, the energy requirements in 1980, a high-energy year, have been reduced nearly to those of the lowest energy years.

Unfortunately, the requirement that three planets be in correct relative position may impose some operational restrictions on the use of Venus swingby trajectories. Although many attractive launch opportunities of this type exist -- at least two thirds as many as for opposition-class trips -- the launch windows within each opportunity are sometimes narrow. That is, in some years the penalty for deviating from the optimum launch date may be so severe that it is impractical to provide a reasonable spread of launch dates. Consequently, it is not yet clear that the Venus swingby mode will be widely used, provided nuclear rockets are available to make opposition-class trips possible. However, if the launch window restrictions are not prohibitive, this mode may be useful for special reasons. For example, it might offer the only opportunity to be at Mars during a particularly interesting season. More analysis is needed before the usefulness of this mode will be completely understood.

Energy Requirements

Figure 6 summarizes much of the foregoing qualitative comparison of energy requirements. The sum of velocity increments is plotted against Earth launch date. For this illustration the total propulsive velocity increment is the sum of those at Earth departure, Mars arrival and Mars departure plus the retro at Earth return required to reduce the atmosphere-entry speed to 50,000 feet per second. Such a minimization of total velocity increment does not always give the lowest values of initial weight, because of the way the individual ΔV's are distributed, but the trends are much the same.

The sharp curves show the ΔV variations in each launch opportunity; the envelope lines connect the minimum points as though the variations were continuous with time. The trend of the loci of ΔV minima for 400-500 day roundtrips illustrates the effect of the eccentricity of Mars' orbit. During the late 1970's, the spacecraft makes its rendezvous with Mars near that planet's aphelion (furthest distance from the sun). The spacecraft must not only travel further out in the sun's gravitational field but must do so in roughly the same time as in easy years because it must catch up with the Earth on the return leg. Thus the total ΔV is shown to be about 60,000 ft/sec for a 1979 launch but under 40,000 ft/sec in 1986.

The two branches of the opposition-class trajectory family need not concern us. Although the total ΔV's differ during the 1970's, the resultant initial weights are very nearly the same. Furthermore, this time period is probably too early for manned planetary flight. The next such period is in the late 80's and early 90's. During the 1979-86 period the two branches are so nearly alike in total ΔV and ΔV distribution that the shorter trips seem the obvious choice.

The lower curve for 950-1050 day (conjunction-class) roundtrips shows very little variation with launch date. The sum of the velocity increments remains near 25,000 ft/sec throughout the cycle. The eccentricity of Mars' orbit has little effect because both legs are relatively-slow, low-energy trajectories and a wide flexibility exists in selecting staytime.

The individual velocity increments are also of considerable importance. As previously mentioned, this plot includes only the ΔV at Earth return down to an atmosphere-entry speed of 50,000 ft/sec. If an essentially unlimited entry speed could be handled aerodynamically, the total velocity increment in the unfavorable years would be reduced by 15,000-20,000 ft/sec. As a result, the sum would be but little higher than in the easiest year. The latter value would not change (in 1986 for example) with a higher entry velocity capability because approach speed is less than 50,000 ft/sec in a favorable launch year. On the other hand, if 50,000 ft/sec were too high and retro ΔV's became excessive, the entire trajectory selection would change to even out the individual velocity increments.

In a similar fashion, the ability to use aerocapture at Mars would alter the selection of trajectories and further reduce the variation in total ΔV from year to year. Such energy-reduction techniques, which are important possibilities in such a mission, will be discussed again after we have considered the subject of payload requirements.

Of course, this lengthy examination of trajectories and energy requirements is leading back to the initial-weight comparison which you have already seen in preview. Graphs like figure 6 point up some of the difficulties of the missions; the discussion of payload requirements will bring in others. The total picture will add up to a strong mandate for nuclear rocket propulsion.

Payload Requirements

The third major factor in the determination of overall performance, in addition to energy requirements and propulsion system performance, is the payload weight. It is customary to think of a manned Mars mission in terms of a specified set of payload weights being transported through particular energy changes in space. We have described the magnitudes of the energy changes assuming propulsive accomplishment of all but the Earth reentry requirements. The principal characteristic of the propulsion systems under consideration is their specific impulse -- about 800 seconds for nuclear rockets as opposed to a maximum of about 450 seconds for chemical rockets. Let us now describe the payload components and their weights.

Four elements of the payload for a Mars landing mission are important:

(1) Mission module
(2) Radiation shelter
(3) Earth reentry module
(4) Exploration payload

The mission module consists of the living and working quarters for the crew. It may include the control center, which is always manned by the crewmen on duty, although the control center could be the radiation shelter. The latter is the shielded compartment in which all crew members are sheltered during a dangerous solar flare. The Earth reentry module (ERM) is a spacecraft in which the crew enters the atmosphere for deceleration and controlled descent to the landing site. These three payload elements compose the return payload. That is, they are transported to Mars and back to the vicinity of Earth. In those cases wherein retro thrust is needed at Earth return, the weight of the retro propulsion system and propellant would be considered part of the ERM weight. The exploration payload is simply the sum of all items carried to Mars and left there, including Mars Excursion Modules, unmanned probes and other data-gathering equipment. A Mars Excursion Module (MEM) would carry the landing party to the surface and return them to the orbiting spacecraft.

Mission Module -- The main parts of the mission module are the crew compartment structure, life support system and power supply. These items are functions of crew size and trip duration, which are related parameters. Crew size depends upon considerations of task assignments, duty cycles, skill specialization, and possible incapacitation during the mission. Very little quantitative information is available in these areas. Speculation on crew duties has been reported in several mission studies, and estimates of the variation of crew reliability with trip duration have been attempted. The results appear to call for a minimum of 6-8 men for Mars landing missions of 400-500 days and 12-16 men for 800-1000 days. This increase in crew size for conjunction-class roundtrips counter balances much of the low-energy advantage of the long-trip mission mode. Power supply weight is also a function of crew size and trip duration because life-support power requirements tend to be 1-2 kwe per man and power supply weights will rise to provide the redundancy for several years' operation. However, if a nuclear reactor power supply is used, the largest part of the weight may be the radiation shield. Shield weights could be reduced if the vehicle configuration would permit a large reactor-crew separation distance.

Typical mission module weights are generally estimated at the 50,000-100,000 pound range for 6-8 man crew and a 400-500 day trip. It is usually assumed that technology will permit a partially-closed life support system (water loop closed, at least), and many studies include a SNAP-8 reactor power supply. The need for artificial gravity cannot yet be specified, but provision for rotation would certainly add to the mission module weight.

Radiation Shelter -- The important parameters in determining radiation shelter weights are integrated dose and crew size. Of course, this presupposes the knowledge of (1) solar flare fluxes throughout the pertinent regions of the solar system and (2) crew tolerance, including the effect of body recovery rate. With the environment and dose criteria known, the shelter can be designed to provide protection for a specified number of people from a particular flux-time input.

Solar-flare protons are the most important radiation in determining shield thickness; rocket-reactor radiations give a lesser dose increment, and galactic radiation is a background contributor which is a problem only when the shield is thick enough to produce large amounts of secondary radiations. Van Allen belt radiation is negligible in high-acceleration flight modes. The integrated dose from solar flares depends upon solar cycle (launch date), number of major flares (trip duration) and trajectory type (minimum heliocentric radius). For lack of better or more detailed information, the proton flux is assumed to vary inversely as the square of the heliocentric radius.

We have previously seen the relationship between trip duration and approach distance to the sun. Short trips generally involve approaches to within the radius of Venus' orbit (0.7 A.U.); long trips stay outside of Earth's orbit (1 A.U.). Time appears to be the more important parameter, resulting in greater shield weight per man for conjunction-class trips than for opposition-class trips. In addition, the larger crews needed for long missions increase the difference in radiation shelter weights.

Shelter weights of about 15,000 pounds seem reasonable for 6-8 men and 400-500 day trips. The complicating factor, however, is that the shielding material is not necessarily all extra weight. Much of the peripheral equipment and supplies for the mission module can be arranged so as to provide radiation protection. This is particularly true for directional radiation from the rocket reactors, if such radiation is at all significant. Another interesting possibility is the use of chemical rocket propellants for solar flare shielding -- actually pumping a liquid from the shield into propellant tanks in the ERM at the end of the mission. Depending upon how you look at it, this is a way to get free shielding or free retro propulsion, although it is not all free. The chemical rocket propellants may not be the best shield material, and the pumps and liquid-handling system will be a weight penalty. In fact, the introduction of another pumping system with its additional reliability concern may make this technique undesirable for early missions.

Earth Reentry Module -- The weight of the Earth reentry module is a function of atmosphere-entry speed and crew size. Let us assume that a basic ERM shape has been selected. That is, a lift/drag ratio is provided so that landing maneuverability is satisfactory. Assume also that a minimum corridor depth is specified, consistent with projected guidance capability, and a maximum g-load for the crew (about 10 g's) is imposed. A lower limit may also be placed on reaction time for performing aerodynamic maneuvers. From such a set of ground rules will come a variation of ERM weight (for 6-8 men, in this case) with entry speed. This variation will be due to changes in heat-shield and structure weights. At some high-speed point the use of all aerodynamic braking will have reached its limit; for higher approach velocities either a retro rocket must be added or a higher L/D shape assumed, whichever is lighter.

This variation of ERM weight with entry speed influences the selection of the trajectory. Optimum entry speeds for opposition-class trips are in the range of 50,000-75,000 ft/sec without Venus swingby and 40,000-55,000 ft/sec with Venus swingby. The highest velocities occur in the unfavorable years, when the spacecraft meets Mars near aphelion of the planet's orbit. For 800-1000 day trips the optimum entry speeds are in the range of 40,000-45,000 ft/sec and would, therefore, probably be compatible with direct atmospheric entry capabilities.

A basic ERM weight (without retro propellant) for 6-8 man crew and a 50,000 ft/sec entry capability will be something like 15,000 pounds. For the launch opportunities in the 1980's, the approach velocities will be moderate (<60,000ft/sec) and technology may be available to handle an entry speed of at least 50,000 ft/sec. Thus some retro thrust may be needed but not a prohibitive amount. Nevertheless, the retro propellant may double the weight of the basic ERM.

Exploration Payload -- The principal component of the exploration payload will be the Mars Excursion Module (MEM). The landing craft will carry several crewmen to the surface and return them to the orbiting spacecraft. The other weights of data-gathering equipment are lesser in magnitude but relatively undefined. Due to the meager amount of Mars atmosphere data, any landing craft or probe weights are subject to large changes. If the Mars atmosphere density is as low as current speculation suggests, landing systems may require some propulsion in addition to their aerodynamic devices (i.e. parachutes plus touchdown rockets). Furthermore, until the nature of the surface operations and other scientific activities is known, the exploration payload cannot be well defined.

Several studies have estimated MEM weights at 50,000-80,000 pounds. Such a lander would carry 2-3 men to the surface for an exploration time of a week or two. Approximately a ton of exploration gear would be taken to the surface; only a small weight of samples would be returned. This is obviously a small, possibly even a minimal mission. Ascent propulsion would be by storable chemical rockets. If the mission mode were of the conjunction-class, so that about a year was to be spent at Mars, there would have to be an entirely different approach to surface operations. In fact, depending upon the environmental situation, there might be a strong desire to land the entire spacecraft instead of using Mars orbit rendezvous. Certainly, the amount of exploration equipment to be used in a years' staytime should be much greater than for a 10-40 day stay. The real problems of the very long staytimes (a year) appear to be formidable in terms of man and machine; they are a very long way off if they will ever be done. Thus, there are several reasons why a satisfying analysis of the 800-1000 day missions has not been conducted for comparison with those of 400-500 day missions.

Relative Importance of Payloads -- To put the foregoing discussion of payloads in perspective, figure 7 has been drawn. Its purpose is simply to illustrate the relationship between particular payload weights and initial weight in orbit. In the center of the display are two blocks, labeled return payload and exploration payload. The weights assigned are 130,000 pounds for the sum of mission module, radiation shelter, Earth reentry module and retro propellant weights and 100,000 pounds for the payload left at Mars.

Each pound of return payload will have more of an effect on initial weight in Earth orbit than will a pound of exploration payload because the former has been propelled through an additional energy change (at Mars departure). This is illustrated by means of the diverging areas extending from the payload blocks to the initial weight bars. Note that the build-up of nuclear-rocket Earth-orbit weight is at the left and that for chemical rocket propulsion is at the right. A 100,000 pound payload transported to Mars by nuclear rocket propulsion requires 400,000 pounds in Earth orbit; in other words, the leverage factor is 4. A 130,000 pound payload transported onto an Earth-return trajectory, all by nuclear propulsion, contributes an additional 1,700,000 to the initial weight because the leverage factor is 13. Of course, these leverage factors are functions of the propulsion system characteristics, namely specific impulse and stage mass fraction, and the velocity increments. Figure 7 is for a 1979 opposition-class trajectory.

The corresponding leverage factors for chemical rockets, shown on the right of the figure, are larger because of the less efficient propulsion system. The 100,000 pound exploration payload accounts for 800,000 pounds of initial weight in Earth orbit; the 130,000 pound return payload corresponds to 5,700,000 pounds of initial weight. Not only are these initial weights larger than in the nuclear rocket case, but the chemical rocket is shown to be relatively more sensitive to return payload than is the nuclear rocket. This is shown by the ratios of leverage factors: $44/8 = 4.5$ in the chemical case; $13/4 = 3.25$ in the nuclear case.

Figure 7 should illustrate two points: (1) that the magnitude of the return payload is much more important than the magnitude of the exploration payload and (2) that chemical rockets are more sensitive to changes in payload weights than are nuclear rockets.

Initial-Weight Requirements

Now, let us return to a consideration of figure 2, the plot of initial weights in Earth orbit for Mars landing missions over a long span of launch years. Keeping in mind that these data are for a single mode, opposition-class trips with aerobraking only at Earth return, the following are illustrated:

(1) The comparison of nuclear and chemical rockets
(2) The effect of a variation in allowable Earth-atmosphere-entry speed
(3) The effect of a variation in nuclear-rocket specific impulse

Recall that the basic payloads have been fixed, although variations with entry speed and integrated solar-proton dose are included, and that each trajectory has been optimized to give minimum initial weight for a particular set of energy requirements, payload weights and performance parameters. Alternate mission modes will be discussed later.

Nuclear-Chemical Comparison -- The propulsion-system type, as mentioned previously, affects both the absolute and the relative magnitudes of initial vehicle weight over the span of years. An obvious indication from figure 2 is that the increased specific impulse of nuclear rockets greatly reduces the variation between favorable and unfavorable launch years. This is explained by referring to the mass-ratio equation, which is an exponential relationship:

$$\frac{W_O}{W_E} = \exp\left(\frac{\Delta V}{gI_{sp}}\right)$$

where W_O is initial weight, W_E is empty weight, ΔV is in ft/sec when g is in ft/sec^2 and I_{sp} is in sec. In the unfavorable launch years of the late 1970's, the values of ΔV are so high that the exponential gives very large mass ratios. With a chemical-rocket value of I_{sp}, the propulsion stages are near the extreme end of their capability. Consequently, substitution of a higher value of specific impulse -- nearly twice as large in the nuclear-rocket case -- has a powerful effect. Whereas a single chemical-rocket stage would be a marginal situation, a nuclear stage would be efficient and adequate under the same circumstances. To be specific, the nuclear-rocket initial weight for 1979 is 50 to 80 percent greater than in 1988, whereas the chemical-rocket weights differ by 110 to 180 percent, depending on the entry speed. The smaller the difference from one launch opportunity to another the more reasonable is the expectation that a basic propulsion capability can be set up which will serve the Mars exploration program for many launch years.

Figure 2 also shows that nuclear rocket propulsion provides a large performance advantage over chemical rockets. For example, in 1979, when nuclear propulsion gives an initial weight of about 2.5 million pounds, a chemical-rocket vehicle would weigh 6-8 million. In a low energy year the comparison is between 1.5 and 3 million pounds in orbit. Such a large difference in gross weight leads immediately to the conclusion that the cost savings from only a few Mars missions, and possibly only one, would pay for the entire nuclear rocket development program. Furthermore, initial weights of 6-8 million pounds are generally considered impractical, even with rendezvous capability and post-Saturn launch vehicles capable of putting about a million pounds into Earth orbit. If chemical rocket propulsion were to be used for Mars landing missions in a high-energy year, different mission modes would be required, perhaps a conjunction-class trip (very long trips and Mars staytimes) or one using Mars aerocapture and a Venus swingby. However, even the adoption of such energy-reduction techniques would still give the nuclear rocket an advantage of about a factor of two.

Effect of Entry Speed -- Figure 2 shows two sets of curves for each propulsion-system type, one for full aerodynamic braking at Earth return and the other for chemical retro-thrust down to an atmosphere-entry speed of 50,000 feet per second. The effect on initial weight of this difference in entry speed varies from 0-20% for nuclear vehicles and 0-35% for chemical-rocket craft. There is no effect in favorable years because the minimum-weight trajectories do not involve entry velocities greater than 50,000 ft/sec. The effect is large in high-energy years because optimum trajectories, which account for variations in ERM and retro-propellant weights, call for Earth-approach speeds to over 70,000 ft/sec.

Even though the ERM being decelerated weighs only about 15,000 pounds, the chemical rocket required to effect a 20,000 ft/sec retro is very expensive in terms of initial weight (ΔW_0 = 500,000 lb for nuclear; 1,800,000 lb for chemical). If a storable-propellant retro were used instead of the cryogenic rocket assumed for this comparison, the penalty would be greater. However, the change in initial weight would still be small in the mid-1980's, and there may be configurational advantages to use of a non-cryogenic, high-density propellant. Another possibility is that the retro propellant could also serve as radiation shielding around the crew compartment. This scheme should be investigated when the overall mission-module and reentry-module configurations can be defined.

Effect of Specific Impulse -- The widths of the nuclear-rocket performance bands correspond to a 100-second spread in specific impulse. The mean I_{sp} is assumed to be 800 seconds; the upper bound is for 750 and the lower for 850. The effect of a ± 50 second I_{sp} change is shown to be about a ± 15% change in initial weight. This magnitude of gain or loss is important but does not alter the basic and large superiority of nuclear rockets over chemical rockets.

Weight-Reduction Techniques -- The fact that figure 2 and much of the preceding discussion was based on a specific mission mode should not lead to a conclusion that a mode selection has been made. Preference can be expressed for the one described on the basis of performance and practicality. However, other modes are certainly possible. As a general rule, the alternate modes involve complexities or technical achievements which are currently regarded as uncertainties. Any decision to adopt one of these alternate modes must be accompanied by the decision to develop fully the particular flight technique and technical competence involved. These undertakings will in no case be inexpensive, even in comparison to the cost of the whole Mars mission.

Listed below are five weight-reduction techniques and their corresponding percentage reductions in initial weight in Earth orbit.

Technique	% W_o Reduction
Venus Swingby	to 40
Mars Aerocapture	20-35
Mars Elliptic Capture	20-35
Hyperbolic Rendezvous	5-25
Multi-Vehicle Modes	<10

All the techniques are applied to basic opposition-class landing missions, as previously described, using nuclear rockets for all major propulsion except at Earth return, where an entry-speed limit of 50,000 ft/sec is assumed.

The advantages and limitations of the Venus-swingby Mars roundtrip were mentioned in the discussion of figure 5. The performance advantage in unfavorable years may be impressive (perhaps up to 40%), but launch windows may be restrictive. The usefulness of the Venus-swingby mode must be left open at this time, subject to further study.

Mars aerocapture, on the other hand, is a technique very much like high-speed aerodynamic braking at Earth. A difference is the maneuver which carries the spacecraft back out of the atmosphere and into a circular parking orbit. The velocities, guidance requirements, heating rates, accelerations, etc. are modest. If they were encountered near Earth, they would not be considered formidable problems. The fact that the spacecraft must be designed, developed, and qualified to perform in the Mars atmosphere is the biggest drawback. Another problem is that of providing a stable structural configuration for the entry vehicle, which must include the return and exploration payloads plus the Mars departure propulsion stage. The weight penalty associated with this structure will strongly degrade the potential gains of the technique and may, in some instances, nearly eliminate the advantage. The 30% weight reduction listed is for a relatively small weight penalty.

On the basis of such considerations, Mars aerocapture is held to be a possibility for later missions, when adequate tests can have been made in the Mars atmosphere, but is a problematical candidate mode for early manned flights.

Mars elliptic capture, in which the spacecraft enters an eccentric orbit at Mars rather than a circular orbit, reduces the ΔV's for entering and departing the capture orbit but complicates the landing phase of the mission. Generally a Mars orbiting spacecraft will have the problem of orbit regression. Proper orbit orientation at departure must be assured, and provision for a launch delay will incur a performance penalty. When the orbit is elliptic, both the orbit inclination and the periapsis position must be controlled for departure. Therefore, some of the potential ΔV advantage of elliptic capture will be lost. On the other hand, orbit changes are more efficiently made from an eccentric orbit than from a circular orbit. The net effect is not easily evaluated. Furthermore, the Mars Excursion Module must undergo larger velocity changes, especially in effecting rendezvous with the elliptic-orbit spacecraft. Because the MEM weighs so much less than the main spacecraft, the transfer of ΔV requirements (for orbit circularization) from the large vehicle to the small may result in a net performance gain. However, the use of a lower-Isp ascent propulsion system and provisions for more-sensitive rendezvous timing will reduce the gain. Over-all performance and operational suitability is not clear. The 20-35% W_0 reduction listed is the ideal to which the various penalties must be applied. The utility of elliptic capture orbits will be better known after a more thorough operations analysis of Mars exploration, both on the surface and from orbit.

The other two techniques, hyperbolic rendezvous and multi-vehicle modes, are of lesser interest. In the former the returning Mars spacecraft is met in the vicinity of Earth by an Earth-based re-entry vehicle. Thus, the weight of an ERM need not be included in the payload of the Mars craft. However, the total energy and weight requirements of the two vehicles may be higher than that of a conventional Mars spacecraft; the only gain is that of reducing the weight of the initial vehicle departing for Mars. This is the percentage weight reduction listed. Multi-vehicle modes are typified by one in which a cargo (unmanned) flight follows a low-energy trajectory separate from a manned flight on a faster path. There is little to be gained for the complexity. Neither of these techniques seems worth much consideration, at least for early missions.

Operational Complications -- While we are busily seeking new means of reducing weight, the practical demands of the mission are adding a little here, a little there. Launch windows must be provided -- excess capability so that the mission can leave either Earth or Mars orbit anytime during allotted time periods. Mid-course velocity increments must be imparted by auxiliary propulsion systems to insure extreme accuracy of the trajectory. Non-optimum subsystems and components will ultimately have to be accepted in some systems to keep down the cost of many new developments. Although each such factor makes only a small addition to initial weight, the cumulative effect will not be negligible. Mid-course ΔV's will be in the hundreds of feet per second; launch delay ΔV's may be in the thousands because of the regression of inclined orbits at both Mars and Earth. The over-all effect is likely to be a 10-20% increase in initial weight from the ideal case usually analyzed.

Thrust Requirements

A multi-stage Mars vehicle will include two or three nuclear-rocket propulsion stages: one to depart Earth orbit and one to depart Mars orbit plus another for braking into Mars orbit when the mission mode requires it. Since the payload and gross weight diminishes from one propulsion phase to the next throughout the flight, the thrust requirement also decreases from stage to stage.

As a general rule, for the Mars-propulsive-braking mode, stage thrust requirements differ by factors of two. That is, the Earth-departure thrust should be twice that of the Mars arrival stage; the latter should be about twice the Mars-departure thrust. Such large differences in thrust may be built up from nearly-identical engine and tank modules in cluster form.

Figure 8 shows the relationship between number of engines in the first-stage cluster and initial weight in Earth orbit. The example is one which results in a two-million pound gross weight. The assumptions are made that Earth-departure thrust is provided by a cluster of nuclear-rocket engines and that single engines of the same power are used in both Mars arrival and Mars departure stages. Curves of W_O vs total first-stage power are presented for three values of unit-engine power: 2000, 3500 and 5000 MW. These powers correspond to roughly 100,000, 175,000 and 250,000 pounds of thrust, respectively. The dashed lines are for constant numbers of engines in the cluster, ranging from one to four.

The primary indication of figure 8 is that the first-stage total thrust should be 400,000 - 500,000 pounds (8000-10,000 MW). If each engine has a power of 5000 MW, only two engines would be needed for Earth departure; if the unit power is 2000 MW, a cluster of four engines would be optimum. The effect of the weight penalty associated with engine clustering is the difference between 2.1 and 2.4 million pounds of gross weight. The difference corresponding to a change in engine power from 5000 to 3500 MW is much less -- about a three percent rise in orbital departure weight.

Figure 8 also shows an insensitivity to total thrust, at least on the high side of the optimum. Thus, if other arguments favor the development of a relatively high-thrust engine to provide growth potential or a performance margin, the performance penalty in this application would be small. Another factor not shown is engine operating time. The lower the vehicle thrust-weight ratio, the longer the operating time. For example, the two 5000 MW engines would be at power for about 30 minutes, whereas the four 2000 MW engines would operate for about 45 minutes. Furthermore, Mars arrival with a single 2000 MW engine would entail an operating time of over 50 minutes.

Thus, there are several factors tending to favor a high unit-engine power, including the desire to keep operating times and numbers of engines low and a general policy of aiming high to provide for growth. Balanced against these factors is the increased difficulty and expense of developing high-power engines. Figure 8 indicates that a cluster of 2-3 engines of 3000-5000 MW will be satisfactory in the gross weight range expected for manned Mars landing missions. Corresponding operating times could be held to about 30 minutes if necessary.

Thrust requirements for Mars arrival and departure are taken into account in figure 8, in the sense that thrust for these phases must be provided by single engines of the specified power level. As a result, an acceptable operating time in the second stage means that the third-stage thrust will be higher than optimum for that phase. A cluster of two engines for Mars arrival could have been assumed, but the additional propulsion-system development does not seem warranted by the performance difference.

Figure 9 shows the effect on initial weight of using a low-thrust engine in the Mars-departure stage. The base point is the 5000-MW two-engine calculation from the previous figure. Figure 9 illustrates what could be gained by lowering the power of the large reactor and using appropriate non-nuclear engine components; the curve labelled "large core" shows that there is almost nothing to be gained thereby. Note that the ordinate is a very expanded scale. The shaded area labelled "small core" shows the initial weights corresponding to use of a completely different third-stage engine including a smaller reactor core similar to the KIWI-NERVA core. The vertical width of the band accounts for some uncertainty in engine weight. The resulting initial weight is lower, but the difference of about 100,000 pounds may not be sufficient justification for a separate engine development if such an engine is not needed for other uses. For example, if a suitable small-core engine were available from a lunar application and had been developed to high reliability, it would be selected for the Mars-departure stage because of its reliability.

The gross weight comparisons in figures 8 and 9 indicate that the propulsion requirements of the Mars mission permit considerable flexibility in engine thrust. Neither the design goal nor the actual power attained is particularly critical, provided clustering is feasible, as we now believe. Performance calculations are thus only one element of engine-thrust selection along with many programmatic and policy considerations.

Earth Launch Vehicle Requirements

The initial weights in Earth orbit of manned Mars spacecraft also fix the requirement for a post-Saturn Earth Launch Vehicle (ELV). We have seen that nominal gross weights in orbit are 1.5 - 2.5 million pounds with nuclear-rocket propulsion and a straight forward mission mode. We should also be aware that uncertainties in the mission cause other estimates to vary from under 1.0 to 4.0 million pounds with the same propulsion but other mode and weight assumptions. Thus, from a launch vehicle point of view, Mars exploration will require gross weights in Earth orbit (assembled, checked-out and topped-off) many times the Saturn V capability.

Figure 10 illustrates the members of the Saturn ELV family and their low-orbit payloads. Shown also is a hypothetical post-Saturn ELV capable of boosting 1-2 million pounds into orbit. Only the Saturn V, uprated versions thereof, and the post-Saturn are of interest in a discussion of manned planetary missions.

If the Mars craft has an initial weight of one million pounds, it could probably be assembled in orbit from Saturn V payloads. However, growth potential would be almost nil, and an unfortunate contingency could imperil the entire mission for lack of payload margin. On the other hand, if the gross weight is four million pounds, the larger end of the post-Saturn payload spectrum would be desirable. Such a launch vehicle could be enormous. In between, at an initial weight of about two million pounds, a strong case for some kind of advanced ELV can be put forth. The number of rendezvous with Saturn V's would be too great; a one-million pound payload would be quite satisfactory.

The possibility of using nuclear rockets in Earth launch vehicle stages has been proposed and studied. Three reasons for lack of prime interest in this application have been identified: (1) reduced performance advantage of the relatively-heavy propulsion system when the vehicle thrust weight ratio must be near unity, (2) uncertain compatibility of nuclear engines with recovery and reuse of booster stages, and (3) very high total thrust requirements of post-Saturn propulsion systems. The latter is the most serious. A second stage of a one-million-pound-to-orbit ELV would require a total thrust of at least 3-4 million pounds. A configuration of reactors to provide such a large thrust might very well be impractical. Consequently, boost-phase propulsion is not considered a likely application for nuclear rockets in the time period of manned planetary flight and post-Saturn launch vehicles.

MANNED LUNAR EXPLORATION

In the lunar program, the use of nuclear rocket propulsion must be viewed in a different light because the performance advantage over chemical rockets is less than in planetary missions and chemical systems capable of some lunar exploration are under development. However, performance is not the only criterion; in fact, the benefits to be gained from operational experience with nuclear propulsion build a strong case for early application. Thus, the use of nuclear rockets in lunar exploration must be considered in the context of planetary exploration and future space flight programs. Many additional elements of the lunar program support manned planetary flight, including experience in human factors, life support, orbital and landing operations, and development of systems for long endurance in space. The early introduction of advanced propulsion would be likewise consistent with an over-all policy of providing technical continuity among space program.

A possible application in lunar missions would be a nuclear-rocket third stage on the Saturn V launch vehicle. Substitution of high specific impulse propulsion for the oxygen/hydrogen S-IVB stage would result in a lunar payload gain of 35-65 percent, as will be described later. This gain is large enough to be interesting but is not, in itself, sufficient justification for the development of a new propulsion system and the associated new stages and spacecraft. The programmatic considerations mentioned previously are equally as important as the performance comparison.

Operational Mode

The manner in which the nuclear third stage would be used in lunar missions is illustrated in figure 11. The launch from Cape Kennedy is by means of the first two stages of the Saturn V and possibly a first "burn" of the nuclear stage. The lunar payload and the third stage are injected into a parking orbit. At the correct longitudinal position the nuclear engine starts, and the third-stage payload is injected onto a lunar transfer trajectory. The lunar vehicle proceeds along the dash-line path to the vicinity of the moon where a chemical retro puts the payload either into lunar orbit or directly down on the surface. Meanwhile, the jettisoned nuclear stage, having been separated from the lunar vehicle at the end of nuclear thrust, travels along the dash-dot path behind the moon and, with the aid of a lunar-gravity kick, out into a heliocentric orbit. The radioactive nuclear engine is thus neatly dispatched without requiring additional thrust.

I referred to the possible use of the nuclear engine prior to parking-orbit injection, that is, before reaching orbital energy. This mode is called suborbit start and is the same mode as planned for the S-IVB in the all-chemical Apollo mission. This mode requires the third-stage engine to restart for orbit departure. In the nuclear-rocket application, suborbit start entails engine aftercooling with hydrogen to remove the heat of radioactive decay. The after-cooling period is only during the stay in parking orbit -- hopefully just a matter of hours -- so the amount of propellant used for this purpose is small. Furthermore, the thrust produced by the cooling hydrogen need not be wasted but can be used to raise the energy of the orbiting vehicle. Suborbit start involves additional flight safety requirements not prevalent with orbit start. However, the development of a reactor destruct system would prevent a malfunction from causing any radiation hazard.

The alternative of orbit start would require the first and second stages to put the entire third stage and payload into a low parking orbit. The nuclear engine would operate for the first and only time during injection into lunar transfer orbit. Flight safety would be much easier to achieve, perhaps requiring no destruct system, but payload could be reduced.

The use of the nuclear-rocket propulsion system for braking into lunar orbit is another possibility although aftercooling propellant loss would be relatively large and cryogenic storage during the lunar transfer would add insulation and boiloff weight penalties. Furthermore, the lunar velocity increment is small and the third-stage engine relatively heavy for efficient use of nuclear-rocket propulsion. The complication of aftercooling and the impaired lunar-orbit operations (due to the presence of a radioactive engine) must be balanced against the small gain in lunar payload.

Performance Comparison

The extent of the payload advantage resulting from substitution of a nuclear third stage on the Saturn V is shown in figure 12. Lunar landed payload is plotted against weight propelled to lunar transfer velocity. The upper line is for direct-landed cargo, that is, the payload of an unmanned lunar logistic vehicle. Parallel to it and slightly lower is the line of manned direct-landed payloads. The difference is due to guidance and shielding differences and may not be a real difference. The horizontal broken line corresponds to the minimum landed payload to provide return capability for three astronauts. Thus the intersection of the Apollo/Direct and Return Liftoff lines defines the minimum vehicle weight for direct-flight capability. Oxygen/hydrogen propulsion is assumed for all lunar landing and take off stages.

Along the abscissa are the lunar-transfer payloads of several three-stage Saturn V configurations. At the far left is the standard, all-chemical Saturn V. The nearly 95,000-pound injected weight corresponds to 28,000 pounds of landed cargo. This launch vehicle does not provide manned direct-landing capability. Several points are shown for Saturn V upratings, with or without a nuclear-rocket third stage. S-NA refers to an orbit-start nuclear version; S-NB has a suborbit-start nuclear stage. The NERVA engine is utilized in both nuclear stages. In neither case is any uprating or significant modification of the first two stages assumed. The point labelled All-Chemical Uprated is one of a nearly-infinite array of improved-performance possibilities based on chemical upratings of the first two stages, this one being about a 40 percent uprating. This could be achieved by increasing the thrust of the first and second stage propulsion systems, increasing stage propellant capacities, and strengthening the vehicle structure. At the far righthand end of the scale is a point corresponding to an orbit-start (S-NA) nuclear stage on the same uprated lower stages.

Any of the improved-performance Saturn V configurations would provide manned direct-landing capability. This mode is considered to be an important step in proceeding toward extensive lunar exploration, particularly because it permits manned operations at any lunar latitude. Thus, an uprating of the Saturn V will provide a unique capability. Further improvement beyond the minimum for manned direct landing will increase the payload capability, thereby adding landed cargo to a manned flight (as noted on figure 12) or allowing an increase in the size of the crew. The result would be expanded lunar exploration or fewer launches for a given level of activity.

The introduction of nuclear-rocket propulsion is viewed as an excellent method of uprating Saturn V performance for lunar as well as deep space missions and, at the same time, acquiring valuable operating experience and establishing flight confidence in a new area of rocket propulsion. Improved performance of the first two Saturn V stages would, of course, increase Earth-orbit payload as well as lunar payload. Therefore, the combination of lower-stage uprating and a nuclear third stage is of considerable interest. Figure 12 shows a lunar-landed payload of 60,000 pounds for this configuration, which uses a single NERVA engine in an orbit-launch stage.

UNMANNED SOLAR SYSTEM MISSIONS

Another important missions area in which nuclear rockets may find application is unmanned solar system exploration. Emphasis in this area will increase, not only due to scientific interest, but also because many of the missions will provide support for manned planetary ventures.

The particular missions which would benefit from the use of nuclear rockets include: Mars and Venus orbiters, Jupiter and Saturn flyby probes, Solar probes (close approach and extra-ecliptic), and interplanetary probes (e.g. solar system escape). These applications are characterized by requirements for either large payloads or very high velocity increments. My discussion will be further limited to applications of the nuclear Saturn V described in the lunar program section since it is the first system that can be provided using hardware that is available or close at hand.

Figure 13 depicts one of the prime destinations, Mars, and lists some typical payloads. The weights put into Mars orbit are shown in the table at lower right. Three Saturn V third stages are listed: S-NA (orbit-start nuclear), S-NB (suborbit-start nuclear) and S-IVB (suborbit-start, chemical). Three injection stages are considered: MMM (an oxygen/hydrogen multi-mission module), SM (the storable-propellant Apollo Service Module) and S-N (the same nuclear third stage, cooled and restarted). The table shows that use of a nuclear-rocket third stage increases Mars orbit payloads by 45-80 percent over all-chemical propulsion. As in the lunar case, reuse of the nuclear stage at destination offers little advantage and probably does not warrant development of aftercooling and restart capability. Although the payload composition cannot now be specified, one possibility is that a SNAP-8 system, probably to power a television system, could take up to 100 percent of the all-chemical useful payload. After inclusion of an appropriate weight of instruments and probes and allowing for structure and other essentials of the mission, an orbit payload of 40,000-50,000 pounds may be necessary.

The flyby payloads are for flyby round trips, and the listed payloads are the weights returned to the vicinity of Earth. This mission mode could be used in the unmanned program for special purposes, such as sample and data return. Although these payloads are too low for single-launch manned flyby round trips, it may be possible to cluster stages or put them in tandem to achieve the required injected payload.

Figure 14 summarizes for a variety of missions the performance gains due to substitution of a nuclear third stage. A suborbit-start nuclear stage (S-NB) and a cryogenic fourth stage (MMM) are assumed. In all cases the gains are about 70 percent. The Jupiter and Saturn missions are flybys without return to Earth. The solar probes include a close-approach mission to 0.2 of an astronomical unit and an out-of-the-ecliptic mission to an inclination of 25°. A probe to the outer reaches of the solar system is represented by the bar labelled Solar System Escape. The payload magnitude shown may be very desirable in order to provide adequate power and communications capability.

SUMMARY

This discussion of nuclear rocket missions has emphasized manned planetary exploration as the principal application and justification for nuclear rocket development. We have dwelt at some length on the characteristics and requirements of manned Mars roundtrips. We have emphasized the difficulty of doing an adequate job of planetary exploration with proper performance margin and growth potential. The use of nuclear rocket propulsion has been shown to permit planetary exploration missions with reasonable initial weights in Earth orbit, without reliance on questionable modes or technologies, and with the ability to perform the mission at all planetary opportunities.

Among the many items contributing to the feasibility of manned planetary flight, the development of nuclear rockets is the most important. At the same time, it is close at hand, as will be indicated in the later discussions of our development program. Depending on orbit departure weights, Mars missions may require the development of nuclear rocket engines in the thrust range of 150,000-250,000 pounds. Engine clustering will probably be necessary as may a post-Saturn launch vehicle because of the magnitude of spacecraft gross weights. Current programs give assurance that these nuclear rocket propulsion systems and post-Saturn launch vehicles can be developed.

Extended manned lunar exploration could make good use of a nuclear-rocket third stage on Saturn V to permit a direct-flight manned exploration capability. Such a configuration would also provide increased capability for many unmanned solar-system missions. Furthermore, early experience with nuclear rockets would contribute to the reliability and operational readiness of propulsion systems for manned planetary flight. The Saturn V nuclear stage should have a thrust of approximately 50,000 pounds, or slightly more, as typified by the NERVA engine currently undergoing investigation. Operating times would be in the range of 20-30 minutes, and no more than one restart would be required in flight operations.

Consequently, the applications picture is composed of (1) early uses of nuclear rockets in which they contribute valuable performance gains and operational experience, followed by (2) use in manned planetary missions in which they affect initial weights by factors of 2-3, thereby keeping orbital weights within reason. If lunar and planetary operations expand, there will be demands for heavier payloads, shorter trip times to the near planets and, perhaps, manned flight capability to more distant planets. Nuclear rocket propulsion can serve advantageously in this phase of space flight also, thereby making a vital contribution to our space goals of exploration and scientific understanding.

Advanced Propulsion Applications

In conclusion, I would like to indicate briefly the potential impact of advanced high-thrust propulsion systems on space flight capabilities. Many propulsion concepts have been proposed and research programs are underway to determine their feasibility. The prominant concepts are (1) cavity reactors, with the nuclear fuel contained in the form of dust, liquid or gas and (2) nuclear pulse propulsion, which utilizes the energy release of explosive charges to propel the spacecraft. Electric-thrust concepts and controlled-fusion, direct-thrust propulsion are in the low-acceleration category and, thus, require a somewhat different treatment in a comparison with high-thrust schemes. The major parameter with low-acceleration, power-limited propulsion systems is the ratio of engine weight to power, and that could be an entire lecture of its own.

Figure 15 shows the important relationships which determine the relative performance of high-thrust propulsion systems. Payload fraction -- that is, the ratio of payload to initial weight -- is plotted against engine thrust-weight ratio. The curves for an I_{sp} of 850 seconds are indicative of what we can expect for well-developed solid-core nuclear rockets. Most 400-500 day Mars roundtrip trajectories involve individual-stage ΔV's in the 15,000-20,000 ft/sec range, resulting in payload ratios of 0.45-0.35. Higher ΔV's would enter the region in which two stages would do better than one. For example, a 300-day roundtrip to Mars would involve propulsive ΔV's of 25,000-35,000 ft/sec per stage, and the low payload ratios would result in sharply increased gross weights compared to the values quoted for 400-500 day trips. A factor of two reduction in payload ratio in each of three propulsion stages would be a factor of eight in initial weight.

The question is, what could be accomplished with a system producing a specific impulse of 2000 or 5000 seconds? Figure 15 shows that, at 2000 seconds I_{sp} and an engine thrust-weight ratio of unity, a 300-day Mars trip would have the same range of individual-stage payload ratios as a 450-day trip with 850 sec I_{sp}. At 2000 seconds I_{sp} and an engine thrust-weight ratio of 10, the entire 300-day mission could be done with a single stage. The total ΔV would be about 90,000 ft/sec. In comparison, a specific impulse of 5000 seconds combined with high specific thrust would be revolutionary. Unfortunately, the prospects are very slim for attaining this level of performance in the foreseeable future.

The importance of engine weight is a significant factor which has been frequently overlooked in considerations of advanced propulsion concepts. The pair of curves for 2000-seconds I_{sp} reveal the sensitivity of performance to engine thrust-weight ratio. A ratio much less than unity would eliminate a large part of the potential advantage of the high specific impulse. At an I_{sp} of 1500 seconds, a thrust-weight ratio of 10 is comparable to unity at 2000 seconds I_{sp}; a thrust-weight ratio of unity at 1500 is not much better than two stages at 850 seconds I_{sp}.

Therefore, evaluation of the usefulness of high-I_{sp} concepts must generally await sufficient knowledge of the systems to estimate engine weight. The solid-core nuclear rocket, being relatively well defined at this time, serves as a basis for comparison. Accordingly, because of engine-weight effects, a number of advanced concepts are faced with the job of proving not only feasibility but satisfactory engine thrust-weight ratio as well.

Perhaps the best thought on which to end is this: While the demands of space missions far in the future will ultimately be satisfied by advanced propulsion systems, the major space missions beyond the first phases of manned lunar exploration will be the domain of the solid-core nuclear rocket.

III. THE NERVA ENGINE

The NERVA engine represents the 50,000 lb. thrust class nuclear rocket engine which uses a solid core graphite reactor. This is the size engine we have chosen to carry out our early nuclear rocket development efforts. It will be our first nuclear rocket engine and will ultimately be our first nuclear rocket engine used in space missions. The reactor designs for NERVA rely on the KIWI reactor technology. The KIWI reactor efforts began in 1955 at the Los Alamos Scientific Laboratory. However, nuclear rocket engine development began in 1961 with selection of the industrial contractors, Aerojet and Westinghouse, to develop the NERVA engine. As part of the development effort, the program planned for flight testing a nuclear rocket engine.

At the end of 1963 the nuclear rocket program was reviewed and redirected towards a ground engine technology program. This redirected program allows us to concentrate resources and technical and management attention upon critical components such as the nozzle, reactor, turbopump assembly, and engine system operating and control characteristics. Such a program will, therefore, provide the information needed to permit flight system development to be undertaken with confidence when missions beyond Apollo are better defined.

The nuclear rocket program today includes major effort on the reactor, work on engine technology and on supporting technology. The reactor design, development and test effort has been conducted by the Los Alamos Scientific Laboratory and the Westinghouse Astronuclear Laboratory. The LASL part of the work includes the remaining tests of the KIWI-B4 reactor, and the Phoebus project, which is an advanced graphite reactor technology effort. Engine technology work is conducted within the NERVA project by the Aerojet-General Corporation with the Westinghouse Astronuclear Laboratory as principal subcontractor for the nuclear subsystem. This effort includes adaptation of a KIWI reactor design for engine operation and investigation of engine system characteristics.

Engine Reliability vs. Performance

Engine efforts have been based upon a ground rule that reliability is more important than weight and performance. The first nuclear rocket engine can provide almost twice the specific impulse of advanced chemical rockets. Since a big step increase in performance is possible with the first engine, it does not seem important to squeeze the maximum possible specific impulse out of the system if reliability may be sacrificed thereby. One example of this ground rule is that while the highest reactor exhaust temperature is desirable, the average exhaust temperature is penalized in our early designs to simplify reactor structure and flow paths. This penalty is caused by flows, used to cool the reactor structure and the reactor periphery, which are mixed with the hot fuel element exhaust gas. The attendant specific impulse penalty is preferred to the complications of a design with regenerative cooling paths at this early stage of the development work.

Another example of reliability through simplicity is selection of the hot bleed cycle wherein hot exhaust gas is diluted to lower the temperature to obtain turbine inlet gas compatible with turbine material capability. The turbine exhaust is subsequently routed overboard, again lowering engine specific impulse. Higher specific impulse is obtainable through the topping cycle; however, the reactor design complexity does not warrant attempting to obtain the higher specific impulse for that cycle.

Some weight penalty was accepted through the use of aluminum rather than titanium in the pressure vessel. We decided that the benefits of greater experience with aluminum alloys was more important than the lower weight theoretically possible with titanium.

Development Program Philosophy

Nuclear rocket engine development is being conducted as a technology program. Our efforts are being devoted to development of critical engine components which significantly affect engine system characteristics. Engine system tests are to be conducted with components and configurations that are not necessarily flyable but contain the essential components that determine system characteristics, including the dynamic operating characteristics.

The program includes component, subsystem, and system tests. The progression from component through system testing represents increasing complexity. Our philosophy in nuclear rocket development is to predict and understand component performance and operation before going to more complex subsystem tests.

The reactor is the critical engine system component and has paced the development program. The reactor is tested as a major subsystem at the Nuclear Rocket Development Station in Nevada. The reactor test depends upon a facility to provide liquid hydrogen coolant under pressure to the test assembly which consists of the reactor, pressure vessel and nozzle. Prior to reactor testing, its components are tested in the laboratory simulating as well as possible the reactor environment to ensure, as far as possible, that reactor components and the complete reactor design will behave as predicted.

Non-nuclear engine components are also tested. These components are tested at various industrial and government labs and do not require remote test operation. Components under test include the turbopump assembly, nozzle and control system components. The next level of test complexity will be engine system testing. Engine system tests will include the reactor subsystem closely coupled to the turbopump with the reactor providing the energy to drive the turbopump. These engine systems tests will include cold flow and power testing.

Our test facilities were built to satisfy the needs of reactor and engine testing. However, facility characteristics limit the test operation when compared to conditions during nuclear rocket engine operation in space. For example, the nozzle, exhausting to the atmosphere in the reactor test cells, results in back pressure which prevents the nozzle from flowing full at low chamber pressures. The facility limitations must be considered during low flow operation of the reactor. The engine test stand provides a liquid hydrogen run tank above the engine and the engine fires in a downward direction. The stand provides some altitude simulation capability so that test conditions are somewhat closer to space operation.

Our ability to maintain equipment remotely affects engine development. Current test techniques require commitment to finish a test once a reactor has a significant power history. Remote handling techniques are adequate to disassemble a test article, but remote maintenance and reassembly are not practical with today's capabilities. For example, the KIWI-B4D reactor test was terminated after a nozzle hydrogen leak. Nozzle replacement involves carefully seating seals, fastening about 75 bolts with a torque wrench, and hooking up several dozen instrumentation channels. We are unable to do this remotely today and there are no prospects to improve our capability here in the very near future. However, the engine is more complex than the reactor test assembly with more items which will need maintenance before a test series is completed. It appears feasible to conduct remote maintenance on some of the lines, valves, and feed system in the experimental test system. This will provide limited remote maintenance capability for experimental engine test systems which is a requirement for reasonable engine development.

THE NERVA ENGINE DESCRIPTION

The NERVA engine is our first nuclear rocket engine. Its current performance goals are to provide about 50,000 pounds thrust at greater than 700 seconds specific impulse. The engine uses a graphite reactor operating at 1000 megawatts nominal thermal power.

The engine is intended to be a self-contained propulsion package. This means that it has the ability to start on command without additional power or control, other than the electrical power for the control circuits. The engine must restart in similar fashion, which means that provisions are needed for shutdown and subsequent cooling.

Even though NERVA is a ground experimental engine system technology project, I will discuss the flight engine design which serves as a reference nuclear rocket engine. This will indicate all components which make the nuclear rocket engine a self-contained propulsion package. The experimental ground engine system, which is being developed to investigate engine system technology, will also be described and differences from the exact flight configuration will be apparent.

Configuration Description

Flight engine configuration development has been underway since 1960. Major decisions during this period included selection of the reactor design concept, the engine cycle to provide the energy source for propellant pressurization, turbopump design, and the use of pneumatics for control and valve actuation. The considerations which influenced these decisions will be discussed.

A mockup of the NERVA engine concept on which we have been working is shown in Figure III-1. The engine stands 22 feet high from the top flange, which mates to a hydrogen propellant tank in the rocket vehicle, to the exhaust exit of the jet nozzle. The reactor is located within the pressure vessel in the central portion of the engine.

Twelve pneumatic reactor control drum actuators are attached to the pressure vessel dome. The thrust structure, which transmits thrust and acceleration loads between the pressure vessel and rocket vehicle, is composed of upper and lower subassemblies connected by a gimbal which allows the engine to swivel for thrust vector adjustment. The turbopump is mounted within the lower thrust structure and the upper thrust structure contains the propellant tank shutoff valve.

The large spheres contain hydrogen gas under pressure to provide for actuation during engine startup periods.

The nozzle consists of a convergent-divergent section, cooled by the main propellant flow, and an additional divergent skirt which increases the nozzle expansion ratio to whatever value may be desired.

A pump discharge line connects the pump to the nozzle inlet manifold. A turbine inlet line connects the hot bleed port, where nozzle chamber gas is tapped, to the turbine inlet. Roll control thrust, if required, could be provided by roll control nozzles using the turbine exhaust.

Flow Description

Figure III-2 shows a drawing of a nuclear rocket on which we can follow the propellant flow paths typical of a solid core nuclear rocket. The flow paths shown are main propellant flow, turbine drive flow, and pneumatic gas supply.

Main Propellant Flow -- During steady state operation, main propellant flow begins with propellant, under tank pressurization, passing through the tank shutoff valve into the pump suction line. This line, which contains a gimbal bearing for thrust vector adjustment, provides propellant to the pump inlet. A centrifugal flow pump pressurizes the propellant. The pressurized propellant enters the pump discharge line and flows to the nozzle inlet manifold. The main propellant flow passes through the nozzle cooling passages, removing heat transferred to the nozzle from the main exhaust stream as well as heat generated in the nozzle pressure vessel by deposition of nuclear radiation energy. The coolant leaves the nozzle as a low temperature, low density fluid and is split into parallel flows to cool the pressure vessel, reflector and control drums. Propellant exits from the reflector region and is directed along the pressure vessel dome to remove radiation energy deposited there.

The flow then cools the shield and enters the reactor inlet plenum. Propellant is distributed from the inlet plenum into several parallel paths. The bulk of the flow enters the reactor fuel element cooling passages and is heated to high temperature. The remaining propellant is distributed to flow passages which provide coolant to various reactor structural elements and to the peripheral region between the hot reactor core and the regeneratively-cooled reflector. These various cooling flows merge in the reactor exit plenum which is also the nozzle inlet chamber. The propellant is expanded through the nozzle to exhaust velocities greater than 23,000 feet per second corresponding to more than 700 second specific impulse, typical of a nuclear rocket. The main fuel element exhaust is at a high temperature and provides greater than the average engine exhaust temperature.

Turbine Drive Flow -- In the NERVA engine, a hot bleed cycle is used wherein turbine drive flow will be provided by mixing the hot main drive gas, tapped from the nozzle inlet chamber, with cold diluent fluid taken from a suitable part of the main flow path. The diluent flow is mixed with the hot bleed flow, providing gas cooled to any temperature desired for turbine inlet conditions.

The mixed gas flows through the turbine inlet line, passing through a turbine power control valve that regulates turbine flow and turbine inlet pressure and, therefore, the turbine power and pump speed. The flow is expanded through the turbine which extracts the power required to maintain the turbopump speed and operating point required.

The turbine exhaust is expanded through nozzles to add a small contribution to the engine thrust. This thrust contribution, from gas at relatively low temperatures, lowers the overall engine specific impulse below the specific impulse obtained from the main propellant flow. This use of turbine exhaust fluid as indicated in Figure III-1 could be a source of vehicle roll control thrust if that is desired. An alternate use of turbine exhaust would be to provide cooling for a nozzle skirt extension, if required, and then exhausting it overboard to provide a small thrust contribution.

Pneumatic Gas Supply -- The pneumatic system must supply actuation gas from the engine during operation and from pressurized gas storage bottles during startup operations. This subsystem, while made up of components within our technological capabilities, is complex. The pneumatic subsystem is, however, essential in providing a self-contained propulsion package.

The pressurized gas supply for the pneumatic system is drawn from a suitable point in the main propellant flow path. The gas must be filtered and regulated to supply constant pressure fluid to actuators for reactor control drums, turbopump control valve, thrust vector control, roll control, and tank shutoff valve, etc. Check valves and additional regulators can be arranged to use the gas stored in pressurized bottles during startup and to recharge the storage bottles during engine operating cycles for restarts.

Tank pressurization can be provided from the pneumatic supply by flowing gas at a relatively constant supply pressure to a tank pressure regulator which admits gas to the propellant tank to maintain its required pressure level.

Ground Experimental Engine System Configuration

Although flight engine designs have been studied to define configurations, component requirements, and problems, the NERVA project is not yet undertaking full development of the flight engine configuration described above. Work on critical engine components and ground experimental engine testing is planned and proceeding so as to establish the technology and operating understanding of these systems before flight application. A ground experimental engine system (XE) configuration is being selected to investigate engine startup characteristics and component interactions during startup, power operation, and shutdown.

The XE engine concept is based upon the following considerations:

a) The XE engine will use NERVA engine components where component characteristics have an important influence on overall system characteristics. However, component development and reliability requirements will be relaxed to be consistent with the technology objectives of the program. An example is the need for continuing development of a flight type turbopump because component mass and inertia will influence chilldown and acceleration characteristics.

b) Facility-type components will be used to the extent possible where flight-type designs are not required for component interaction and system dynamics tests. An example is replacing the pneumatic gas storage spheres and pneumatic supply system with a facility gas supply. Other examples include a facility valve in place of the flight propellant tank shutoff valve, elimination of the adjustable gimbal, and use of a simpler thrust structure which provides more room for component mounting.

c) An external shield will be added to the configuration to protect engine components. This will eliminate the need to assure full radiation hardening of all components before the experimental system tests are conducted. However, the shield design concept will allow reduced attenuation as components are demonstrated to be capable of withstanding the nuclear radiation levels.

CONFIGURATION DESIGN AND DEVELOPMENT

This section discusses the engine cycle selection and the design and development status of the major components of the engine.

Cycle Selection

The choice of the turbine drive cycle in the nuclear rocket engine is one of the principal system design selections required. The nuclear rocket engine in Project NERVA uses a hot bleed cycle wherein hot gas is tapped from the nozzle chamber, diluted with a cold gas, and passed through the turbine to drive the turbopump. The turbine exhaust is dumped overboard with some thrust recovery. This cycle was selected from several alternatives. Alternatives included pressurized gas cycle, chemical gas generator system, topping cycle, and various bleed cycles, hot bleed, cold bleed, and heated bleed. Figure III-3, which will be discussed shortly, shows the bleed cycles.

The gas pressurization cycle consists of a gas pressure system to expel liquid hydrogen from a propellant tank through a flow control system to the reactor. This system, while extremely simple and highly reliable, is dismissed easily because the high inlet pressure requirements of a nuclear rocket engine and the large volumes associated with storage of low density liquid hydrogen propellant would yield extremely high tank structural weights and result in an impractical system.

The chemical gas generator cycle uses a turbopump to deliver propellant at design pressure and flow rate. Tank pressurization is needed only to hold the propellant under sufficient pressure to provide adequate suction head at the pump inlet. This chemical cycle supplies turbopump driving energy with a chemical gas generator burning the liquid hydrogen with liquid oxygen. Combustion products are expanded through a turbine to extract power to drive the turbopump. The main advantage of a chemical cycle is that the turbopump is driven by an independent power source, thereby eliminating the system integration problem of extracting energy to drive the turbopump from the reactor. Reasonable performance can be achieved with such a system. The main disadvantages of a chemical system are: a separate propellant, liquid oxygen, is carried and complicates stage design; the engine system is complicated by the gas generator and the need for a pressurized feed system to supply the liquid oxygen to the gas generator; and, engine weight and number of components are increased, complicating the design. The chemical gas generator cycle was not selected because these complications appeared to limit the usefulness and, possibly, the reliability of the engine in operational applications.

The remaining cycles use reactor heat energy to drive the turbopump. Such an approach appears obviously reasonable because of the large source of such energy available. These cycles are the topping, hot bleed, cold bleed and heated bleed cycles, all using gas heated by the reactor.

The topping cycle provides the highest specific impulse obtainable from a turbopump driven nuclear rocket engine cycle. In this cycle, propellant flows through the pump, nozzle cooling passages, and reactor reflector. Almost all propellant is then passed through a low-pressure ratio turbine. Some propellant would be bypassed around the turbine to provide for speed control. Turbine exhaust and the bypassed propellant are then passed through the reactor core and expanded in the nozzle. The energy for operating the cycle is obtained from heat transfer within the reflector or, if that heat addition is insufficient, a second core region could be obtained by adding uranium fuel to the reflector material. The main advantage of this cycle is that all propellant is exhausted at reactor exhaust temperatures so that no specific impulse penalty is imposed by the turbopump cycle. An additional cycle advantage is that the turbine operates at low temperatures, allowing the use of low density aluminum alloys for construction. This would tend to minimize radiation energy deposition in the turbine if that becomes a problem. The major disadvantage to the topping cycle is that a sufficient energy source is needed to heat all propellant to the turbine inlet temperature. This affects the reactor design, hence reactor and feed system interactions remain which can only be investigated in system tests. An additional complication is caused by the turbine pressure drop which occurs between reflector outlet and core inlet. This pressure drop increases the strength and sealing requirements for the flow separation structure between the core and reflector. The KIWI-B4 reactor, selected for nuclear rocket engine development, was not designed to provide the necessary heat pickup. Selection of the topping cycles would have complicated reactor development efforts; therefore, the cycle was discarded, at least for the first generation nuclear rocket engine.

The bleed cycles, illustrated in Figure III-3, all involve tapping hydrogen from various points in the main propellant flow path and expanding this fraction of total propellant flow through the turbine and then exhausting it overboard. The differences between the bleed cycles are the different bleed locations.

In the cold cycle, hydrogen is bled from the pressure vessel dome. The energy pickup in the gas up to this point includes only heat transferred in nozzle coolant passages and in the nozzle reflector. The main advantage of this cycle is that the bleed gas is relatively low temperature and components are not subjected to different environmental conditions. Aluminum alloys can be used in the turbine, and simple turbine machinery can be employed. The major disadvantage is that a relatively large fraction of the main propellant flow is needed to power the cycle and a significant specific impulse penalty is imposed by this cycle. In addition, it is desirable to start the nuclear rocket by bootstrapping, wherein the latent heat of the reflector provides the energy to accelerate the turbopump. The cold bleed cycle would not provide as much acceleration margin as would be desired for a bootstrap start in vacuum. Therefore, the cold bleed cycle was discarded because of the high performance penalty at steady state conditions and the lack of startup acceleration margin.

The heated bleed cycle is a variant of the cold bleed cycle where gas from the cold bleed port is routed through the internal shield and a nozzle skirt extension where additional heat is transferred. The heated hydrogen is ducted to the turbine to provide turbopump power and is exhausted overboard. The heated bleed cycle removes many of the performance penalties and acceleration margin limitations of the cold bleed cycle. The additional component development of the nozzle skirt does not appear difficult. However, the cycle's disadvantages are:

(1) The heated bleed cycle depends, for operation, on a large area altitude type nozzle. Therefore, all engine and nozzle development testing must be conducted in a high altitude facility, complicating development testing and increasing test facility costs and design problems;

(2) Feed system performance becomes interdependent with reactor performance, shield heating, and nozzle transfer characteristics. The system interactions cannot be investigated with component testing alone in simpler and less costly programs than system tests, and increases the uncertainties which exist during system testing.

(3) Additional design criteria are imposed upon the shield and nozzle in that not only must heat be removed satisfactorily, but the coolant must also be raised to a minimum temperature. The heating in these components must be determined accurately for this cycle.

The need for a high area ratio nozzle, shield cooling uncertainties, and the system test complications led to discarding the heated bleed cycle for the first engine.

The hot bleed cycle depends upon extracting hydrogen at reactor exhaust temperature from the nozzle chamber. The hot gas is immediately cooled by diluent flow to a temperature compatible with turbine material capability. The mixed gas is expanded through the turbine to provide turbopump energy. An advantage of this cycle is that bleed gas flow rate is determined by allowable turbine inlet temperature and pressure rather than by the heating rates available in components. This cycle can provide a relatively small specific impulse penalty; however, turbine inlet temperature can be readily varied to tradeoff turbine reliability against performance. The nozzle and its bleed port can be developed as a separate component. In addition, turbine fluid characteristics can be experimentally investigated during chemical firings of the nozzle and also reactor power tests. The disadvantage of this cycle is that the drive gas is extracted at reactor exhaust temperatures. This imposes an extremely difficult design and development problem for the bleed port and the turbine inlet line. These components will be susceptible to hot spots and burnout unless a sound design is obtained. The hot bleed cycle was selected for the engine development program because the problem of providing turbine inlet fluid can be isolated into a component development which can be investigated in both nozzle and reactor testing. In addition, the system does not need a high area-ratio nozzle, eliminating the essential need for high quality altitude facilities for early system testing, although altitude tests of an engine required to operate at altitude are considered essential. Finally, high turbopump reliability and early system tests can be conducted at low turbine inlet temperature but with the option of later increasing the turbine inlet temperature to provide a capability for engine performance growth.

KIWI/NERVA Reactor

The reactor is the major new subsystem that must be developed in nuclear rocket systems. As might be expected, therefore, it has been pacing nuclear rocket engine development.

The reactor design and development progress are discussed in this section.

General Reactor Design and Selection -- The reactor design, selected for application to the NERVA engine, is a version of the KIWI-B4 reactor which was designed and is being tested by the Los Alamos Scientific Laboratory. The reactor assembly includes reactor core, reflector, control drums and internal shield. A schematic illustration of the basic reactor is shown in Figure III-4. The reactor is composed mainly of graphite using graphite fuel elements impregnated with uranium carbide. The reactor core is made up of clusters of these fuel elements and is supported by both a lateral and an axial support system. These support systems must accommodate large changes in core dimensions arising from core thermal expansion, while providing support for the static and dynamic loads imposed on the core. A hot end seal is used in the KIWI reactor to prevent major amounts of flow bypassing the core. The NERVA reactor uses a distributed seal arrangement. The outer reflector is made of beryllium. Twelve rotary control drums made of beryllium with a boral sheet subtending 120° of arc are located in the reflector and used to control the reactor.

Reactor Operating Characteristics -- Design specific impulse and thrust are obtained only when the reactor operates at design point. However, the engine and the reactor must operate stably over a wide range of conditions. Generalized limitations to steady state operation of a nuclear rocket reactor are discussed herein.

Reactor limits are of two types. There are limits which, if exceeded, cause irreversible changes or reactor damage. These are limits of allowable temperatures or structural load. Other limitations, imposed by core neutronics, involve operating conditions where behavior is either unknown or where steady state operation cannot be maintained. In some instances, this type of limit can probably be exceeded during a transient.

General trends of typical steady state reactor operating characteristics are shown as a generalized reactor operating map in Figure III-5. The general trends of reactor exit temperature are plotted against exit pressure. The operating limitations imposed by various reactor conditions are indicated.

The maximum allowable exit gas temperature is obviously limited by fuel element material temperature limits. Reactor structure, heated primarily by heat conduction and radiation (thermal and nuclear) may limit the allowable temperature in the low chamber pressure regime where structural cooling flow may be insufficient to remove the heat conducted to the structure.

Core structural loads vary with core pressure drop. Limitations due to maximum core pressure drop affect operation in the high pressure, high temperature portion of the operating map.

Neutronics effects cause concern rather than causing definite limits for core operation. For example, a minimum core inlet temperature limit is indicated in the figure to avoid possible problems due to liquid hydrogen or high density hydrogen entering the core and affecting power distribution, reactivity, and control. A limit of constant reactivity increase is presented to indicate the operating regime where the hydrogen in the core increases reactivity to the point that reactor control shutdown margin approaches zero. Operation beyond the limitation can only be transient since power, hence exit temperature will increase, forcing propellant out, bringing steady state operation back within the operating map. The reactor operating conditions are therefore generally bounded as follows:

a) high exhaust temperature - fuel element temperature;

b) low flow, high temperature - structural temperature;

c) high flow - core structural load limit;

d) low temperature - neutronic limitations to avoid high density;

e) low temperature, high flow - constant reactivity limit to avoid elimination of control span shutdown margin.

The magnitude of the effect of these limits on operating range of a particular reactor will, of course, depend on the particular design considered. These reactor limits will affect the stable engine operating range but it is certainly possible that non-reactor components can be more controlling in parts of the operating regime. These other engine operating limits will be discussed further in the system characteristics discussion.

Reactor Development Status -- Reactor research and development is being conducted in the KIWI and the NERVA projects. This discussion of reactor progress will cover both efforts. The KIWI reactor work is aimed at establishing the basic reactor technology and sound design concepts. The NERVA work is aimed at establishing flight suitable designs based on the KIWI concepts and engineering optimization of that design to achieve the maximum performance of which the system is capable with high reliability.

Substantial progress has been made in the reactor area since the Los Alamos Scientific Laboratory (LASL) started its research and development work in 1955. This early work led to testing a series of KIWI-A reactors which, as indicated in Figure III-6, gave important design, materials, control, and nuclear characteristic information. These KIWI-A tests were run in 1959 and 1960.

In the KIWI-B1 series of reactor tests run in 1961 and 1962, (Figure III-7) LASL showed that these reactors could be effectively controlled by control drums in the reflector of the reactor. It was also shown that these reactors could be operated with liquid hydrogen as a coolant and with a liquid hydrogen cooled nozzle, as would be required in flight rocket systems. During this time, Los Alamos scientists and engineers developed methods of fabricating the uranium bearing fuel elements, they developed inspection techniques, they developed an understanding of the effects of the high temperature on the nuclear characteristics of the reactor, they automatically started reactors with liquid hydrogen as the coolant. Such items marked the substantial progress and advancement made during the years since the program started.

Our preferred reactor design and the one intended for NERVA application, the KIWI-B4A reactor, was tested in November, 1962 with resulting damage to the reactor core due to flow-induced vibrations. A short movie of that test run shows the flashes in the jet that indicated graphite damage in the core. During 1963 extensive work was performed to identify, explain, and solve these vibration problems. The reactor program was oriented toward redesign, by LASL and Westinghouse, of the reactor core support structure and toward extensive component subassembly, and full reactor analyses and tests, including cold-flow reactor tests (flow tests in which no uranium fuel is contained so no fission energy is generated), to uncover the source of vibration and damage that had occurred in November 1962 and to avoid it. We now know that our redesigns do not encounter the core vibrations.

One of the important tests run last year was the cold-flow KIWI-B4A reactor test in May 1963 (Figure III-8) to obtain conclusive information that vibrations had indeed occurred in the KIWI-B4A power test of November 1962 and to obtain an understanding of the cause of these vibrations. This cold flow test was needed because core mechanical vibration instrumentation cannot be incorporated into a power reactor so that definitive vibration data were lacking. As had been hypothesized, flow induced vibrations occurred even without any power being generated. This test demonstrated,also, that burning of the hydrogen leaving the jet nozzle, separated flow in the nozzle, two-phase flow in the reactor, and other features of the early start-up portion of the operation were not the cause of vibrations. This test clearly proved that the vibrations were flow-induced.

This test was followed by a second cold-flow, KIWI-B4 experiment (KIWI-B4B-CF) in August 1963, which included simple fixes simulating the redesign features that were being incorporated into the KIWI and NERVA reactors to prevent the vibrations. It was found that with these redesign features, no vibrations occurred under the cold-flow conditions of the tests.

I do not want to leave the impression that cold-flow, full, reactor tests were the only type employed in examining reactor design and operational problems during last year. Figure III-9 shows a reactor flow test rig at Los Alamos, known as the KIWI-Pie because it is a pie-shaped slice of a full reactor. It serves as a tool for studying the flow paths, vibration characteristics, and effects of redesign of reactors and has added to confidence in the redesigns. Tests in this KIWI-Pie have also provided information on the flow-induced vibrations. Similar tests were run on NERVA using a small axi-symmetric section of a reactor core.

Another type of component test is shown in Figure III-10. This is a photograph of test equipment being used for vibration testing of a group of fuel elements at Westinghouse. The shaker is located in the lower portion of the photograph. Fuel elements are contained in the shiny structure. Another Westinghouse component test fixture (Figure III-11) has the capability for flow and thermal shock testing of reactor control drums and control-drum actuators. Full core vibration tests have also been conducted at Westinghouse. The test rig for shaking the core is shown in Figure III-12. Tests such as this one are used to provide design data and evaluation of designs with respect to mechanical vibrations.

Testing, on a sample basis, of fuel element production lots includes hot gas tests to evaluate fuel element flow and corrosion resistance. The Los Alamos hot gas test rig, used to evaluate fuel elements, is shown in Figure III-13. The furnace is capable of temperatures up to 3600°C (6500°F) and pressures up to 1500 psi with flowing hydrogen. Westinghouse conducts similar tests in their facilities in Pennsylvania. These laboratory tests are extremely important in developing and assessing the life of fuel element materials and designs and are a major factor in estimating reactor life before reactor tests.

One of the tests aimed at assuring suitable reactor control capability, flow distribution, power distribution, and temperature distribution through the reactor is shown in Figure III-14. This is a critical experiment test setup of the NERVA reactor in which nuclear data were obtained at low neutron flux levels or essentially zero power. It is an important check of the nuclear characteristics of the reactor which must be determined before a power test of the reactor is run. Power distribution and shutdown control span are a typical parameter measured during the experiment.

Many other flow, structural, nuclear and mechanical tests have been conducted and some are still continuing. This enumeration of only a portion of our reactor development work is intended to illustrate the thorough approach that has been taken to assure integrity of our reactor design and to seek out all problem areas before power testing the reactors. These tests also provide a means for reactor development that is not completely dependent on full system power tests.

The Los Alamos redesigned version of the KIWI-B4A reactor was tested first in the KIWI-B4D cold-flow reactor tests completed by Los Alamos on February 13 of this year. No core vibrations were detected in these tests of the Los Alamos redesign. Analysis of test data and visual examination of that reactor indicated some minor mechanical problems that resulted in minor modification to the reactor design, but the overall design and operation were excellent.

During March and April of this year, the NERVA reactor design, NRX-A1 (Figure III-15) was also cold-flow tested by the NERVA contractors, Westinghouse and Aerojet-General. The Westinghouse reactor also ran well, encountered no vibrations, and looked excellent on disassembly. Again, minor mechanical problems noted in these tests were corrected in the power reactors.

These KIWI-B4D and NRX-A1 cold-flow tests were extremely important accomplishments. When combined with the other component and subsystem tests, they supported our confidence that the reactor [illegible] design and fabrication work being conducted by both Los Alamos and Westinghouse would result in a reactor that did not encounter vibrations a[illegible] [illegible]ountered in November 1962 power test of the KIWI-B4A.

Although we had conducted extensive analyses and tests of all parts of our reactors, including some tests that closely simulated actual operating conditions for these reactor parts, all parts of the system are simultaneously subjected to all of the operating stress, temperature, and flow conditions only in the power tests. In spite of our painstaking work, it would, therefore, not have been unusual if some unexpected engineering problems were uncovered. We were confident that, if any occurred, they would not be of a basic nature.

However, it was most encouraging when the Los Alamos KIWI-B4D reactor (Figure III-16) was tested on May 13, 1963 at power and temperature values that exceeded the planned conditions and were close to design values without encountering any significant reactor problems. The operation of the reactor was completely successful. No fuel elements were cracked and the reactor structure operated as it was designed. The following short movie taken during the power operation of the KIWI-B4D reactor shows how clean the exhaust jet is compared to the earlier KIWI-B4A test run. This indicates the successful operation of the core and the lack of core damage.

The test time was shorter than planned because of a hydrogen leak that occurred in the jet nozzle. Fortunately, this nozzle failure did not compromise our test objectives. The causes of the nozzle leak have been narrowed to one or two possibilities. We are modifying the nozzle to eliminate these possibilities. Although it involves a difficult technology, the nozzle is not an area that affects the basic developability or availability of nuclear rockets.

In addition, the KIWI-B4E (Figure III-17), was operated successfully as another significant milestone in our development program. The test was conducted at planned power flow and temperature conditions for a test duration of over 8 minutes at high power, the maximum time possible with the available propellant supply. The successful operation of the reactor during this test indicated our reactor design is fundamentally sound and suitable for use in a nuclear rocket engine. The following infrared movie shows the power test of the KIWI-B4E. On infrared film the exhaust jet is clearly defined. This film gives a good idea of how long an eight minute firing is.

These tests are major milestones in this country's program to develop advanced rocket propulsion and nuclear rocket propulsion in particular. It will serve as a firm base for the development work that is to follow. Work is still needed to make these systems operate for even longer times and to design and develop them to higher powers. The reactor tests to be run during the rest of this year and the reactor tests to be run next year, including important laboratory tests, will further evaluate the Los Alamos design and will also test the Westinghouse design. This work is aimed at demonstrating operation at longer durations and higher powers and temperatures than those achieved in the KIWI-B4 tests.

Propellant Feed System

The propellant feed system includes a turbopump assembly, tank shutoff valve, turbine power control valve, and associated lines. The purpose of this total system is to provide propellant at the desired pressure and flow rates to the nozzle, pressure vessel, reactor assembly.

Design and Selection -- The turbopump assembly consists of a single stage centrifugal pump driven by a multi-stage axial flow turbine. The pump increases the liquid hydrogen propellant pressure from the tank storage pressure to nozzle inlet pressure at a flow rate of about 70 lbs. per second. Power transmission from the turbine to the pump imposes a significant bearing design and development requirement. A turbopump design similar to the one shown in Figure III-18 is being developed for use in the experimental ground test engine system. Bearings for the turbopump assembly are cooled and lubricated with hydrogen.

The turbine power control valve regulates the turbine drive fluid flow rate. Hot gas turbine inlet flow is controlled through the full valve range from open to closed. The valve is actuated pneumatically and is shown with an actuator attached in Figure III-19.

A tank shutoff valve is provided to keep propellant from reaching the pump until engine operation is desired. The two-position valve is either closed with low or no leakage, or open with low pressure drop at full propellant flow. The valve has provisions for remote connect-disconnect capability for ground testing. However, the tank shutoff valve requirements are not closely coupled to the engine dynamic characteristics. Therefore the experimental ground engine system will use either a low pressure drop facility type valve or an early development model tank shutoff valve.

The turbopump assembly and turbine power control valve have a major effect in establishing the dynamic operating characteristics of the engine. For this reason, development of flight type units is being carried out for use in ground testing the experimental engine system.

The centrifugal flow pump for the engine system was selected from the choice between centrifugal flow and axial flow. While it appeared that either pump type could satisfy the engine requirements at the steady state design point, the centrifugal flow pump offers a wider operating range. Since nuclear rocket engine operating characteristics are difficult to predict at this time with great certainty before extensive testing, the centrifugal flow pump was selected to assure that a single pump design will provide propellant flow over a wide range of possible operating conditions, thus avoiding an operating range limitation. Some further design modification of the centrifugal pump is being carried out to provide the widest possible range of negative characteristic slope. This slope, when zero or slightly positive, may result in a total system instability which could be difficult to avoid. Every effort is being made in design to avoid such a possible condition and to provide broad operating margins.

<u>Operating Characteristics</u> -- A generalized centrifugal flow turbopump operating map is presented in Figure III-20. Limitations caused by pump stall, net positive suction head and a turbine power limit indication at steady stage are shown. Possible engine requirements are superimposed on the operating map. The engine requirement covers only a small portion of the allowable range of pump operation; however, the wide operating limits provide assurance that turbopump characteristics will be compatible if component characteristics require modification during the development process.

<u>Development Status</u> -- The turbopump and turbine power control valve have been under development since shortly after the NERVA Project began. Tank shutoff valve development has been conducted as a slower effort since it does not appear to be a critical item for the experimental engine investigation.

Tests of the turbopump have been conducted and the turbopump operating characteristics have been evaluated. Although design point performance is satisfactory, the range of stable operation and the shape of performance curve is being studied to assure that sufficient operating flexibility is available. As mentioned earlier, the turbopump is being modified to expand the stable operating range and to improve the shape of the performance curve. Tests have been run with the turbopump with and without a simulated pump suction line.

Ambient pressurized gaseous hydrogen has been the drive fluid for turbopump testing. Hot gas tests with the turbopump have been conducted using hydrogen-oxygen combustion products. The test facility is being modified to provide hot hydrogen as the turbine drive fluid.

An important part of turbomachinery development for nuclear rocket applications is development of bearings that will operate satisfactorily with hydrogen cooling under radiation environment. The work accomplished to date in this area represents a significant advance in high speed bearing technology. Figure III-21 shows a bearing test fixture in which bearing configurations used in the turbopump were run for several hour at the loads expected to be imposed in an actual turbopump installation. In addition to running these bearings in the test fixture, significant bearing time has been logged in actual turbopump tests. These same bearing configurations have also been tested in a nuclear radiation field at the Nuclear Aerospace Reactor Facility (NARF), which General Dynamics operates for the Air Force. Preliminary tests of 15-20 minutes have been encouraging in this radiation field. Longer time testing is planned in our program.

The turbine power control valve is being tested to prove satisfactory operation when subjected to hot gas flows. The valve is also tested in conjunction with its pneumatic actuator to assure satisfactory actuator cooling while the valve is handling hot gas, and to prove that temperature gradients do not distort the valve or impair operation.

Thrust Chamber Assembly

The thrust chamber assembly includes the nozzle assembly which consists of a nozzle and nozzle skirt extension, the pressure vessel, and the thrust structure. The thrust chamber assembly is simplified for ground tests of the experimental engine system by eliminating the nozzle skirt extension and replacing the thrust structure with a single semi-monocoque structure. The modified thrust structure consists of an inverted, truncated, conical skin section which is stiffened by ribs.

The nozzle has proven to be a difficult practical development in our nuclear rocket work. A nuclear rocket nozzle is fabricated to include a regeneratively-cooled convergent-divergent flow path for reactor exhaust gas expansion to high velocity. The nozzle also provides a pressure shell for the convergent section to withstand the loads imposed by reactor exhaust pressure. The nozzle is cooled by the main propellant flow path to maintain nozzle materials at an acceptable temperature even though exposed to the high temperature reactor exhaust gas and to remove heat from the nozzle pressure shell due to nuclear radiation energy deposition. The nozzle design must consider fully:

a) the high heat transfer rates from the main exhaust stream to the nozzle coolant tubes due to high heat conductivity of hydrogen;

b) the use of hydrogen as propellant, which requires exclusion of air from the nozzle and prevention of any hydrogen leakage;

c) the large contraction ratio in the convergent section to provide a transition from the reactor outlet diameter to the throat diameter. This high ratio, peculiar to nuclear rockets, results in large tangential and longitudinal stresses which must be contained by the nozzle pressure shell.

Our current efforts in nozzle design and development include:

a) efforts to estimate, more accurately, heat transfer characteristics from the hot exhaust to the nozzle coolant leading to determination of temperatures and stresses in nozzle coolant tubes;

b) investigation of alternative designs and materials to provide added margins between operating temperatures and materials capabilities;

c) determination of energy deposition rate in pressure shell and assurance of adequate cooling provisions to maintain the pressure shell at an acceptable temperature; and

d) investigation of fabrication and quality assurance techniques which allow fabrication and assurance that nozzles are built as required to withstand all operating conditions. This practical area represents our major problem area. As part of this problem, the difficulty of dimulating operating conditions should be pointed out here, although it is discussed in the Advanced Engine Section.

Figure III-22 is a photograph of a nozzle fabricated by Aerojet-General for use in NRX reactor testing. The nozzle is shown with an adaptor and chemical fuel injector used to conduct chemical simulation firings. Figure III-23 is a view of an alternate nozzle design fabricated by Rocketdyne for the KIWI reactor tests. This nozzle can also be used for NRX reactor tests if needed. Although this design will present some improvement over the nozzle used in the KIWI-B4D test, some of the fabrication and inspection problems remain.

Nozzle development for the nuclear rocket is proceeding with a significant amount of effort devoted to it. However, we cannot yet say we have obtained a nozzle design with sufficient demonstrated reliability to meet all of the reactor and engine test requirements including reasonable flight type operating capabilities.

The remaining components of the thrust chamber assembly are the pressure vessel and thrust structure.

The pressure vessel consists of a cylindrical shell and an upper dome closure bolted to it. Ports are provided for pass-throughs for instrumentation leads and control drum actuator shafts.

Pressure vessels have been fabricated both with titanium and with aluminum alloys. The units have been subjected to hydrostatic tests and closure seal tests. Both designs proved acceptable. The aluminum design was selected because of our greater familiarity with the material. A photograph of an aluminum pressure vessel is shown in Figure III-24.

The thrust structures, both upper and lower, are designed and fabricated of stainless steel. The structure supports and transmits thrust and boost loads between the engine and stage and contains provisions for mounting the turbopump assembly and tank shutoff valve. Figure III-25 shows a view of the thrust structure mounted in a fixture for dynamic testing. A simpler unit will be used for the ground tests of the experimental engine system.

Control System

The engine control system must maintain stable engine characteristics during steady state operation in the power range. In addition, the system controls the engine during transient operations such as startup, normal shutdown, and restart, if required for the mission. This system must provide, where practicable, for operation with component degradation or malfunction. A nuclear rocket control system contains control loops to maintain reactor exhaust gas temperature and reactor exhaust gas pressure which, in turn, determine engine specific impulse and thrust.

The NERVA engine control system details have not been fixed yet; however, typical systems have been investigated. A representative pressure and temperature control scheme is discussed below.

Pressure control can be achieved by comparing measured reactor exhaust, i.e., nozzle chamber pressure, with the pressure level demanded by a programmer. The error signal provides a basis for positioning the turbopump control valve with a pneumatic actuator. The valve position changes flow to the turbine, which thereby changes pump speed, flow, and pressure output.

Temperature control can be achieved by trimming the output of an inner loop which controls reactor power. Reactor power can be controlled by comparing actual neutron flux (proportional to reactor power) with desired flux. The error signal provides a basis for positioning the twelve reactor control drums with pneumatic actuators. The desired power is modified by an error signal generated by comparing chamber temperature with desired temperature. The resulting temperature signal provides an additional power demand signal input to the power control loop. The total temperature correction allowed in the power loop can be limited to a fraction of the demanded power, to avoid a major power increase in the event of loss of the temperature signal.

A block diagram of a typical nuclear rocket engine control system is presented in Figure III-26. The pressure feedback control for flow as well as the reactor temperature and power control loops are indicated. The investigation of various operating modes in the ground engine test program will define the control mode to be used in flight systems.

Control system development is proceeding. Control circuits are being packaged for installation in the test facilities, but packaging these circuits for flight hardware will not be required until flight engine development is undertaken. Actuators for reactor control drums and the turbine power control valve are also being developed.

Pneumatic systems have been selected to provide the actuation capability. Pneumatics were selected because pressurized hydrogen gas is available within the engine cycle itself and pneumatic systems tend to be more radiation-resistant than hydraulic systems. Figure III-27 shows a development actuator unit undergoing environmental testing.

Control sensor development is a key problem area. Efforts are being directed to obtain sensors which operate reliably in the reactor environment. Our needs include a wide range neutron flux measurement, a radiation-resistant pressure sensor, and a quickly responding measurement of the high temperature reactor exhaust gas.

SYSTEM CHARACTERISTICS

Component characteristics discussed above are combined to obtain system characteristics that are discussed in this section. Some transient characteristics of the engine are included. In addition, facilities involved with system development are described and development status of the engine system is presented.

Operating Characteristics

Steady State -- Typical trends of steady state operating characteristics of nuclear rocket engine systems are presented on an engine operating map, in Figure III-28. Trends on the chamber temperature vs. chamber pressure plot are generally similar to the ones describing reactor limits. However, representative limits imposed by the turbopump and the nozzle are added. The map shows a regime for steady state operation where components are not limiting. The regime runs from approximately 100 percent power at design point conditions down to approximately 40 percent of design. Flight performance will be at maximum specific impulse and thrust; however, some ground test operations can be conducted profitably at lower power conditions. As you will note in Figure III-28, the limitation to engine operation is determined mainly by the reactor. Turbopump stall may be less critical than reactor structural cooling requirements as indicated, however the relation between these limits may be different for different reactor designs and the relation may vary as component characteristics are modified. The nozzle wall temperature may limit a small region of high temperature-partial flow conditions and the turbopump control valve flow area may affect a region of high flow partial temperature operation.

Transient -- In addition to a need to operate stably within the steady state map, we must find satisfactory operating lines for the engine to follow during all startup and operating conditions without exceeding component limits. The engine must bootstrap, that is, use the latent heat stored within the reflector and reactor core to provide the energy source for feed system acceleration. Nominal studies of startup transient operation in vacuum indicate the fundamental feasibility of bootstrat start. However, realistic startups are complicated by engine childown characteristics preceding pump acceleration, and possible intermittent flow choking in the nozzle cooling passages during start. We are also concerned about back pressure effects during starts in the ground test facilities.

Engine chilldown is being studied in experimental investigations. These investigations are being conducted to determine desirable arrangement and sequencing of cryogenic valves. Some of our concerns in this area include the need to avoid turbine overspeed, acceleration margin of the turbopump, and overchilling of the reflector and core which lowers the energy available for bootstrap start.

During the early portion of engine start, gaseous hydrogen flow may be choked in the nozzle coolant passages. The resultant high pressure drop in the nozzle lowers available turbine inlet pressure and affects bootstrat acceleration margin. This effect is being evaluated.

Facility limitations to testing are also of concern. For example, the altitude simulation system in ETS-1 cannot maintain low pressure during the early period of start and Test Cell A testing is conducted without altitude simulation. The effect of these non-ideal back pressure conditions on engine exhaust and on turbine exhaust are being investigated. The higher back pressures reduce bootstrap acceleration margin and we must be assured that sufficient margin remains to conduct adequate startup tests.

Studies are continuing and experimental data are being obtained which will lead to better understanding of the startup transient, particularly in the initiation of propellant flow, where we lack knowledge of turbopump acceleration characteristics and where hydrogen boiling and two-phase flow are significant. Similar transient studies are being conducted to determine engine system characteristics during shutdown to assure that the control system will avoid conditions which exceed component limitations. Restart studies are also significant in that the initial conditions of reactor temperatures are different than during normal startups.

Development Status

Engine system experiments involve fabrication, assembly and testing three experimental system designs that terminate with an experimental nuclear rocket engine system tested in a downward-firing, simulated-altitude-capability engine stand.

The first system is a cold flow version of the nuclear rocket engine. This will include the critical components of the engine in a close-coupled assembly. The reactor, however, will be assembled with non-fueled components, so that no power will be produced during cold flow tests. This system, the Cold Flow Development Test System (CFDTS), will be used to investigate the early portions of the engine start where turbopump acceleration characteristics and engine chilldown phenomena cannot be predicted accurately.

The CFDTS testing will be conducted in the H-Area test complex of the Aerojet-General Corporation. This complex, shown in Figure III-29, has several test positions for nozzle and turbopump component testing as well as system test capability. The system test position is underneath the large liquid hydrogen run tank. Testing underway at the Lewis Research Center of NASA is also providing information about transient characteristics of generalized nuclear rocket engine. This work is described in the section on our Advanced Research and Technology work.

After completion of the CFDTS test series, the system will be used at the Nuclear Rocket Development Station (NRDS) in Nevada, for preparing the Engine Test Stand for power engine testing. All engine system power tests will be conducted at NRDS.

At NRDS we have made substantial progress in providing the necessary facilities for nuclear testing in the Nuclear Rocket Program. These facilities represent a national capability that is not duplicated anywhere else in the United States. The major facility items currently in existence or now under construction are two reactor test cells, "A", and "C"; an engine test stand, ETS-1; the Reactor Maintenance Assembly and Disassembly Building, R-MAD; an Engine Maintenance Assembly and Disassembly Building, E-MAD; and an administration area.

The second engine system is being designed to investigate engine system characteristics by modifying the reactor configuration tested in the reactor test cells. We plan to install a turbopump and hot bleed nozzle on a test car in conjunction with an NRX reactor test. The assembly is then an engine system experiment; however, the engine fires upward with the nozzle exhausting into the atmosphere. The start transients are obtained under high backpressure conditions. This test system will allow us to study startup characteristics and obtain operating data in parts of the steady state engine operating map.

Testing will be conducted at Test Cell A, shown in Figure III-30. The test cell has storage provisions for liquid hydrogen and gaseous propellants. The propellant feed system, that pressurizes propellant for reactor testing, is bypassed for the engine system tests. Control systems and the data acquisition systems used for reactor testing are considered applicable for the engine system testing.

The third engine system assembly will be used in downward firing tests in an altitude engine test stand. This engine (the XE engine) is modified only slightly in appearance from the flight version of the NERVA engine. The XE engine will not have thrust vector capability or an internally contained pneumatic system. Altitude simulation capability of the test stand is shown as a trend of test cell pressure vs. engine chamber pressure in Figure III-31. The altitude capability provides higher pressure ratios across the turbine and allows faster turbopump acceleration. Bootstrap start in the altitude simulation stand will more closely simulate startups during flight than is possible in the reactor test cells.

The test facility for the XE engine tests will be Engine Test Stand No. 1 (ETS-1) shown in Figure III-32. This facility provides downward firing, and an altitude engine test chamber in which the engine is enclosed during the firings. A 70,000-gallon, liquid hydrogen, run tank is located above the engine to provide flow conditions which approximate propellant flow to a nuclear rocket engine during flight.

IV. ADVANCED RESEARCH AND TECHNOLOGY

ENGINE SIZE SELECTION

The long lead time required for the acquisition of test facilities as well as other technological requirements lead to early establishment of size and performance level goals for the next generation of nuclear rocket reactors. Technological, programmatic and mission implications of the selected goals must be thoroughly considered to assure that nuclear propulsion will be available where it is needed when it is needed. This means that we must know the propulsion system performance that will be required to achieve the missions planned in the next ten to twenty years and as far beyond as possible.

Nuclear Engine Clustering

Clustering nuclear rocket engines extends our ability to provide nuclear propulsion systems for a variety of missions. Clustering also extends our ability to accommodate to changes in performance requirements that may occur. This is extremely important since estimates of the required overall vehicle weight in orbit to make a manned landing on Mars are based on a number of mission assumptions whose validity will not be established for some time.

Comparatively little is lost furtheremore from not having an exactly optimum thrust level in individual engines of clustered nuclear rocket engines. This was indicated earlier in the discussion of vehicle applications. Generally, we believe, it is desirable to limit the number of engines in a cluster to about four to simplify vehicle development. A factor in favor of larger engines is that some increase in engine thrust-to-weight is possible as reactor diameter is increased. So we must pick an engine which is small enough so that most of what we have learned from KIWI and NERVA can be applied to its development, large enough so that we extract as much as possible of the performance gain of bigger reactors, and of a size which will permit the greatest possible range of mission application.

Thrust Requirements

Our studies of the most probable mission of interest after lunar exploration have indicated that the power levels required for the three major propulsive periods in a manned Mars expedition generally are in a fixed ratio regardless of the assumptions used for the mission. It appears reasonable to provide a separate nuclear powered stage for each of these periods. The ratio of the thrust levels needed for Earth orbit departure, braking into a Martian orbit, and Mars orbit departure are approximately four to two to one. This implies that a single nuclear rocket engine, properly clustered, could provide the propulsion required for all firing cycles. An important point to emphasize is that the experience gained in the KIWI and NERVA development makes it possible to predict the time of availability of nuclear rocket propulsion systems and the development effort required. This work will provide a firm basis for developing future systems to meet the needs of manned expeditions to Mars and other future space missions.

From the curves of initial weight in orbit versus power level which you have seen earlier (Figure I-8 and I-9), it appears that interplanetary missions will start in Earth orbit with a spacecraft weighing between 1.5 to 3 million pounds. The optimum first stage thrust level for this initial weight in orbit is then in the range of 450,000 to 900,000 pounds thrust capability. Keeping in mind that the payload required to perform a mission usually grows with time and that follow-on extensive exploration should be possible with the same system, it would not be surprising if somewhat higher thrust levels will be desired by the time such a Mars mission is actually performed. A total thrust of up to a million pounds is, therefore, probably indicated.

Phoebus Reactor Power Level

Since the mission studies of clustered engines indicate great flexibility in engine size, it was decided to establish our next generation engine as the highest thrust engine which can be built using essentially the design concepts, control schemes, and material capabilities demonstrated in the KIWI and NERVA engine. This decision then deliberately avoids difficult new and inventive technology requirements. It is also large enough so that doubling or tripling the payload requirement would not present clustering problems of great difficulty. On the other hand, should our mission for some reason presently unforeseen require a total thrust level much smaller than that chosen, we will need to cluster fewer engines. On this basis, our Phoebus reactor design power level has been chosen at a nominal 5,000 megawatts which would give a nominal engine thrust of 250,000 pounds. Units of two to four could meet our present estimates for Earth departure thrust and either a single or pair of these engines would be suitable for braking into a Mars orbit. This power level is a nominal goal. The flexibility afforded by clustered nuclear rockets adds a safety margin so that obtaining a lower thrust does not seriously compromise mission capability as has been shown in the mission discussion.

Phoebus Development

Other performance goals for the Phoebus reactor technology program are to achieve the longest possible fuel element operating life and the highest possible temperature. We are now able to achieve fuel element lifetime adequate for many nuclear rocket missions and appear to have the technology for improved lifetime close at hand. Longer operating times permit the multiple testing needed to achieve reliability with few engine builds. Restart is needed for the same reason. Except for reusable ferry-type applications, no missions we have considered show any significant gain by more than one restart. Higher gas temperature or high specific impulse is of such great value to the performance of missions that it is always a goal.

Several factors, including the time needed to design, fabricate, and develop larger core reactors, and the lead time for test cell facilities and ground test support equipment prevent us from testing a large Phoebus reactor for several years. We have therefore layed out a program which will allow us to learn about our Phoebus reactor goals using only our present facilities and KIWI-type or size hardware. This program includes several tests in the present KIWI size - reactor cores, called Phoebus-I, in which we will attempt to evaluate important elements of the Phoebus technology. These tests will be started next year.

Before proceeding with the description of the efforts now underway on the Phoebus reactor and the technology for engines based on that reactor technology and describing some of the problems that may have to be solved in their development, I would like to mention the major factors which determine how far we may extrapolate the KIWI/NERVA size and power level before encountering severe development difficulties.

Size and Power Limitations

The KIWI/NERVA is a fairly small diameter graphite reactor. This small size imposes some design difficulties and limitations. From a neutronic standpoint almost any change in the core is difficult. The introduction of new materials, a change in position of materials, increases in void fraction or power density, all require careful consideration of their effects on reactivity. In a large core, however, the decreased surface to mass ratio results in less neutron leakage which in turn results in smaller amounts of fuel per unit volume required for criticality. This additional reactivity available in larger size reactors may be used in several ways. For example, it may be desirable to introduce series cooled or regeneratively cooled structure in places where KIWI or NERVA have used parallel cooled structures. Such regenerative coolant helps to increase attainable Isp. The additional structural material needed to provide a regenerative cooling path would be difficult to introduce in a small core because of the absorptive capture of the additional materials. It is also possible to replace structural materials used in the small core of the KIWI with materials having more favorable structural properties, but with slightly higher neutron poisoning effects which might rule out their use in the smaller core. Another way in which this increased reactivity may be used is in providing high void fractions in large cores. Since void fraction is directly related to power density in these cores, this would produce a lower engine specific weight.

A significant design consideration in determining reactor diameter is reflector control. An in-core control system would produce large flux depressions in areas adjacent to the control rods and would require internal cooling. The development and demonstration of reflector control systems in the KIWI test series was one of its major accomplishments. Reflector control is achieved primarily by converting fast neutrons into thermal neutrons and metering the fraction which is returned to the core. As the reactor core [illegible]meter is increased and fuel is removed the neutron spectrum becomes substantially softer or less energetic [illegible] (Figure IV-1). The decrease in neutron energy results in a relatively less effective reflector control system. Increases in system pressure [illegible] hydrogen density which also [illegible] thermalize the core [illegible] effectiveness, [illegible] increase [illegible] it was possible [illegible] establishment of the Phoebus reactor [illegible] development of an entirely new in-core control concept, which would be a major development job is therefore avoided.

Another way to increase system performance which appears attractive at first glance, is to go to higher pressure systems. Since power density is directly proportional to pressure, doubling the pressure is equivalent to doubling the power in a given reactor core. At first glance, this seems to be an attractive method of extracting greater performance. Careful consideration of the overall engine system response to pressure increase shows, however, that not only are our performance and technology requirements for all components including the non-reactor components made substantially more difficult, but the overall engine weight is not reduced by high pressure operations.

Figure IV-2 and IV-3 are generalized curves of turbopump weight versus reactor pressure and overall engine system weight versus reactor pressure for a given engine. The data used to plot these curves are approximate and represent considerable extrapolations of existing technology. They are, however, indicative of the general trend of change in weight as a function of pressure. In general, the combined weights of the nozzle and pump rise with pressure faster than reactor weight decreases so that there is no advantage by going to higher and higher chamber pressure. Therefore, pressure levels should be chosen on the basis of available or readily developable component technology.

Phoebus Technology Program

The program to provide technology for future generations of nuclear rocket reactors, known as the Phoebus program, includes the design, development, fabrication and test of several reactors in both the KIWI/NERVA size and in a larger size. The initial work in this program began over a year ago with experimental measurements of neutron physics parameters in honeycomb criticals such as this one shown here at the Los Alamos Scientific Laboratory. (Figure IV-4) These simple critical experiments use slabs of fuel, structural material (graphite) and poison materials to get a first estimate of some of the neutronic design parameters. More sophisticated physics experiments known as Zepo's (zero power) which use fuel and structure in the exact form in which it will be used in the reactor, will be carried out as the program progresses and as more precise estimates are required. These simple honeycomb experiments, however, are excellent for determining in a gross sense the effect of introducing additional structure or new materials to the core.

As the Los Alamos Scientific Laboratory has completed their work in the KIWI project, they have shifted their efforts to this Phoebus technology program. As of this time, LASL's work is almost entirely devoted to this advanced effort.

ENGINE TECHNOLOGY

We have begun the development of the technology of major non-reactor components for an engine which would use the Phoebus reactor technology. Certain of these components, such as, feed systems and nozzles, are required for the conduct of reactor tests. There is obviously a close relationship between nozzle and feed system design and performance and the reactor design and performance characteristics. The nozzle problems for advanced Phoebus reactors will be in many ways similar to the KIWI and NERVA nozzle designs. It should be noted that the heat flux from the nozzle is of the same order of magnitude as that inside the reactor. We have been carrying out heat transfer and fluid flow studies and determining the basic properties of hydrogen over the entire temperature range of interest for several years. Much of what we have learned from KIWI and NERVA is available to us here and provides an excellent base of information.

Nozzle Development Testing

One significant problem is nozzle development testing. Exact simulation of operation with a reactor is difficult and we have not accomplished such simulation. Figure IV-5 and IV-6 show a facility which should be available about the middle of next year. It will be the first facility which we have had in the nuclear rocket program which will permit testing of reasonable scale nozzles in hot hydrogen for any period of time. Obviously, in order to provide exactly identical environmental conditions for reactor tests we would need a heat source capable of producing the same power as the reactor during the hot run. When one considers the available heat sources, electrical power requirements become excessive. In fact, one very soon comes to the conclusion that an ideal heat source for testing nozzles is a nuclear reactor--not a very promising or realistic situation at this time. We have instead used hydrogen-oxygen combustion as a test technique but the lower film coefficient of heat transfer for hydrogen-oxygen gas mixtures mean that we must go to much higher pressures in H_2-O_2 firings to achieve the same heat flux conditions. This in turn means that we have either different structural loads or different temperature conditions than would exist during the actual rocket reactor nozzle operation. This facility at the Lewis Research Center will use hot gases to heat graphite balls to a high temperature. The heat capacity of these balls is used to heat 30 pounds per second of hydrogen flow, for tests of up to 25 seconds duration.

Feed Systems Development

We are considering several ways of meeting our requirements for increased hydrogen flow for the Phoebus reactor testing. If it is possible to increase the performance of our present pumping system, the NFS-2 being used in KIWI/NERVA reactor testing, our requirements can be met by coupling two or more in parallel. Ways of modifying this pump to permit the higher performance level required are now under study. Figure IV-7 is a photograph os such a turbo-pump system. We are also giving some consideration to meeting our facility feed system requirements for higher power reactors by modifying pumps under development for high thrust hydrogen-oxygen chemical engines.

Radiation Effects

The nuclear rocket engine systems advanced technology work relies heavily on chemical rocket engine technology. Our nuclear engine technology activities devote extensive effort only at those specific problems peculiar to nuclear rocket engines. Our other development needs are common to H_2-O_2 engines and we draw on this reserve of knowledge when necessary. For that reason, our research in this area is heavily oriented towards the specific problems posed by radiation: tank heating, pumping boiling hydrogen, operating high speed bearings in a radiation field, the development of pneumatic control circuit (radiation resistant) and the determination of materials properties in a combined radiation and cryogenic environment.

There are two factors which will enter heavily into Phoebus which have not been major factors in the KIWI/NERVA design. The first of these is structural damage from radiation effects. In the KIWI and NERVA development, we were able to ignore the classical radiation effects which must be considered by other reactor programs. This is because rocket reactors typically operate for fairly short times by reactor standards. The total dose is therefore below the range where substantial transformation is found. The classic problems of induced crystal structural disorder are therefore not found in KIWI/NERVA. We have instead a class of unusual and difficult problems resulting from the combination of very high radiation heating rates which when combined with our liquid hydrogen coolant create troublesome temperature gradients and temperature asymmetries. While this problem remains with us in the higher power Phoebus reactor, we also reach total doses at the threshold of the range where changes in engineering properties may be anticipated for many materials. As a consequence, we must now give careful consideration of the materials which we choose from the radiation damage standpoint.

Clustered engines are also subject to radiation originating in adjacent engines. This will result in increased dose to control systems and feed system components due to neutrons and gamma rays which emanate from the sides of adjacent engines. We do not anticipate any unusual effects in clustered nuclear engines which might inhibit their use in space missions. A number of critical experiments are being carried out to obtain experimental confirmation of these analyses. Clustered engines may have some partial shielding on the sides facing other engines to reduce this radiation to tolerable levels. (Figure IV-8) In order to keep shield weights to a minimum, we will have to know the failure limits of all the components and materials affected. Some of the facilities and equipment needed to make these determinations are now available for components and materials of interest. For some components and materials, work is already underway.

The prospect of high doses and high dose rates in single units or in clusters of high power density nuclear rocket engines has caused us to make a substantial investment in test facilities capable of testing materials and components of the nuclear rocket in closely simulated environments.

Plumbrook Reactor

This is a picture (Figure IV-9) of the NASA Plumbrook Reactor Facility. The test reactor characteristics of this reactor are shown in this slide. (Figure IV-10) Plumbrook Reactor Facility is an MTR type test reactor designed specifically to provide high flux over large test volumes for space system development testing. As you can see, its major features are its high neutron and gamma flux, and the large diameter of the experimental holes. The last feature is particularly important for the kind of testing needed for components of nuclear rocket engines. In this testing, as I have mentioned previously , the major problem from radiation is the high heating rates generated in the relatively massive parts of components, for example, in the magnetic cores of amplifiers or in actuators. Superimposed on this high energy deposition rate is a high energy removal rate by virtue of the liquid hydrogen flow through the system and the excellent thermal conductivity of materials used. Mounting a transducer against a 50^{o}R aluminum pipe in a 10^{11} ergs/gm C-hr. gamma heating rate generates unusually high temperature gradients. The problem is particularly severe in rotating or sliding components such as actuators or drive motors where changes due to differential thermal expansion may cause binding of moving parts, and in instrumentation. In order to determine what these effects are, it is necessary to exactly simulate the environment and reproduce these temperature gradients.

The most immediate use of our present facilities will be to obtain experimental information on control system components, turbopump bearing materials and components, and on the effect of radiation heating on our ability to store and pump liquid hydrogen in a nuclear rocket. These three items have been chosen both because of their importance to the successful development of the nuclear rocket, and because the obvious complexity of the programs involved in obtaining valid data imply a fairly long lead time.

The first two of the experiments mentioned will be carried out at the Plumbrook Reactor Facility. Components will not be included in radiation effects testing until they have successfully operated under a simulated environment which includes everything except the radiation field. The relative difficulty and expense connected with radiation effects testing makes it imperative to use these other tests as a screening procedure. A control actuator test, for example, involves one hour of irradiation and 47 hours of pre-and post-irradiation checkout examination. The Plumbrook loops will include the availability of two refrigeration systems, one of 20 KW and the second of 1 KW. They are to be capable of operation at temperatures to 30^{o}R and have helium flow of 1/2 pound per second at 100 psi. This slide (Figure IV-11) shows the experimental test equipment used for control components testing. A pneumatic control actuator is shown. Since the space vacuum is one of the environmental factors which may have a detrimental effect on the operation of control actuators, the inlet end of the test equipment contains a vacuum pump and is sealed off from the external environment. As you can see, there are provisions for applying both frictional and inertial loads. The dose rate to the actuator can be varied by moving the capsule towards or away from the centerline of the reactor. The flux gradient can be varied by remotely rotating the experiment in a sixty degree increment. This experiment has been designed and is awaiting the installation of some auxiliary equipment.

A second component which is receiving attention is the turbopump bearing. The only materials which have proven satisfactory as bearing materials for service in liquid hydrogen are glass laminates of teflon, which is well known for its rapid deterioration under radiation. Fortunately, our tests on these materials when immersed in liquid hydrogen indicate that teflon is able to withstand much higher doses when oxygen is excluded and the material is at low temperature. The exact geometry and the amount of frictional heat developed are expected to be critical to obtaining high performance. Candidate materials and configuration which have passed bench tests run at Lewis Research Center in rigs such as the one shown in Figure IV-12 will be irradiated while being operated in a bearing test rig capable of imposing axial and radial loads equivalent to that experienced during turbopump operation. Bearing speed, temperatures and torque are measured during operation.

A knowledge of the thermodynamic state of a liquid is indispensable to predicting the performance of a turbopump, or the best conditions for tank storage. In order to obtain some physical insight into the flow phenomena involved in nuclear heating of a liquid hydrogen tank, an experiment was devised which used infra red radiation absorbed in a tank of tricloroethane and ethyl alchohol to simulate nuclear heating. The centerline heating profile can be altered by changing the relative amount of the two fluids, which altered the spectral absorbtion of the liquid. Schlieren photographs (Figure IV-13) were used to obtain a qualitative understanding of the induced fluid motion.

As you can see, the fluid motion is affected by the relative amount of energy deposited in the fluid and in the tank walls. The wall heating produces a stratified layer of warm fluid which resists participation in the convective turbulent flow caused by the attenuation of infra red by the fluids. These photographs gave enough understanding of the flow regimes to establish plausible temperature profiles in the tank and to derive the form of equations which would satisfy momentum and energy considerations.

Because of the complex heat transfer mechanisms involved, and the thermal flow effects induced, we are using experimental simulation techniques to improve our ability to predict within limits the effect of radiation of tank heating and pump performance. Our effort in this area involves four steps. The first of these is the determination of temperature distribution and energy deposition in liquid hydrogen filled tanks under irradiation. This is being carried out at the ASTR reactor at Fort Worth, Texas. (Figure IV-14) This reactor is an ideal reactor for work with liquid hydrogen, since it is open to the atmosphere and is easily portable. The reactor itself can be turned on its side and immersed in water as shown in

Figure IV-15, so that the experimental tank can be placed immediately next to the core where the highest dose rate is available. This also assures against nuclear accidents being caused by a hydrogen spill into the core. These experiments are carried out in a 125 gallon LH_2 tank with the approximate tank bottom configuration expected in a vehicle. One of the more important parameters influencing convective flow in these tanks will be the relative heating from tank wall and bottom and that from gamma and neutron heating. An electrically heated duplicate of this tank is at the Lewis Research Center where wire resistors inside the tank may be varied to change the spatial distribution of energy. The results of these experiments will be used to create analytical methods for predicting the thermal history of a fluid element in a tank heated by irradiation, and to demonstrate the validity of using electrical heating as a simulation technique for rocket tank heating experiments. If this is demonstrated, 10,000 gallon tanks will be used in the electrically heated experiments to determine the validity of scaling laws. The demonstration that analytical procedures are available to predict thermally induced flows in an electrically heated experiment following demonstration that electrically heated experiments can be devised to accurately simulate radiation heating will provide us with a very powerful tool in engine and rocket stage analysis.

The last step will be to determine the operating performance of our pumps using liquid hydrogen having varying thermal histories. This will be carried out using electrically heated walls in the inlet to operating turbopumps during dynamic simulation testing in our dynamic test stands.

Since most of the materials used external to the reactor core are required to function at cryogenic temperatures, the effect of radiation on the engineering properties of materials must be determined. The low temperatures involved could reduce ductility, for example, by locking in defects produced by the radiation, producing an effect from the combination of low temperature and radiation possibly more severe than a simple addition of two separate effects. On this slide is shown a cryogenic materials test loop installed at the Plumbrook Reactor Facility. This loop has the capability of 1,000 watts cooling capacity at 30°R. It is operated on a helium coolant cycle. The loop is capable of making tensile, creep, or sheer stress tests on miniature specimens. These miniature specimens shown in the center photo of Figure IV-16 are correlated with standard size test specimens under the same conditions but without the effect of radiation. It is most important these tests can be conducted dynamically or while the materials specimen is under irradiation and without removing it from the cryogenic environment. The danger that any defects produced in the crystal structure will be annealed in transfer to a test bench is therefore avoided. We believe that this is necessary. In this loop, we will be able to obtain the material properties for the materials of interest under completely simulated environmental conditions. These data will then be used in the design of the Phoebus, NERVA, and vehicle systems. In order to illustrate the importance of long lead time, the planning and initial work on this facility was started in December 1959. It required almost 3-1/2 years to design, develop, fabricate, install and check out this complex facility.

Dynamic Engine Simulation

Understanding of the dynamic transient characteristics of the engine, and some of the more important subsystems, such as the propellant feed system, is important not only to the success of our present NERVA engine, but to our ability to provide improved engine systems in the future. In this effort, extensive use is made of the two dynamic engine simulation stands, shown in Figure IV-17. The one on the right is the B3 stand which will be completed this year. On the left is the B1 stand which has been in operation since late in 1963. All the components of the engine and the liquid hydrogen tank are close coupled in the configuration in which they will be used. These experiments use a cold flow model of the KIWI reactor, seen in place in Figure IV-18.

These tests provide accurate simulation of the start-up transient in nuclear rockets even though no power source is used. The heat capacity of the core provides accurate simulation of the critical period of start-up. Analog simulation studies indicate that start characteristics of the NERVA engine are accurately simulated over the first 50 seconds from commitment of propellant flow. The heat capacity of the core also serves to provide the energy needed for bootstrap start of the turbine.

The use of a cold flow reactor in experiments like this one permits extensive instrumentation of the core, including the use of motion pictures and television, which would not be possible in the high radiation fields of an engine test. Steam ejectors enable this stand to start at about 1 psia. This is the only facility in the nuclear rocket program which will have that capability for some time. The effect of vacuum on flow and other conditions during start can therefore be obtained from this facility.

This stand also permits experimental evaluation of the effect of changing engine configuration and components on the dynamic characteristics of the engine. For example, the effect of a boost pump or change in inducer design on NPSP can be determined. A wide range of start-up periods, bootstrap programs, and turbine exhaust conditions can also be evaluated.

V. CONCLUDING REMARKS

We have in some detail defined the program that is now being pursued in the United States to provide nuclear rockets for space exploration and we have described the mission applications of these rockets. It is clear, with the accomplishments already achieved, that a new area of rocketry is being developed and is near at hand. We now understand these systems well enough that missions depending on their use can be planned with reasonable assurance that the estimated development programs, time scales, fund requirements, and, most important, required performance levels can be achieved. The progress that has been made and the achievements demonstrated during this year justify the effort that has been devoted to this important area. The space exploration capability these nuclear rocket systems will provide should be a source of substantial benefit to all mankind.

LIST OF ILLUSTRATIONS

IV. ADVANCED RESEARCH AND TECHNOLOGY

SOME SPACE EXPLORATION POSSIBILITIES

MISSIONS	REGION: EARTH ORBIT	REGION: LUNAR	REGION: PLANETARY
UNMANNED	1 **UNMANNED SATELLITES** SCIENTIFIC *EXPLORERS *ORBITING OBSERVATORIES APPLICATION *COMMUNICATION *METEOROLOGY *NAVIGATION *ENGINEERING RESEARCH	2 **LUNAR PROBES** *RANGER *SURVEYOR INTERMEDIATE SPACE PROBES *PIONEER	3 **DEEP SPACE PROBES** *MARINER *INTERPLANETARY MONITOR SATELLITE VOYAGER SOLAR PROBE OUT OF ECLIPTIC OUTER PLANETS AND THEIR SATELLITES LEAVE SOLAR SYSTEM SEARCH FOR EXTRATERRESTRIAL LIFE
MANNED DEVELOPMENTAL	4 **MANNED SATELLITES** *MERCURY *GEMINI INTERIM ORBITAL LABS MANEUVERING REENTRY	5 **BEFORE 1970 MANNED LANDING** *APOLLO LUNAR LOGISTIC SYSTEM (UNMANNED)	6 **MANNED EXPEDITIONS** MARS LANDING VENUS RECONNAISSANCE SEARCH FOR LIFE ON PLANETS No Authorized Programs Yet
MANNED OPERATIONAL	7 **ORBITAL OPERATIONS** MANNED ORBITING LABS OPERATIONAL FERRY VEHICLE RECOVERABLE BOOSTERS ENGINEERING EXPERIMENT AND DEVELOPMENT No Authorized Programs Yet	8 **LUNAR STATION** LUNAR EXPLORATIONS SCIENTIFIC OBSERVATIONS No Authorized Programs Yet	9 **PLANETARY OPERATIONS** MARS STATION ADVANCED MANNED EXPEDITIONS VENUS JUPITER SATELLITES MERCURY ASTEROIDS No Authorized Programs Yet

NASA AE63-1302

Figure II-1

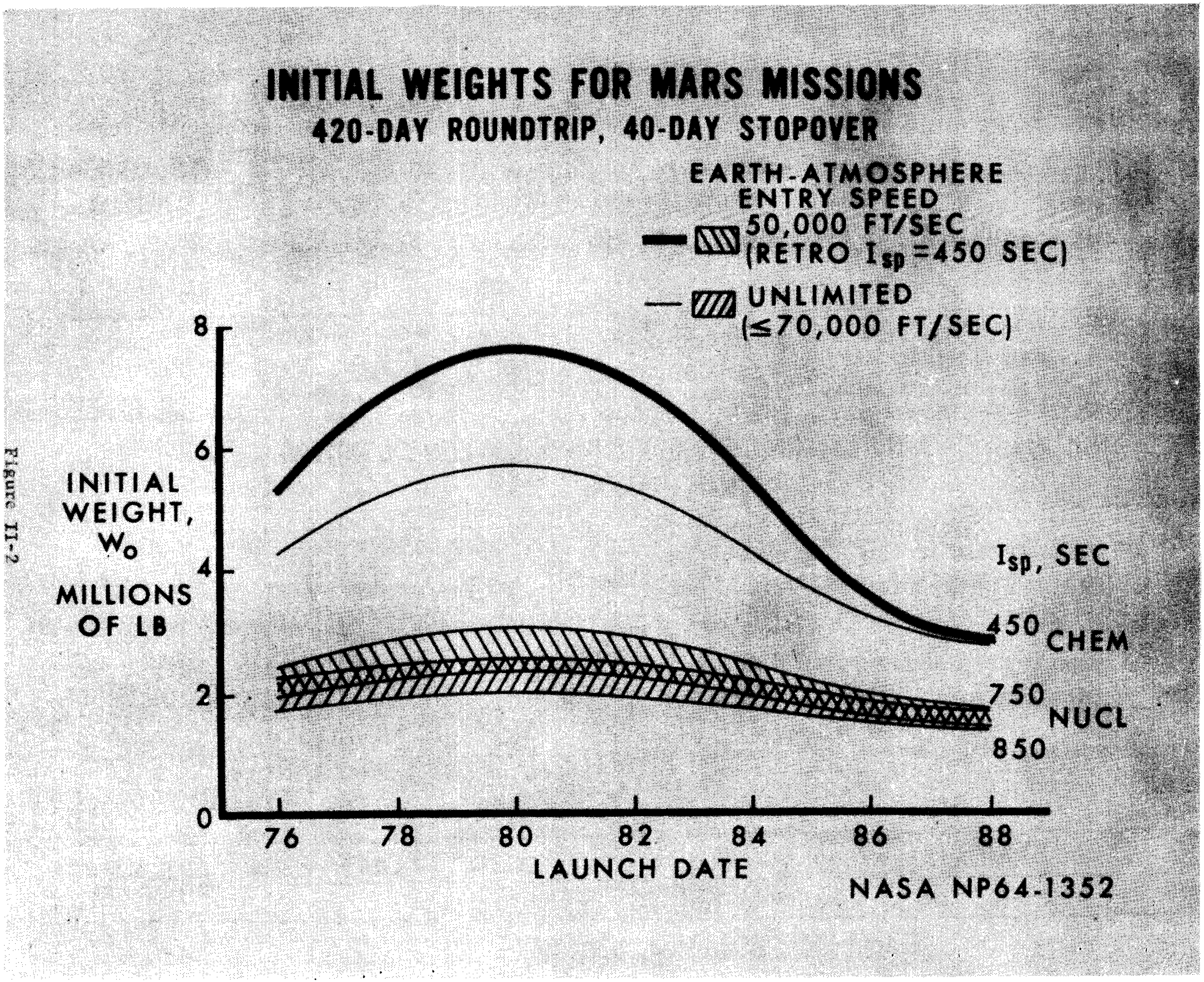
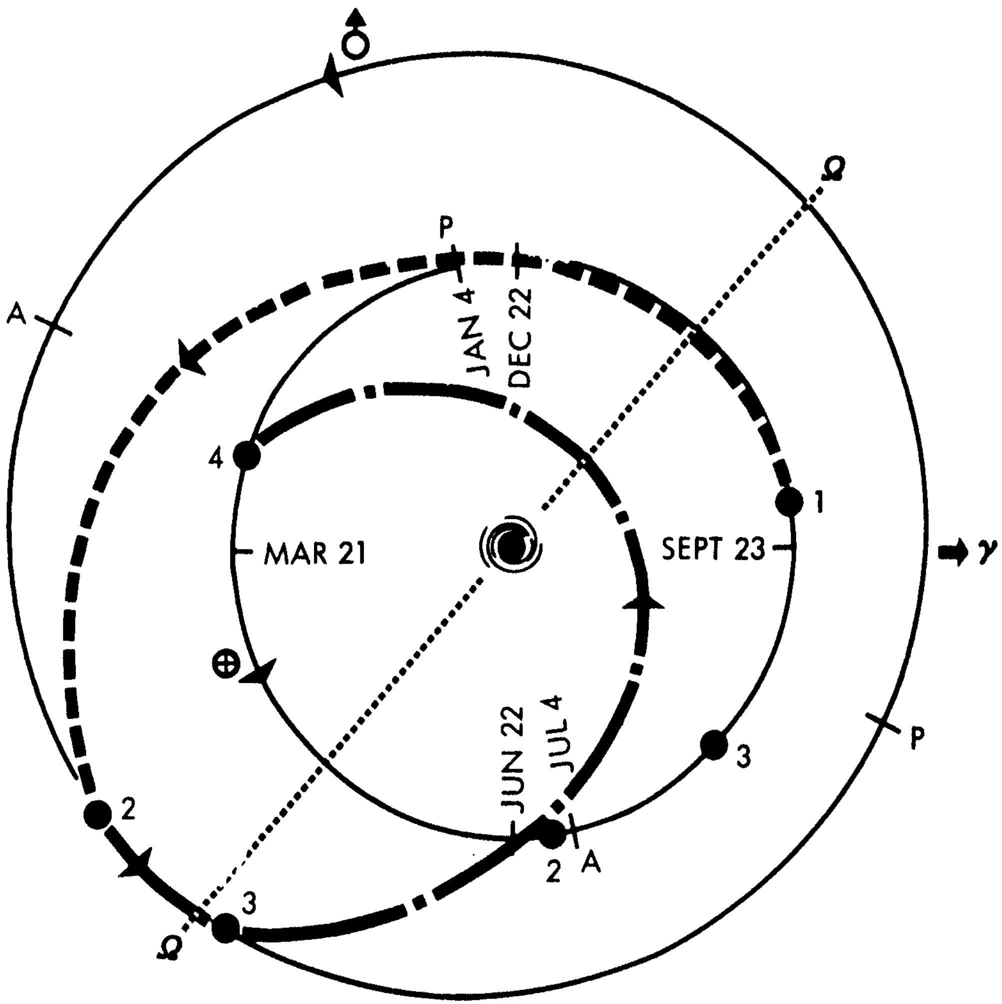

INITIAL WEIGHTS FOR MARS MISSIONS
420-DAY ROUNDTRIP, 40-DAY STOPOVER
EARTH-ATMOSPHERE ENTRY SPEED
50,000 FT/SEC (RETRO I_{sp} =450 SEC)
UNLIMITED (≤70,000 FT/SEC)
INITIAL WEIGHT, W_o MILLIONS OF LB
0
2
4
6
8
76
78
80
82
84
86
88
LAUNCH DATE
I_{sp}, SEC
450 CHEM
750 NUCL
850
NASA NP64-1352

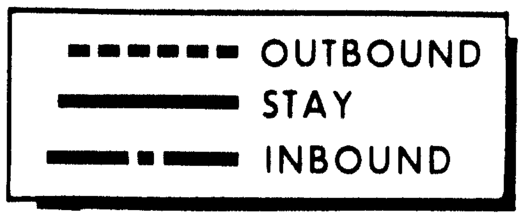
Figure II-2

MISSION GEOMETRY

1979 LAUNCH

OPPOSITION-CLASS TRIP
(~ 500 DAYS)





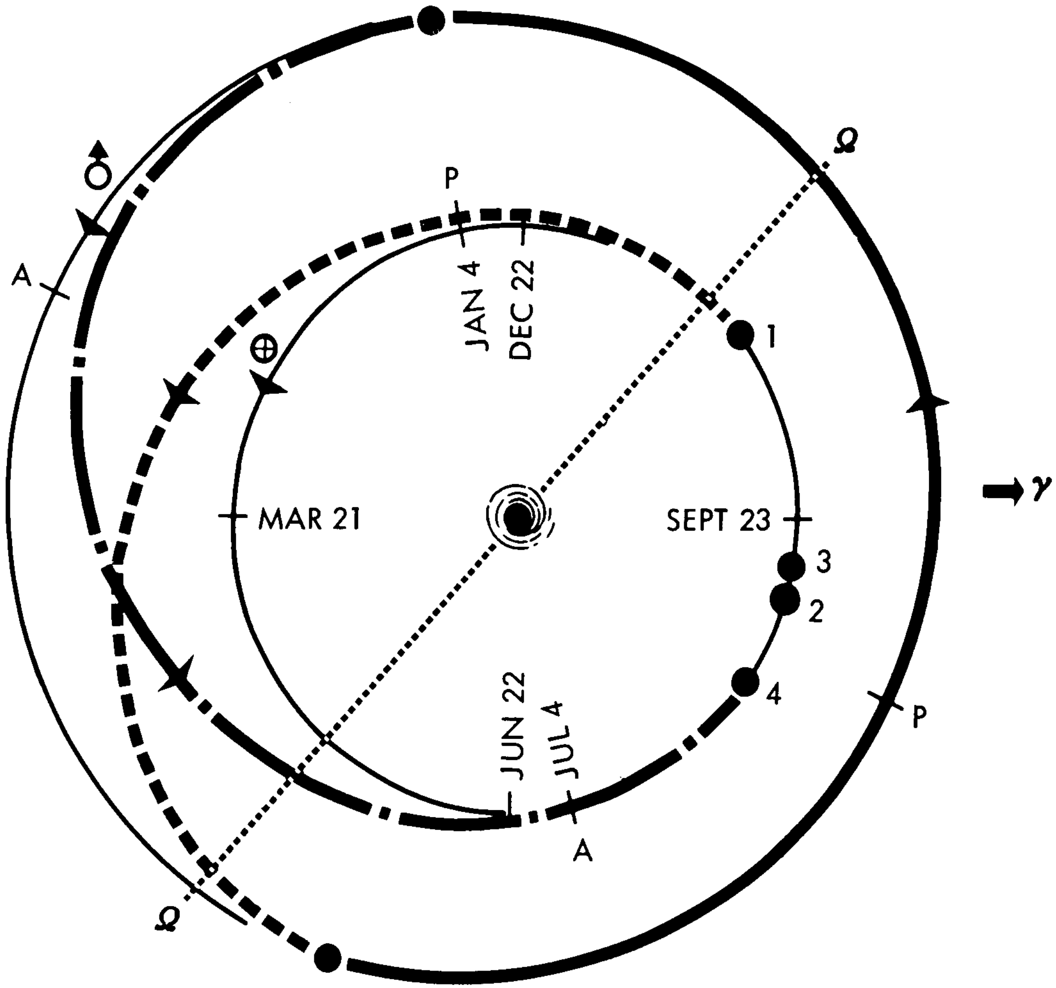
NASA NP64-1337
REV 9-4-64

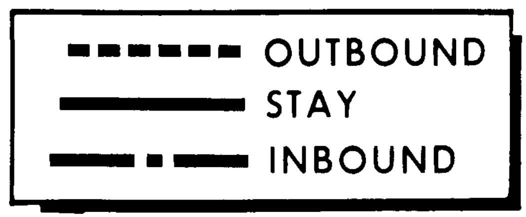
Figure II-3

MISSION GEOMETRY

1979 LAUNCH

CONJUNCTION - CLASS TRIP
(~ 1000 DAYS)





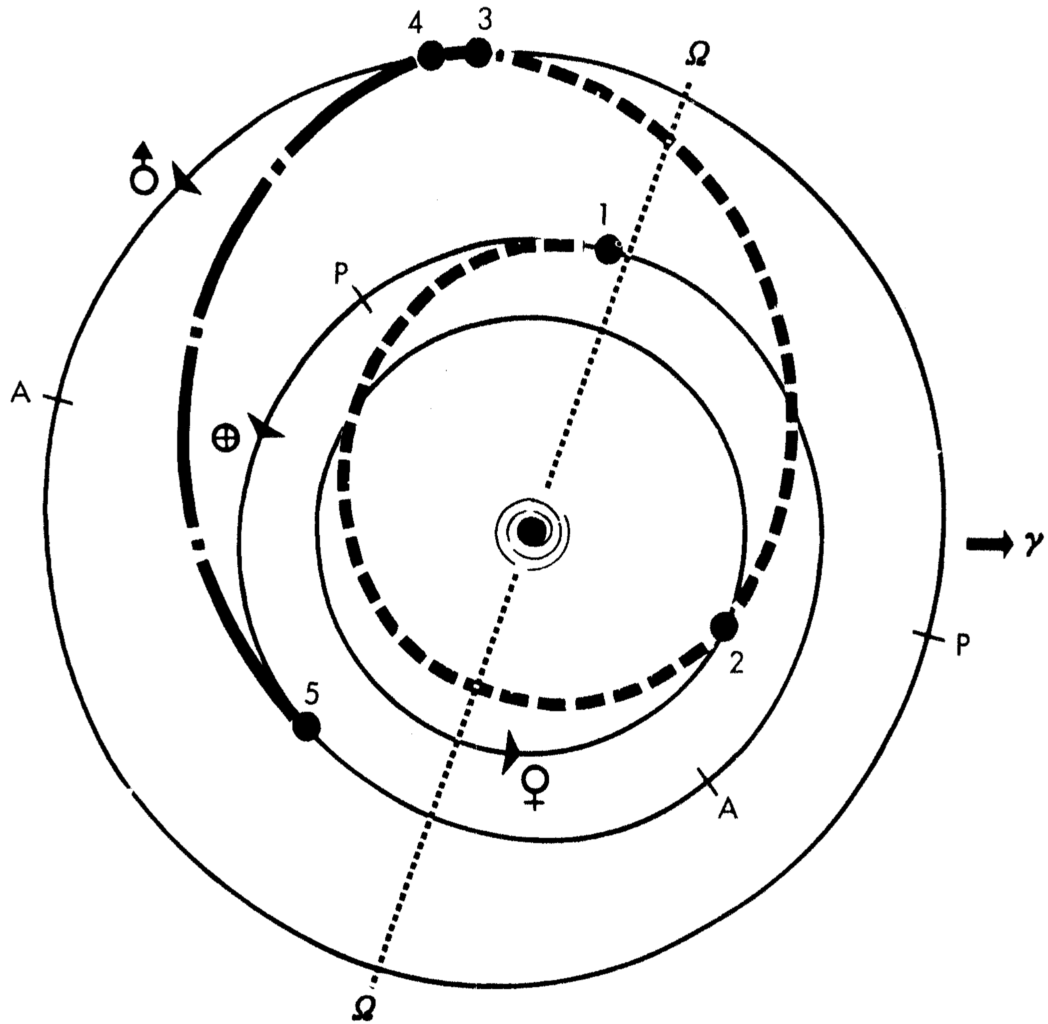
NASA NP64-1336
REV 9-4-64

Figure II-4

MISSION GEOMETRY

1979 LAUNCH

VENUS SWINGBY TRIP
(~ 500 DAYS)



OUTBOUND
STAY
INBOUND

NASA NP64-1737
REV 9-4-64

Figure II-5

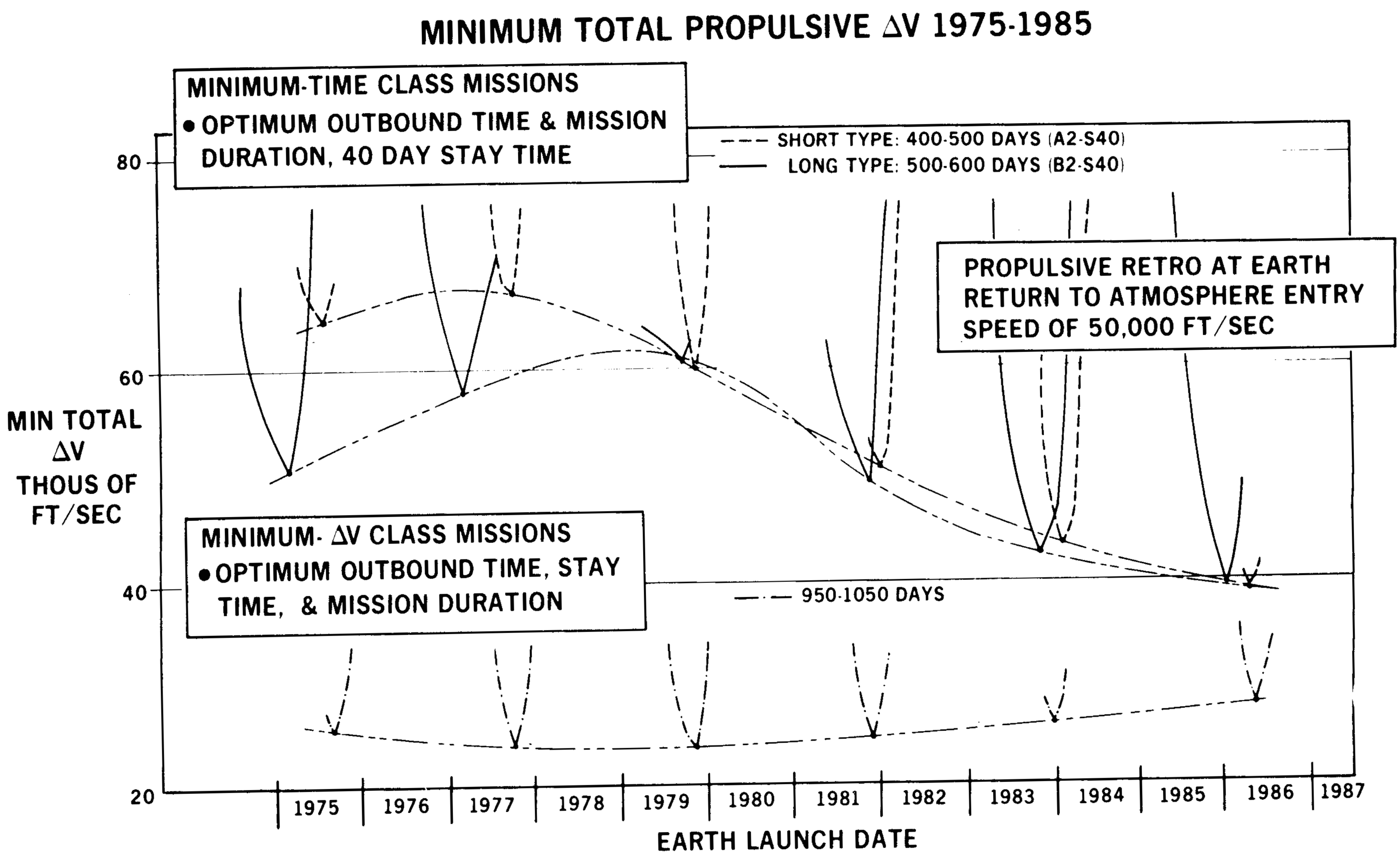


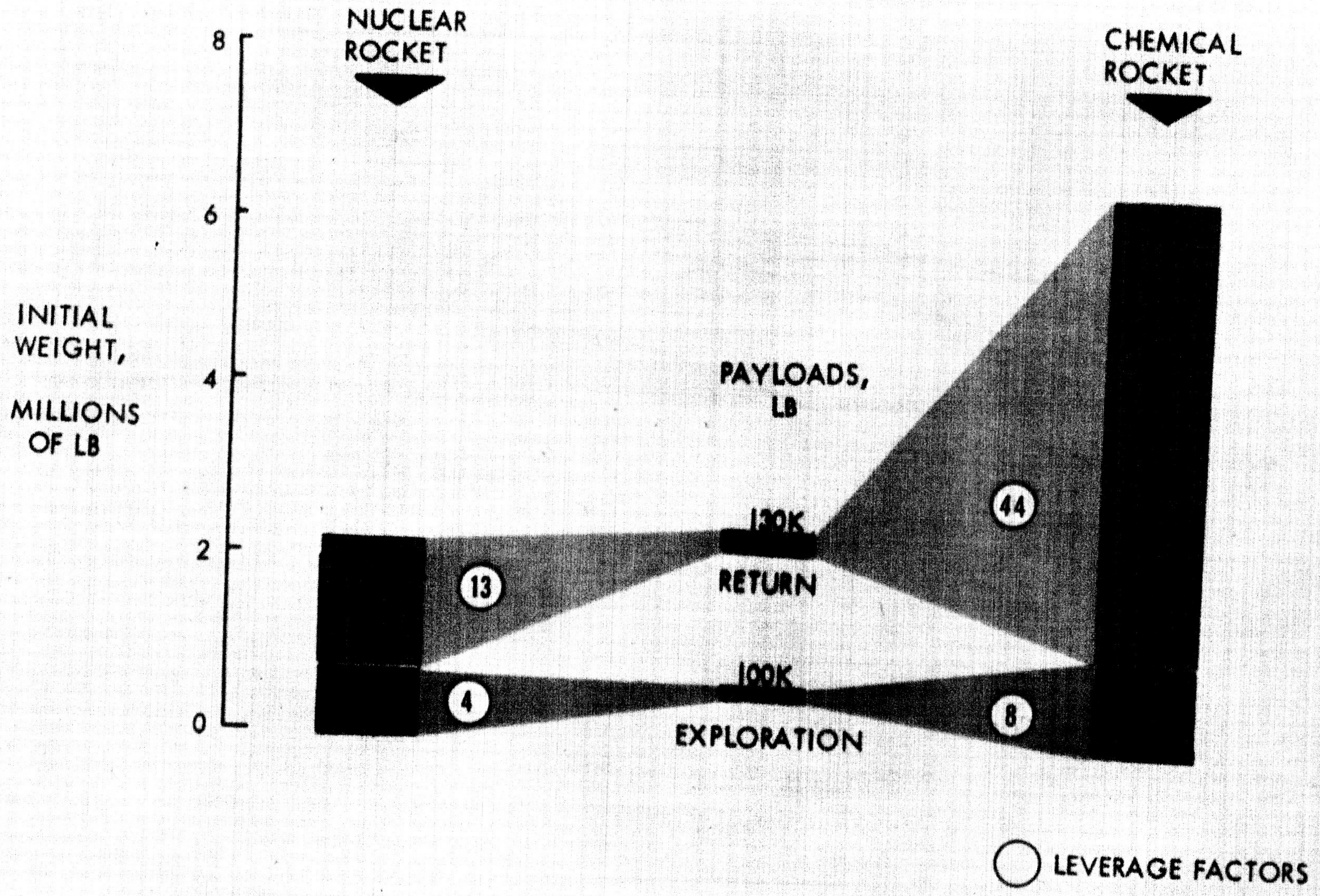
FIG. 6

NASA NP64-1740

Figure II-6

BUILD-UP TO INITIAL WEIGHT IN ORBIT

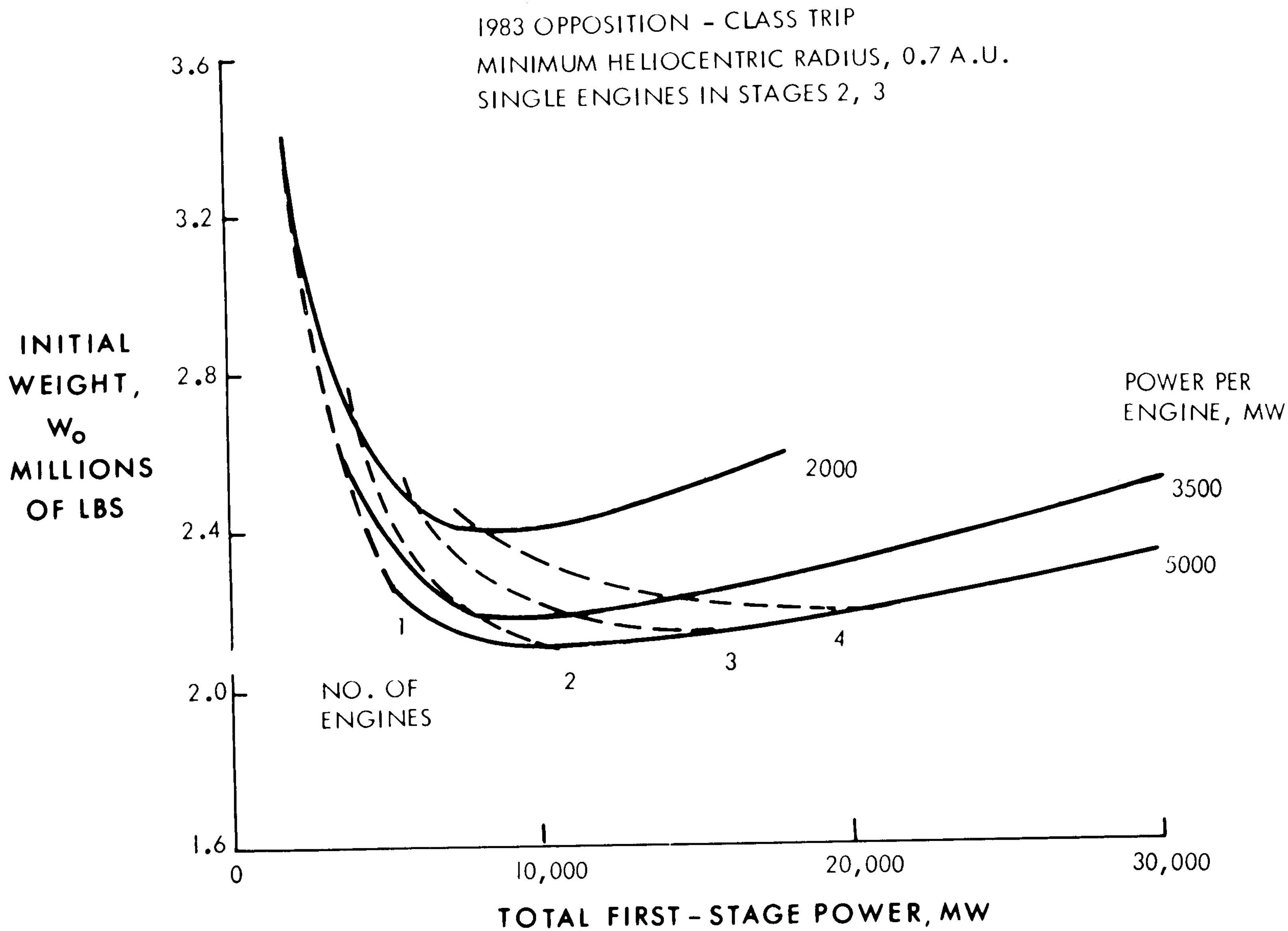
1979 Opposition—Class Mars Trip



NASA NP64-1334
9-2-64 Version-Color Changes only

Figure II-7

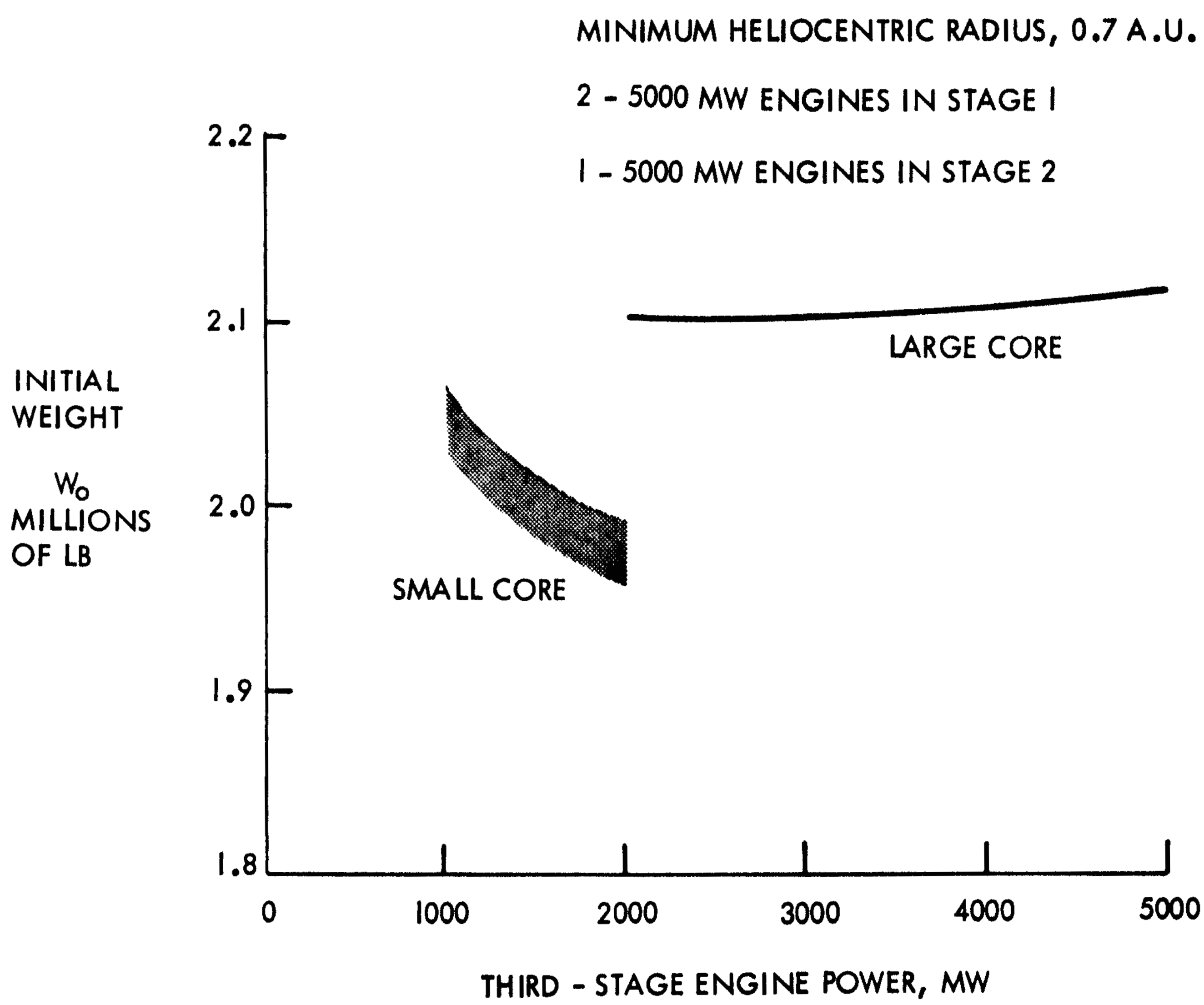
EFFECT OF EARTH-DEPARTURE ENGINE CLUSTERING



NASA NP64-1339

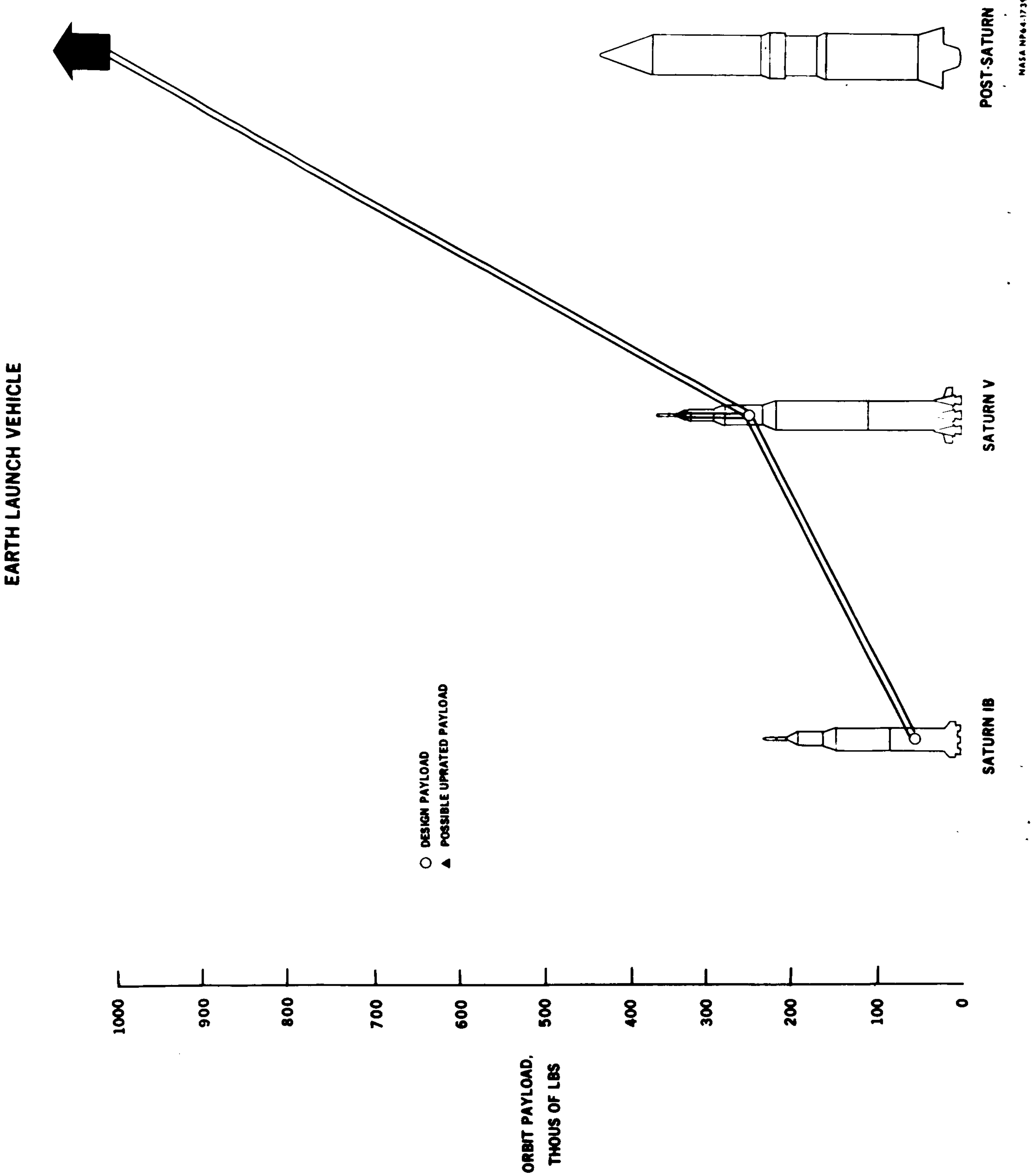
Figure II-8

EFFECT OF MARS-DEPARTURE ENGINE POWER
1983 Opposition—Class Trip



NASA NP64-1335

Figure II-9



EARTH LAUNCH VEHICLE
ORBIT PAYLOAD, THOUS OF LBS
1000
900
800
700
600
500
400
300
200
100
0
DESIGN PAYLOAD
POSSIBLE UPRATED PAYLOAD
SATURN IB
SATURN V
POST-SATURN

Figure II-10

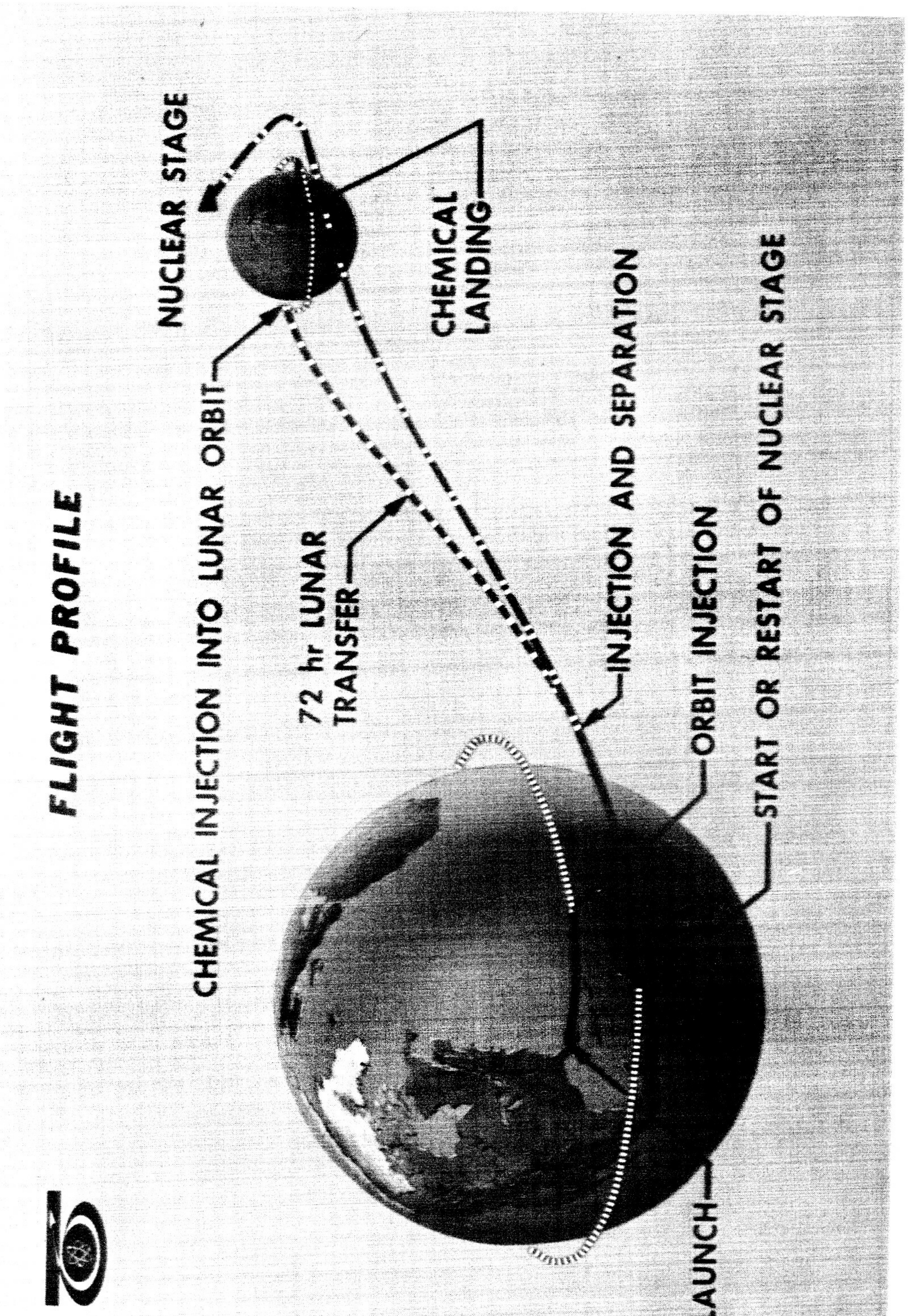

FLIGHT PROFILE
CHEMICAL INJECTION INTO LUNAR ORBIT
NUCLEAR STAGE
72 hr LUNAR TRANSFER
CHEMICAL LANDING
INJECTION AND SEPARATION
ORBIT INJECTION
START OR RESTART OF NUCLEAR STAGE
LAUNCH

Figure II-11

LUNAR LANDED PAYLOADS

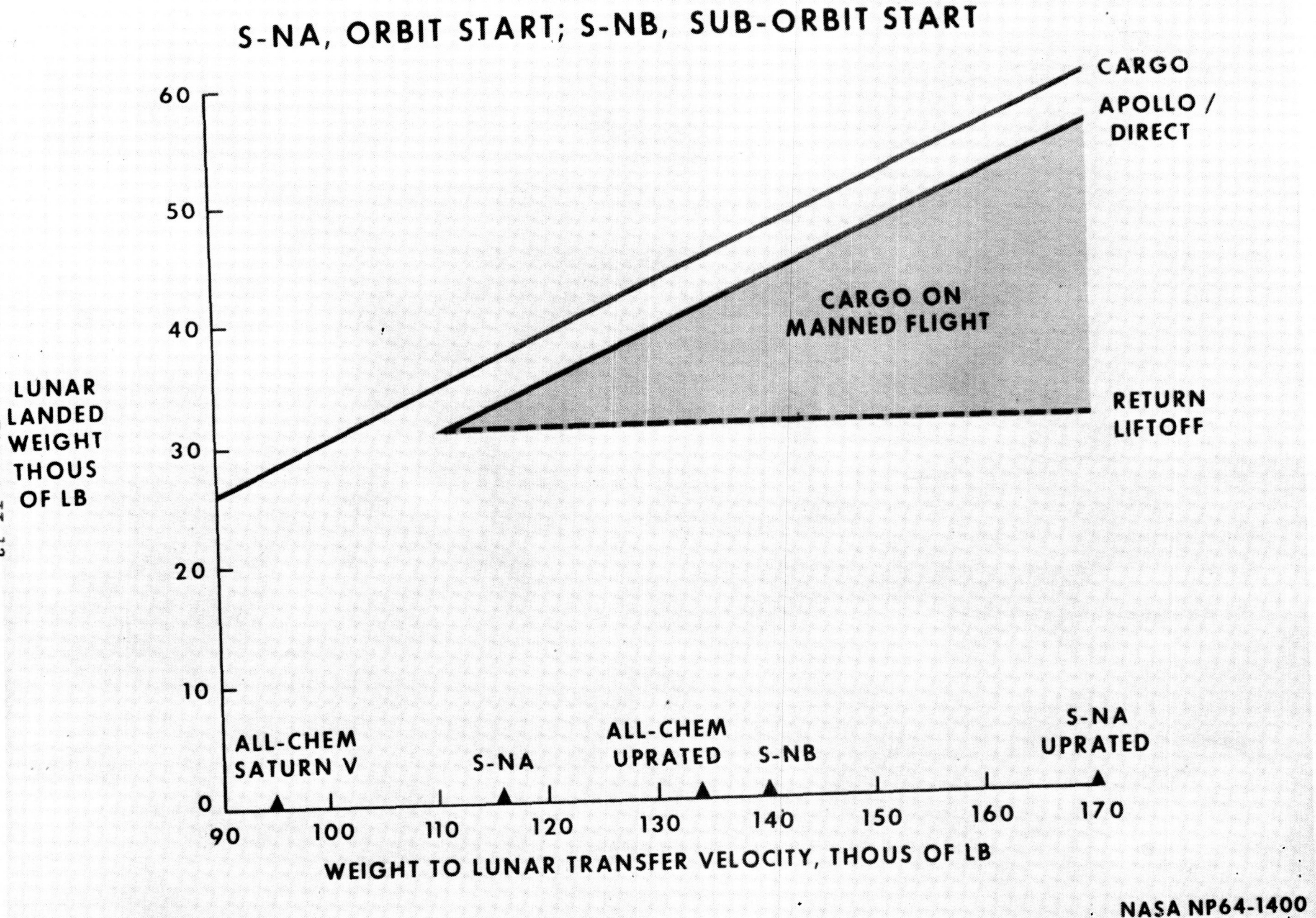
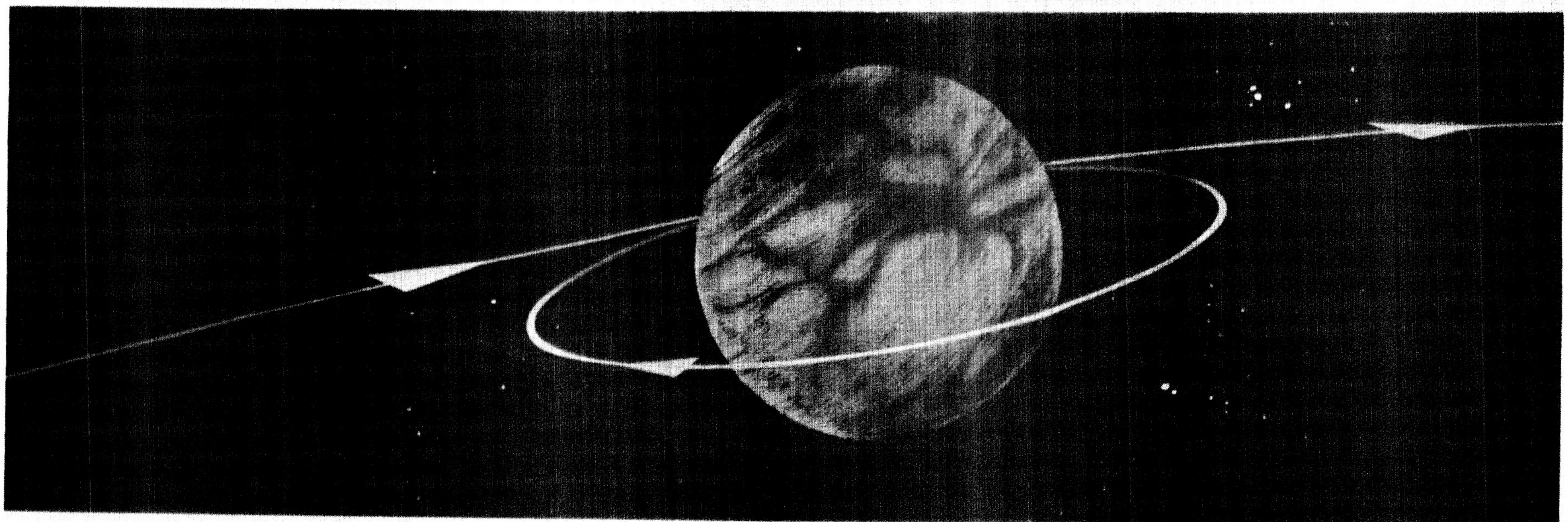


Figure II-12

SUMMARY OF MARS CAPABILITY 1973 LAUNCH



FLYBY WEIGHT (LB)

S-NA	60,000
S-NB	74,000
S-IV B	40,000

ORBIT WEIGHT (LB)

	MMM	SM	S-N
S-NA	45,000	32,000	48,000
S-NB	53,000	38,000	55,000
S-IV B	31,000	21,000	——

Figure II-13

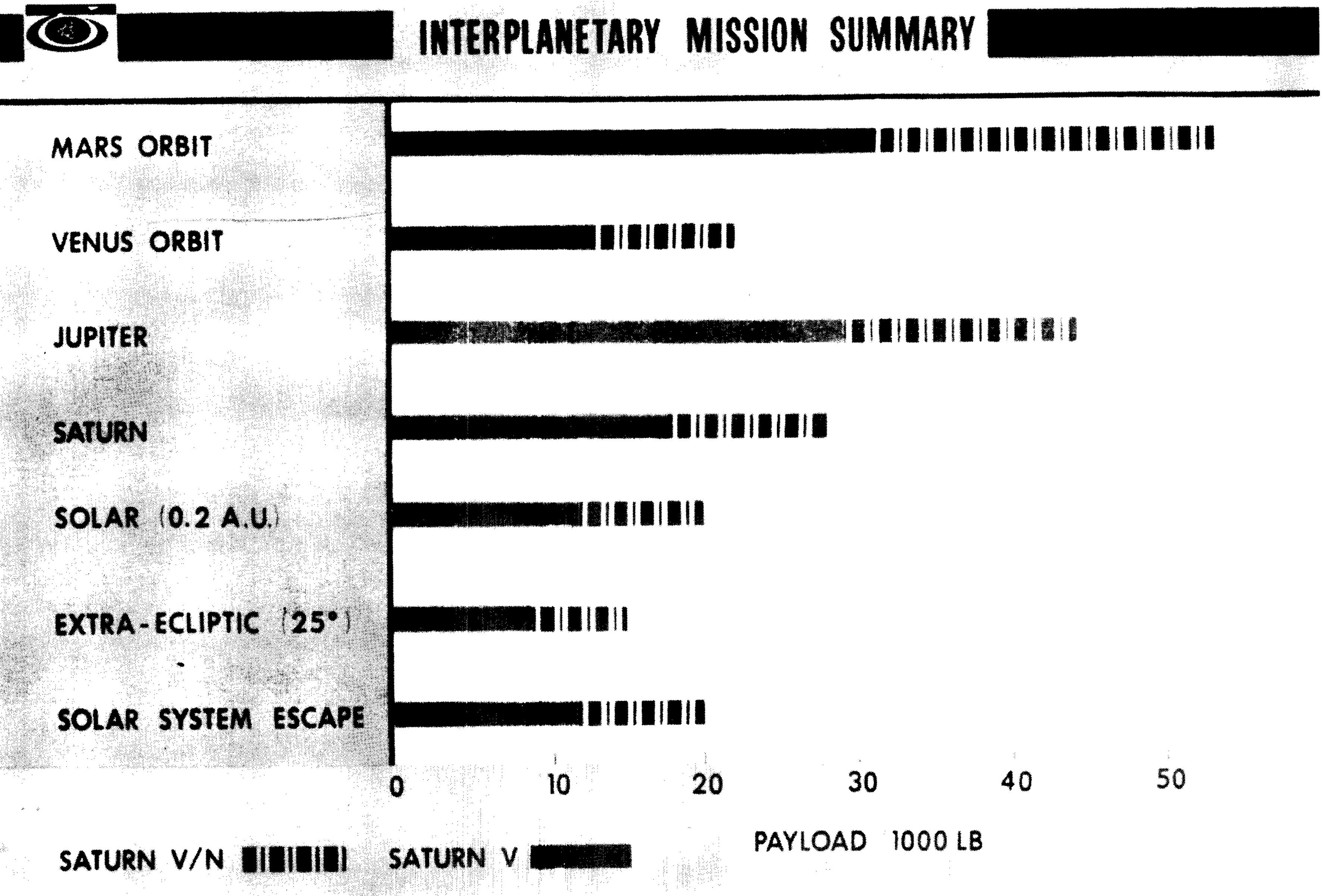

INTERPLANETARY MISSION SUMMARY
MARS ORBIT
VENUS ORBIT
JUPITER
SATURN
SOLAR (0.2 A.U.)
EXTRA-ECLIPTIC (25°)
SOLAR SYSTEM ESCAPE
0
10
20
30
40
50
PAYLOAD 1000 LB
SATURN V/N
SATURN V

Figure II-14

PERFORMANCE OF ADVANCED HIGH-THRUST PROPULSION SYSTEMS

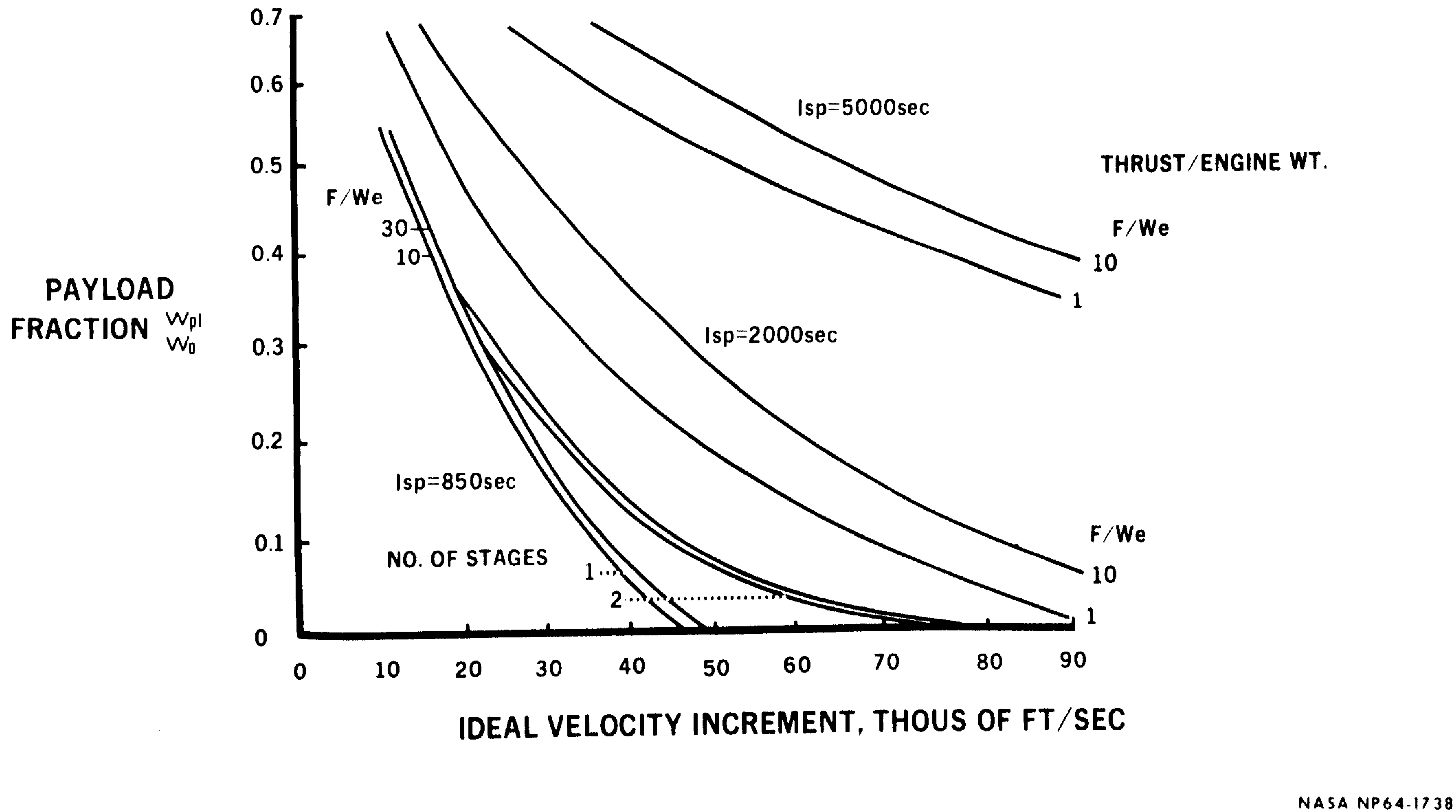
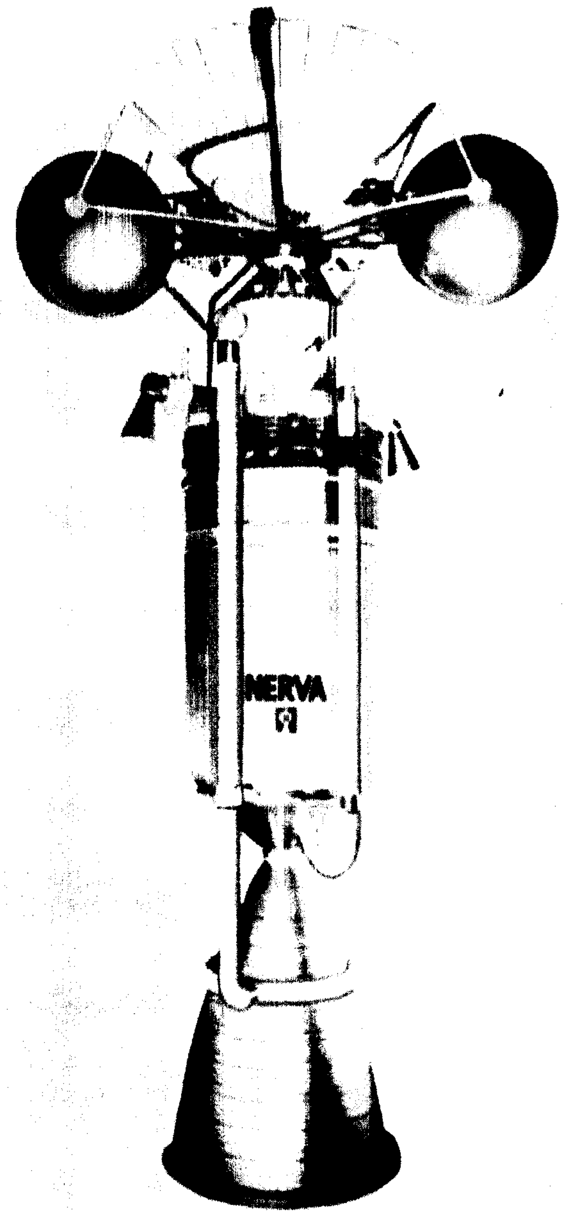


Figure II-15

NERVA
MOCKUP



NASA R63-1094

Figure III-1

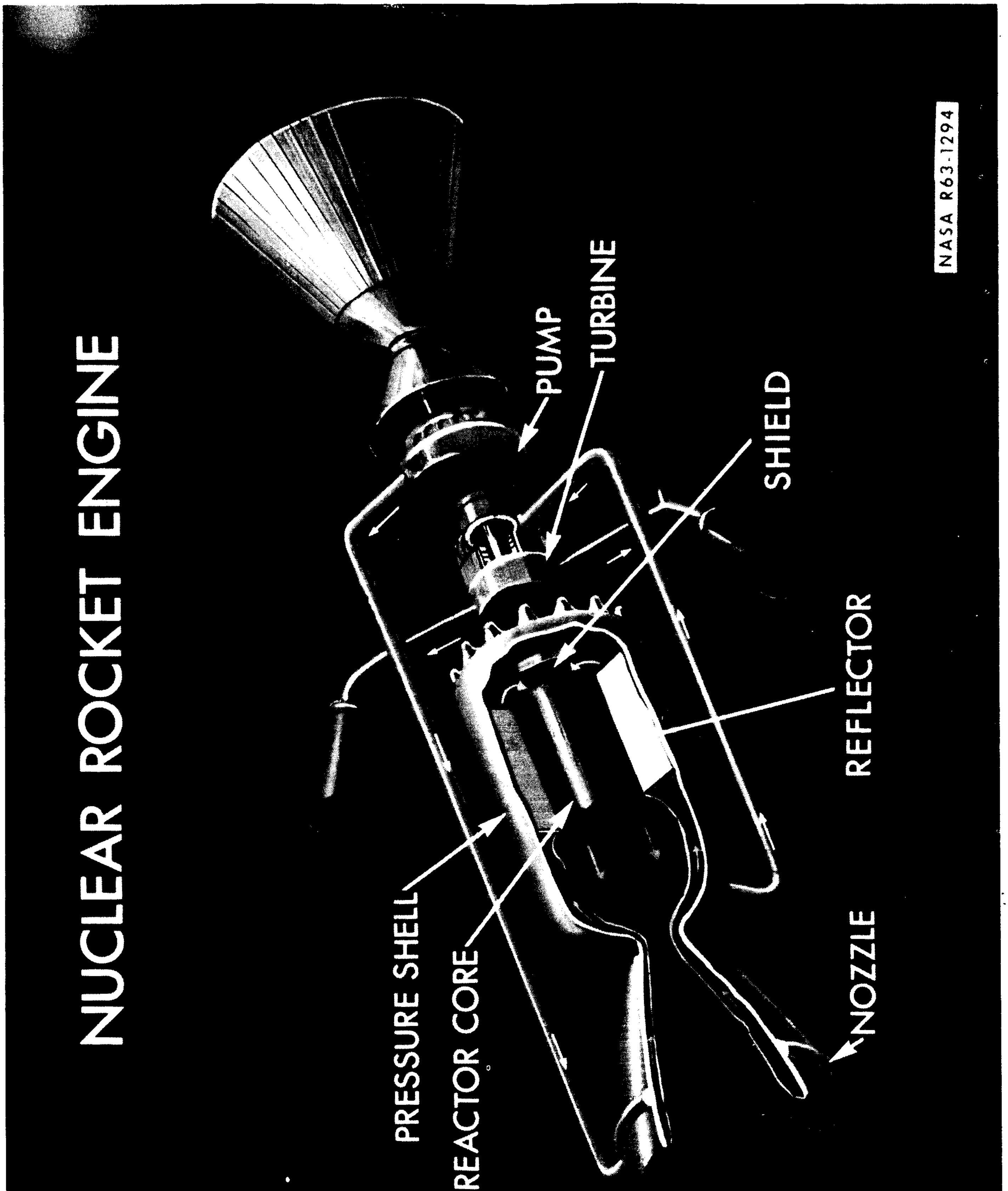
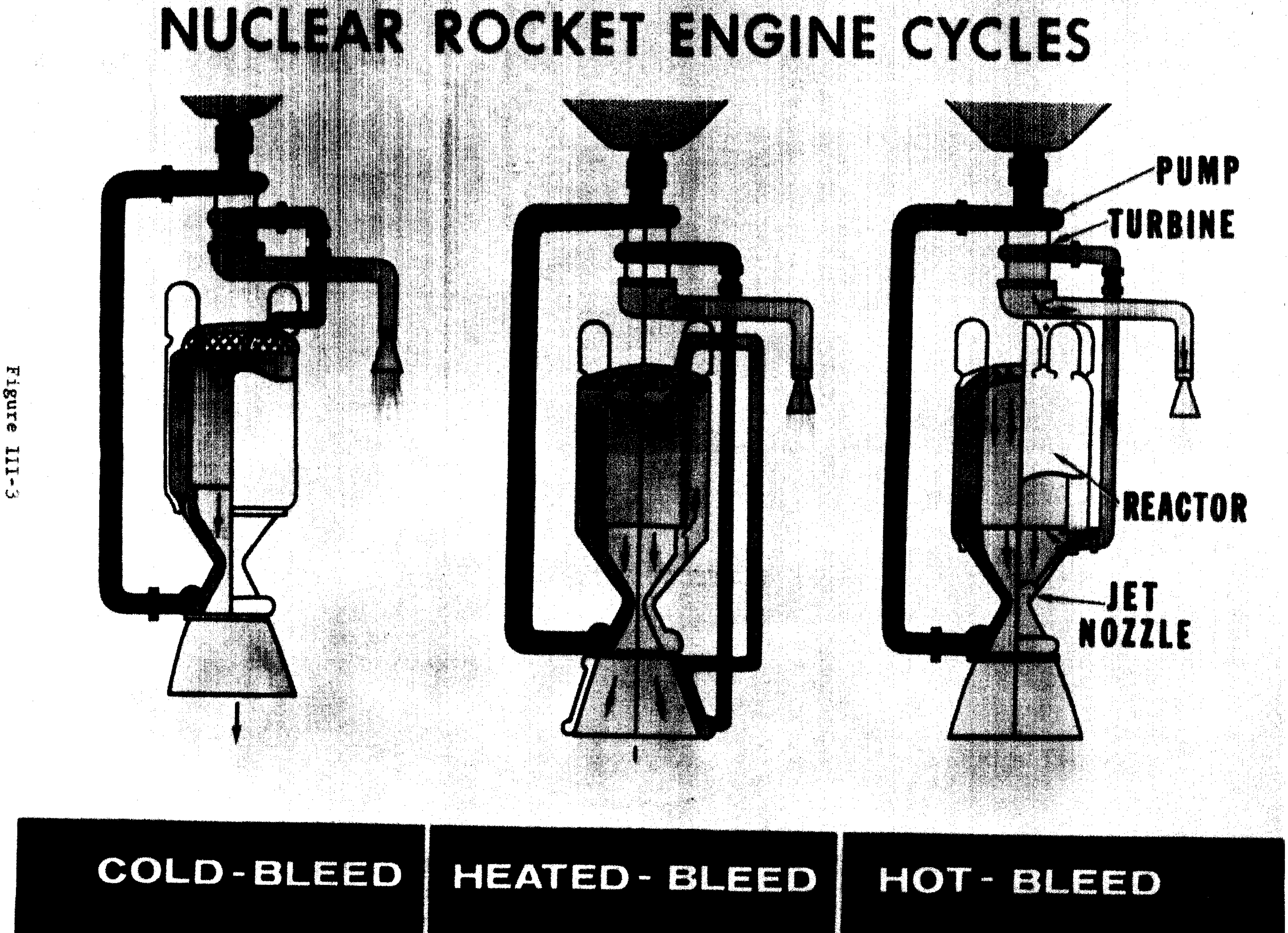

NUCLEAR ROCKET ENGINE
PUMP
TURBINE
SHIELD
REFLECTOR
PRESSURE SHELL
REACTOR CORE
NOZZLE
NASA R63-1294

Figure III-2



NUCLEAR ROCKET ENGINE CYCLES
PUMP
TURBINE
REACTOR
JET
NOZZLE
COLD-BLEED
HEATED-BLEED
HOT-BLEED

Figure III-3

REACTOR ASSEMBLY NRX-A

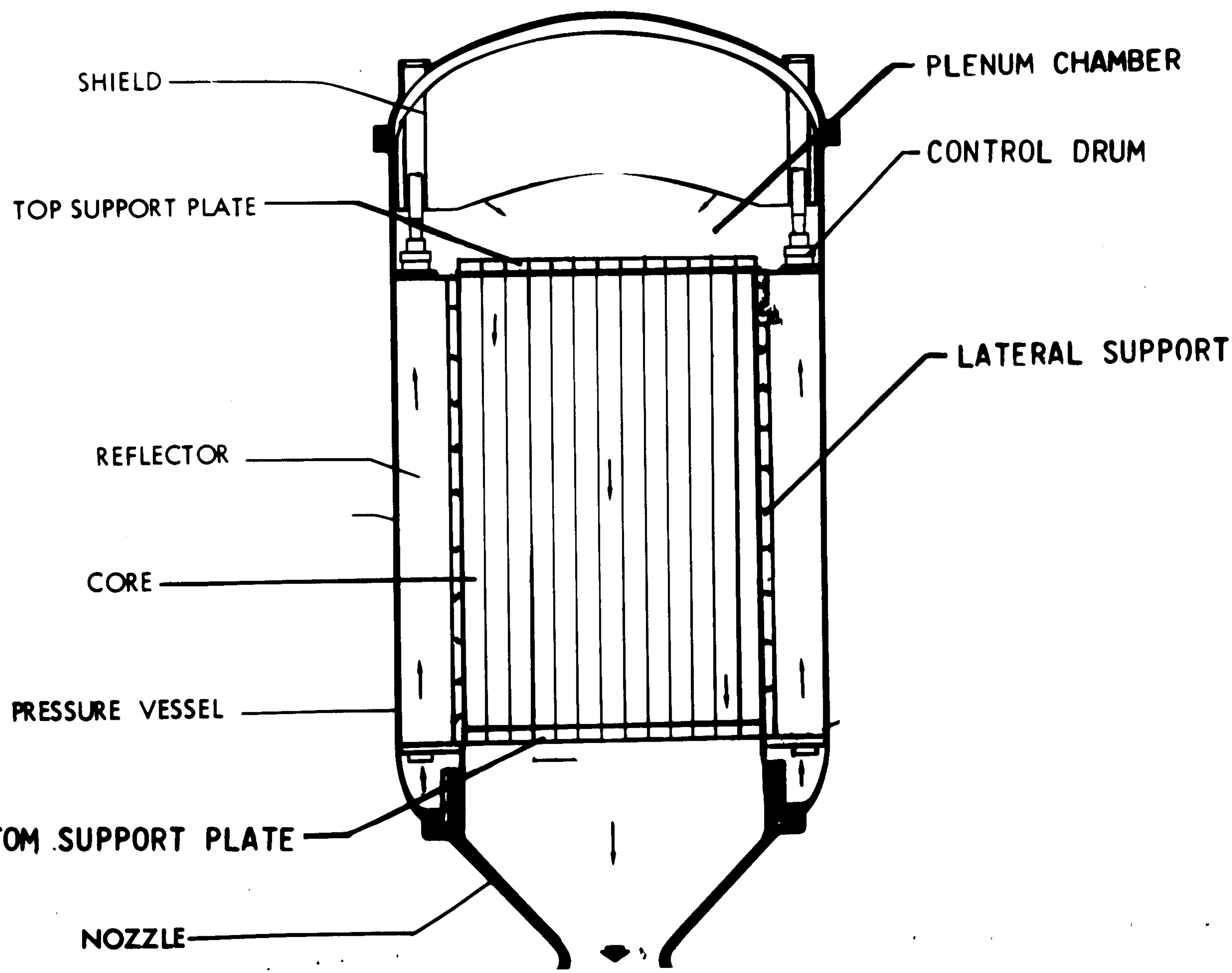


Figure III-4

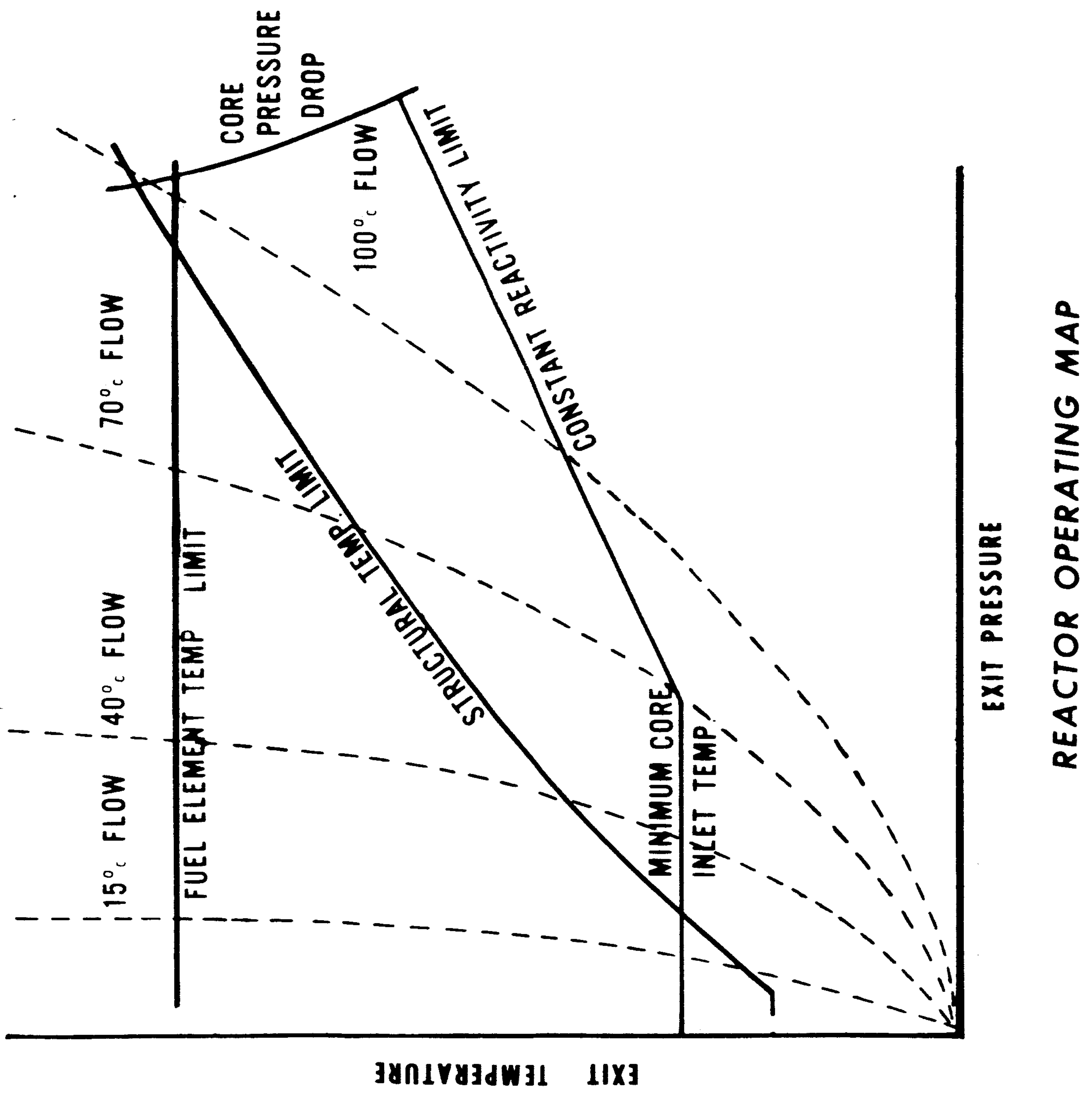
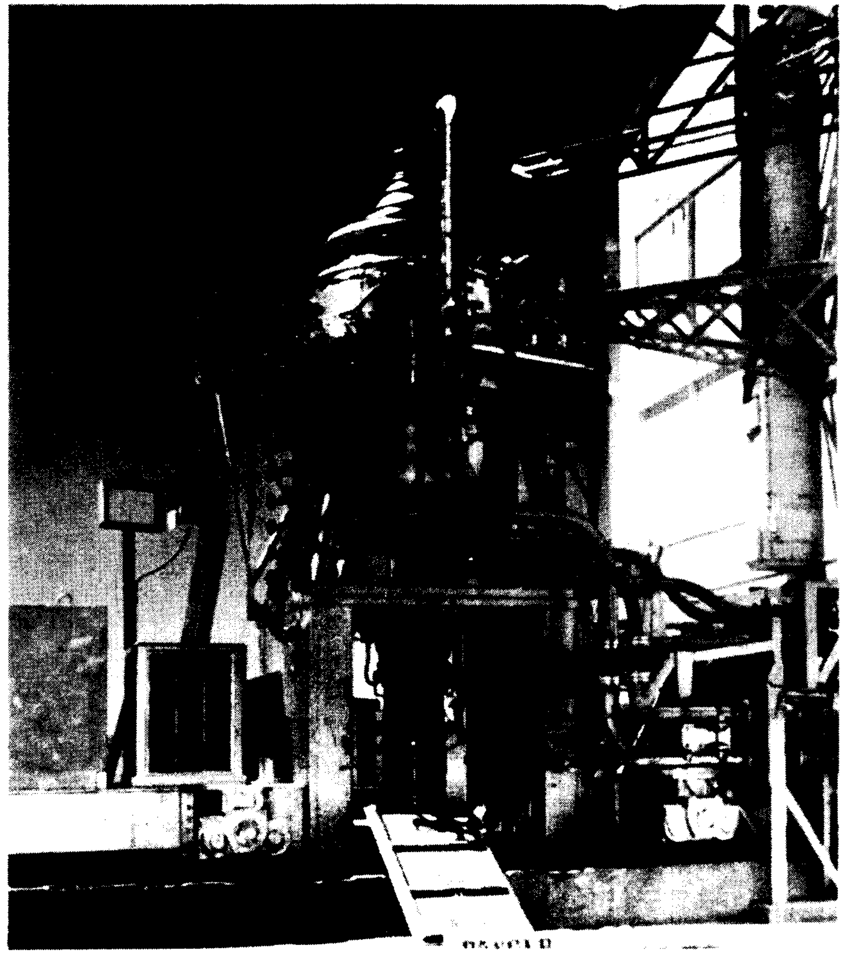

REACTOR OPERATING MAP
CORE PRESSURE DROP
100°C FLOW
70°C FLOW
40°C FLOW
15°C FLOW
CONSTANT REACTIVITY LIMIT
STRUCTURAL TEMP. LIMIT
FUEL ELEMENT TEMP LIMIT
MINIMUM CORE INLET TEMP.
EXIT PRESSURE
EXIT TEMPERATURE

Figure III-5

PROGRESS IN KIWI-A REACTORS

1. DESIGN METHODS
2. MATERIAL DATA
3. CONTROL INFORMATION
4. NUCLEAR CHARACTERISTICS

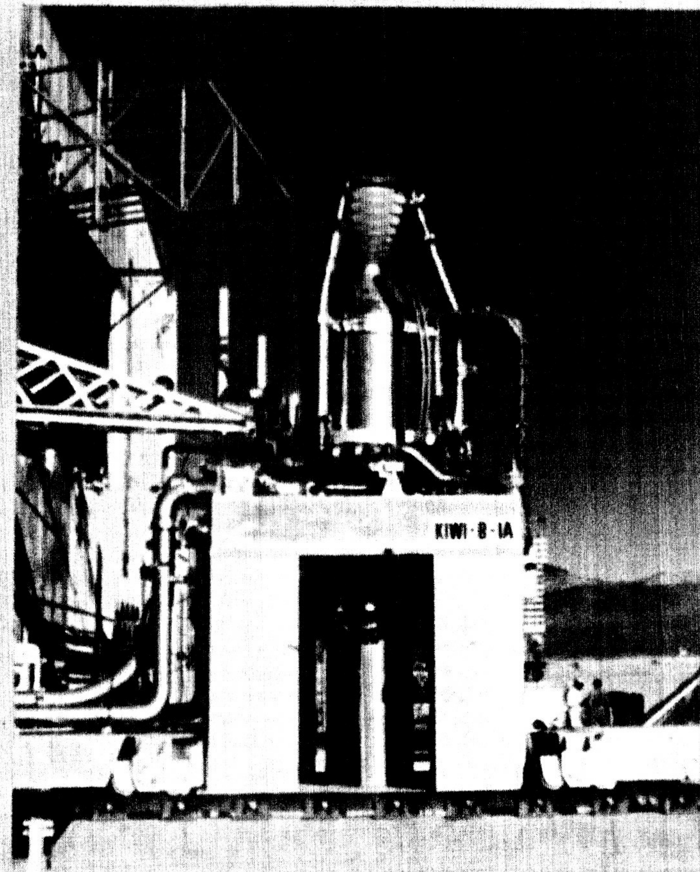


NASA NPO 64-468

Figure III-6

PROGRESS IN KIWI-B-1 REACTORS

1. REFLECTOR CONTROL
2. LIQUID H_2 OPERATION
3. LIQUID H_2 COOLED NOZZLE
4. FUEL-ELEMENT FABRICATION
5. AUTOMATIC CONTROL



NASA NPO 64-467

Figure III-7

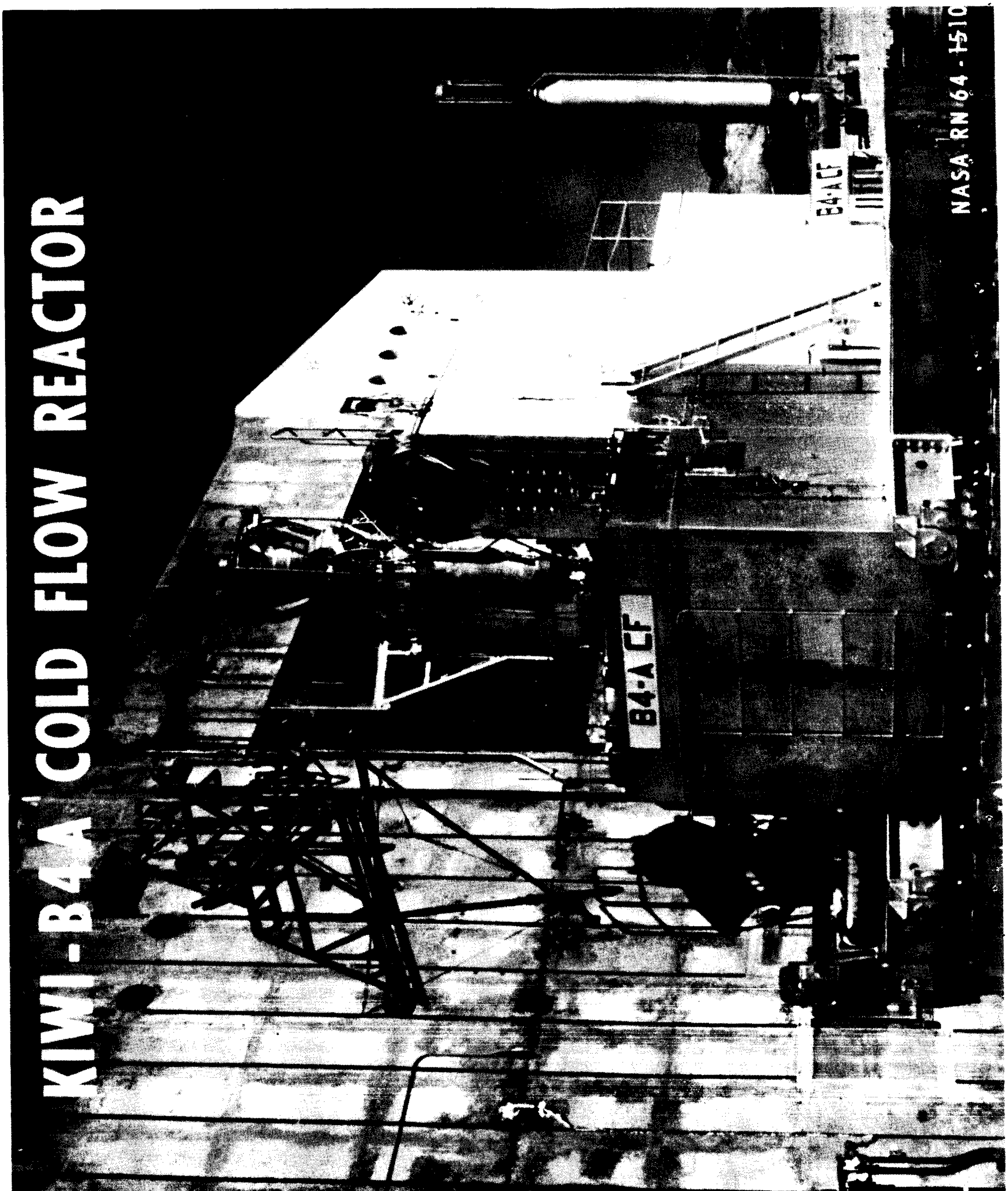


Figure III-8

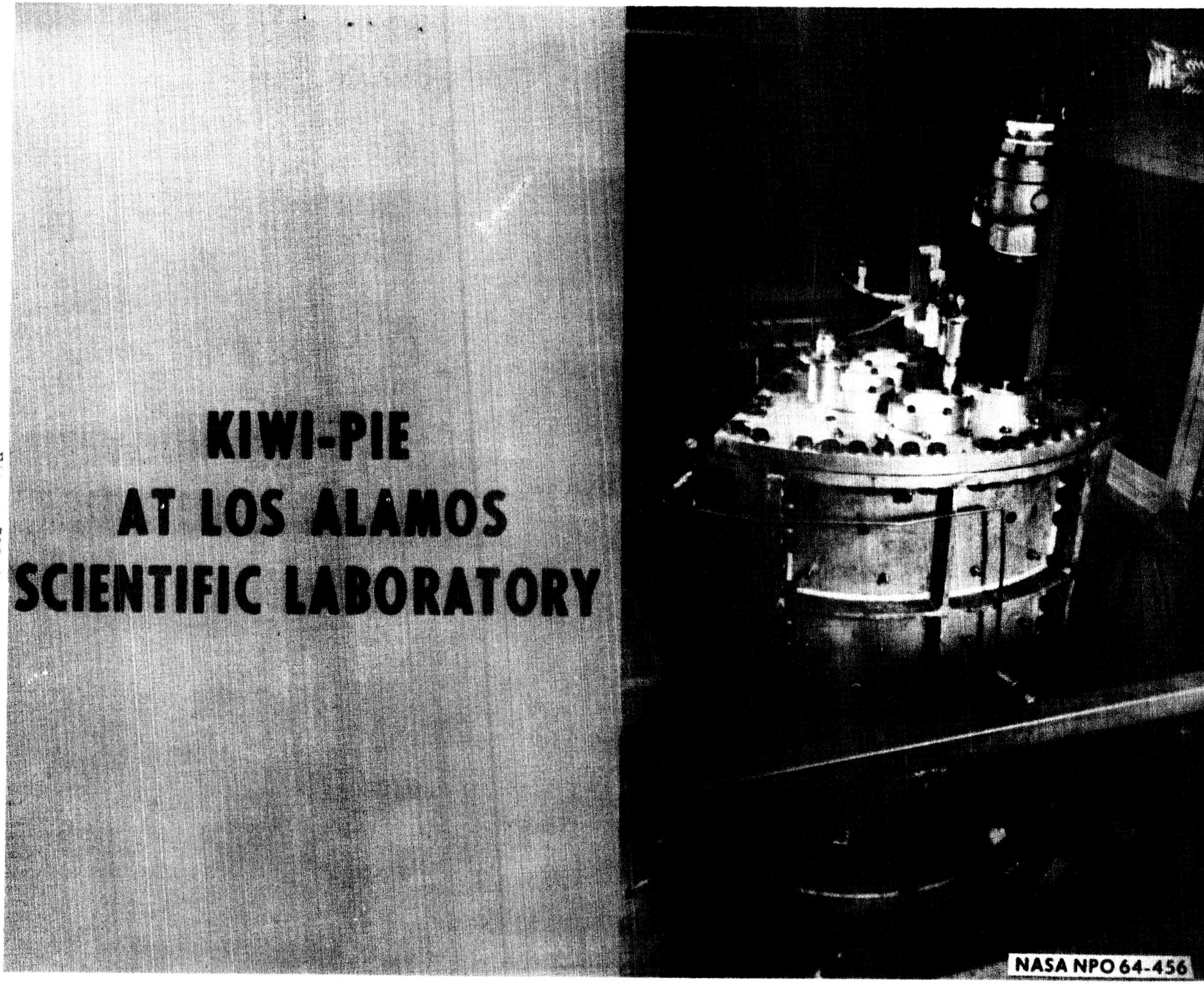
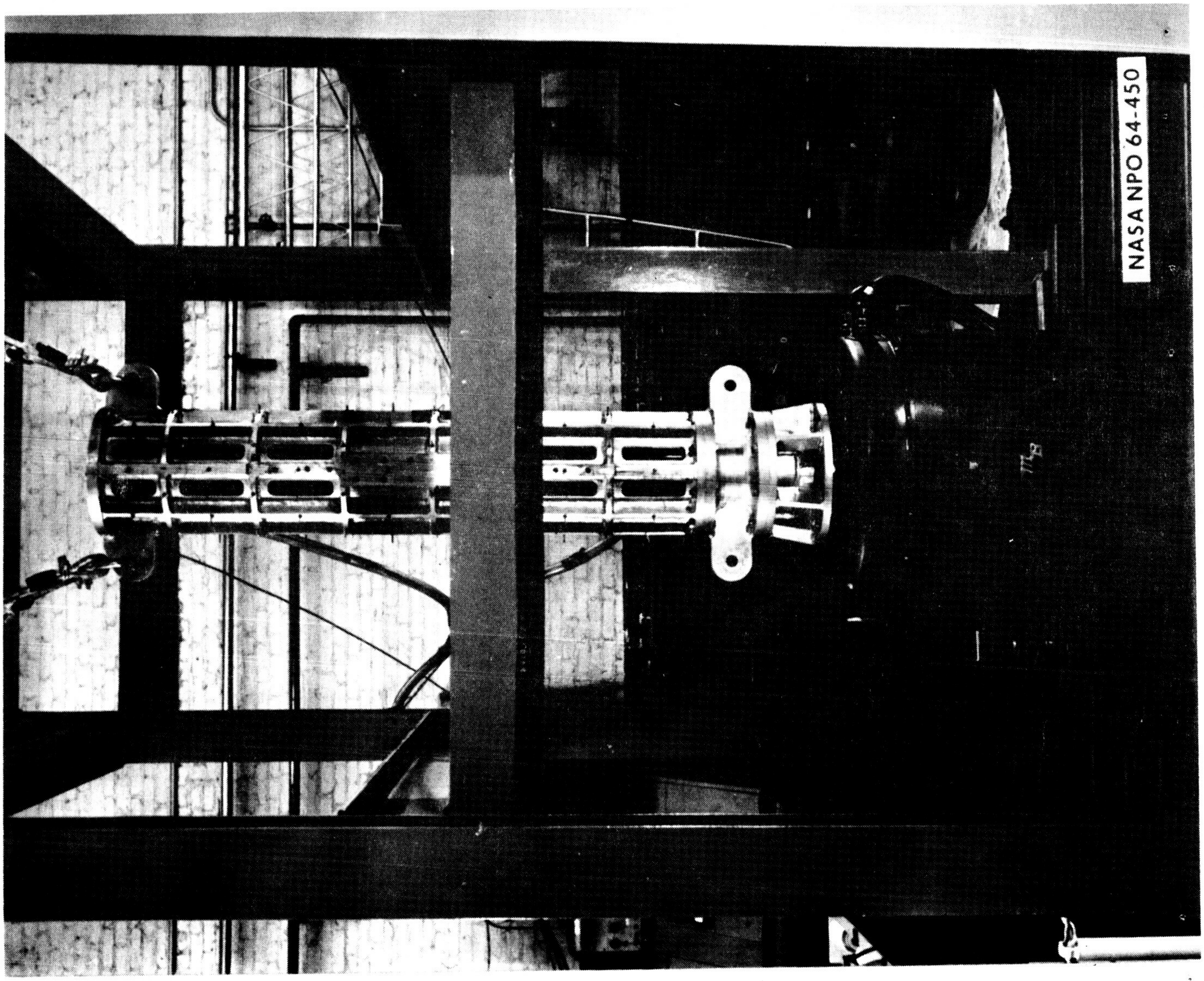

KIWI-PIE
AT LOS ALAMOS
SCIENTIFIC LABORATORY
NASA NPO 64-456

Figure III-9



FUEL ELEMENT VIBRATION TEST

Figure III-10

REACTOR CONTROL DRUM ENVIRONMENTAL TEST CHAMBER

Figure III-11

NERVA REACTOR
VIBRATION TEST

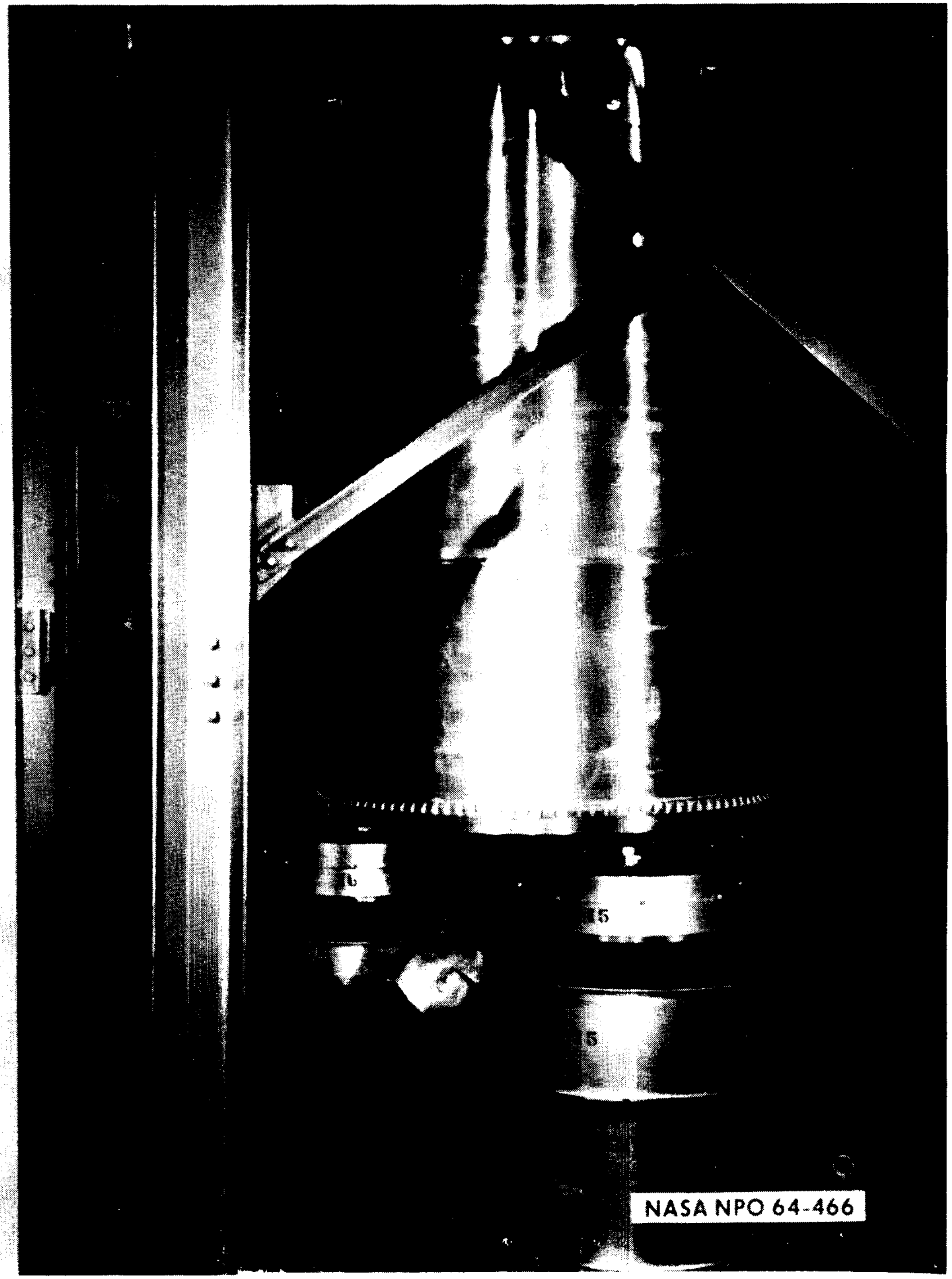


Figure III-12

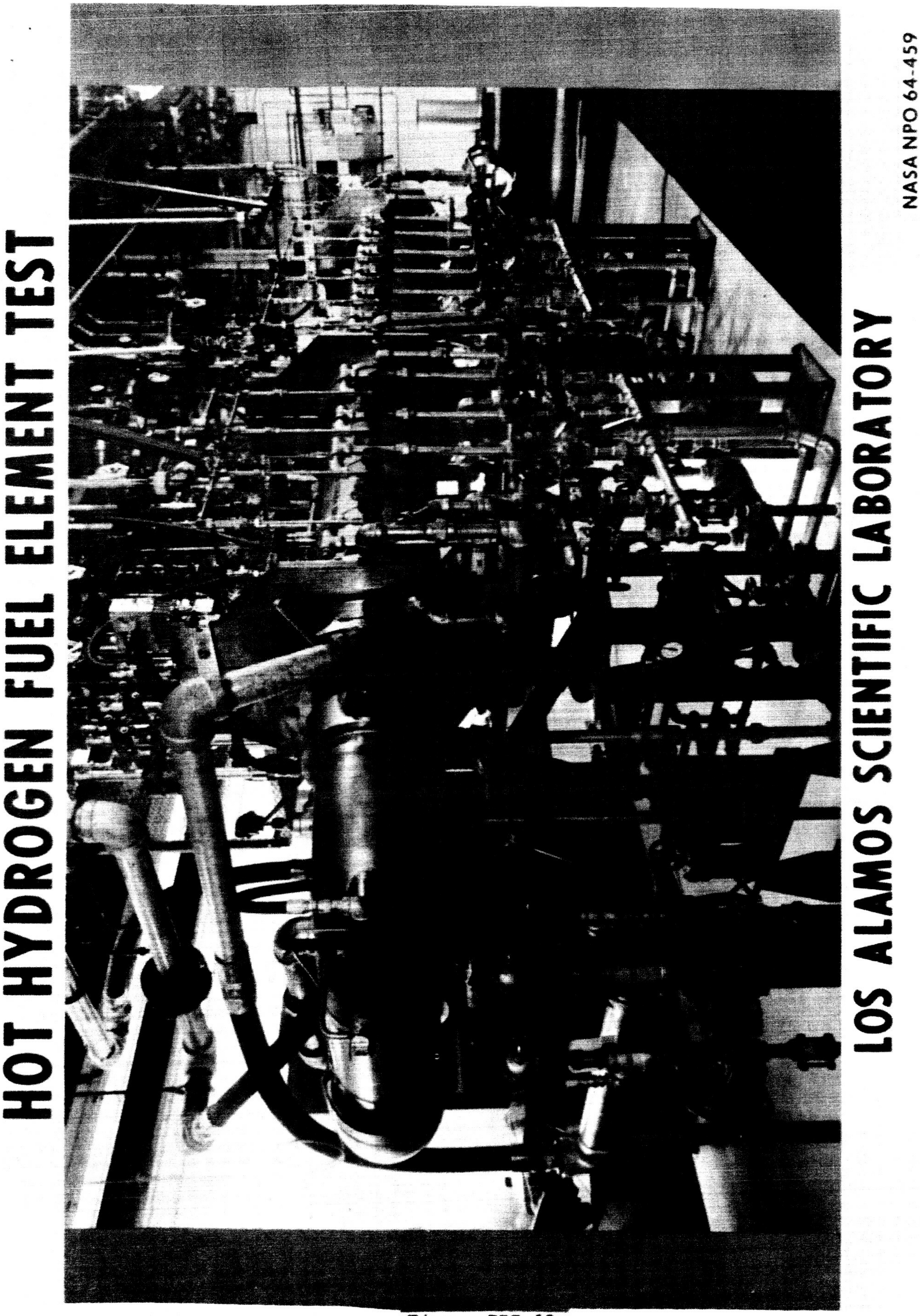


Figure III-13

NERVA REACTOR CRITICAL EXPERIMENT

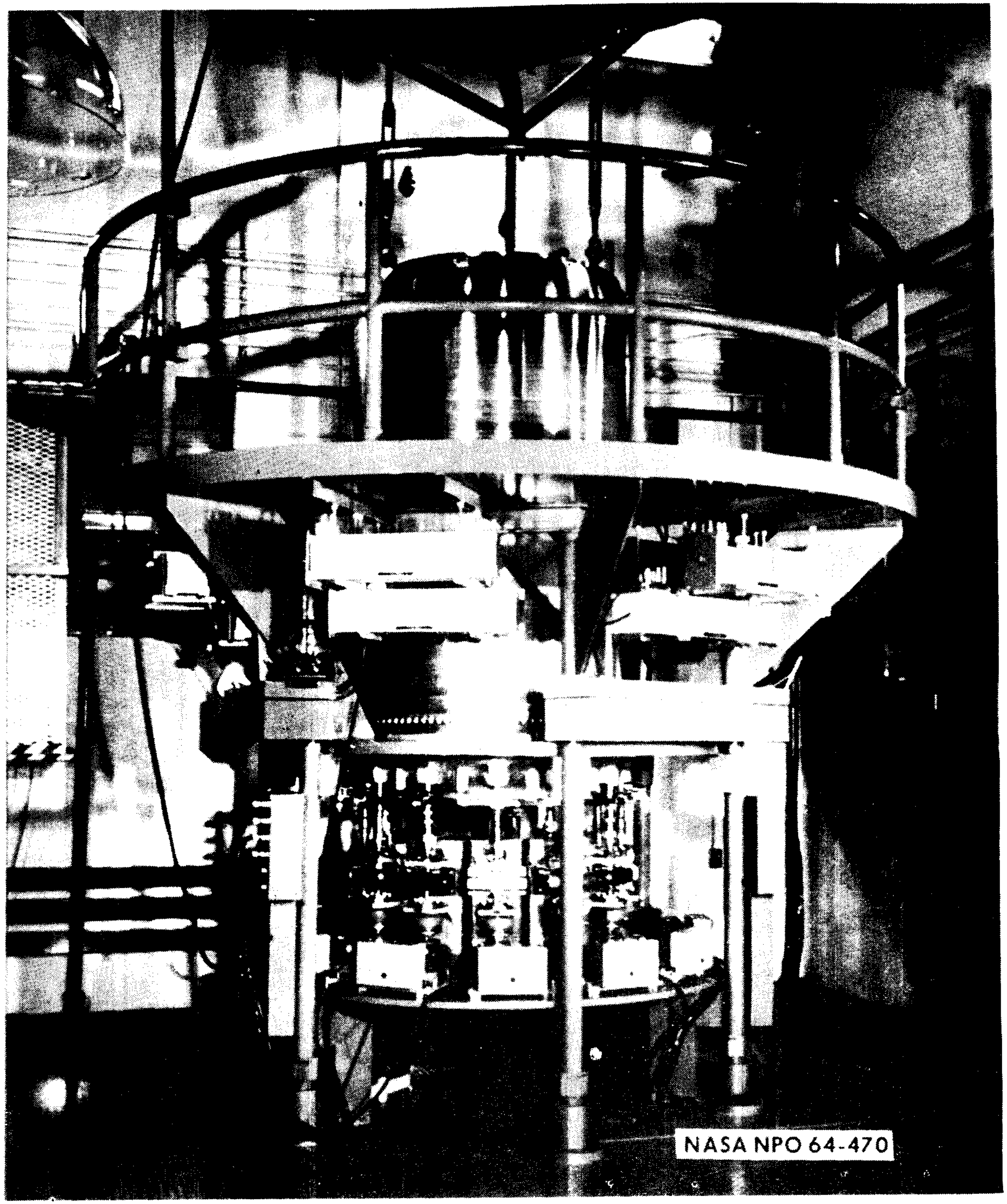


Figure III-14

NRX-A1 REACTOR

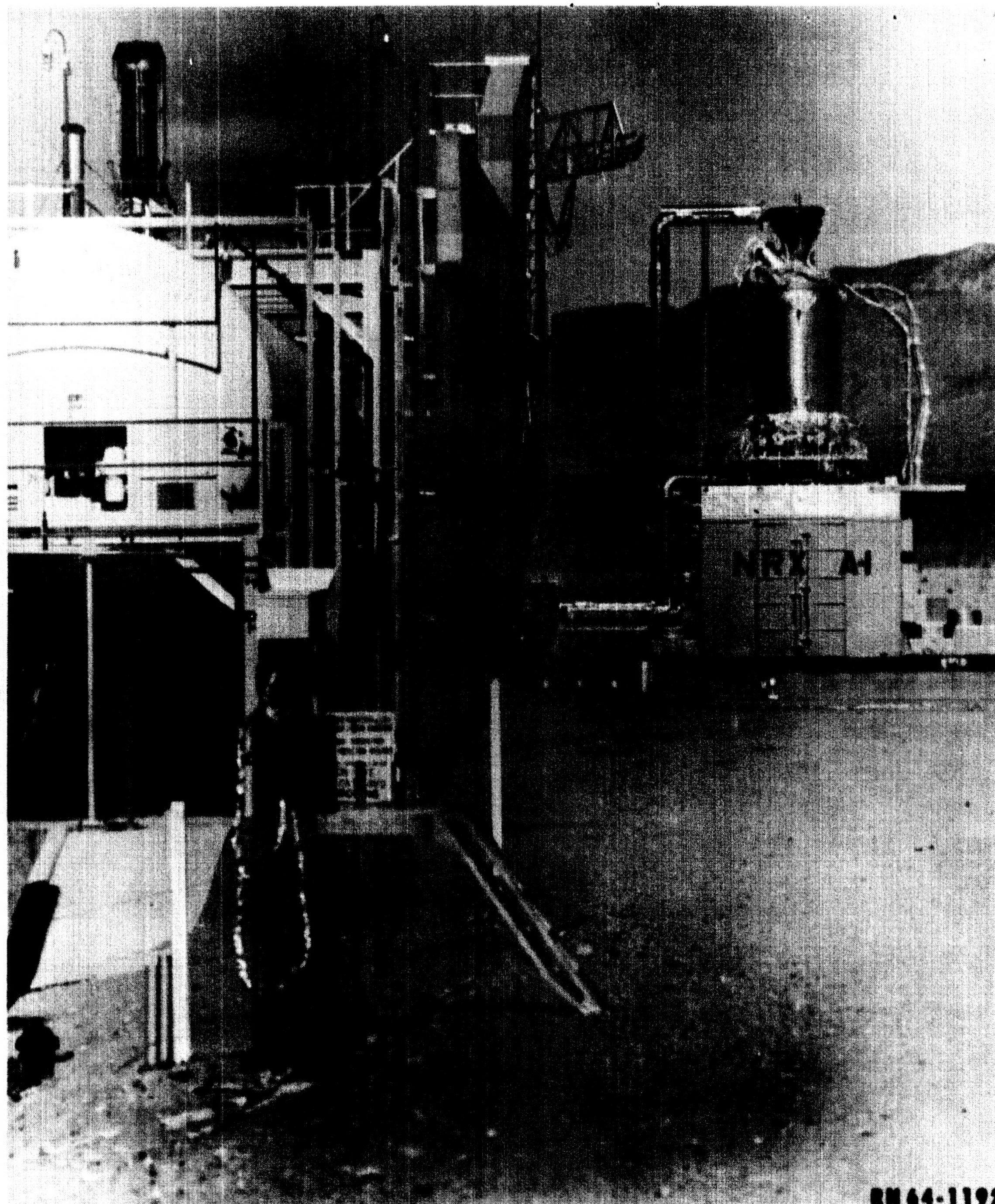


Figure III-15

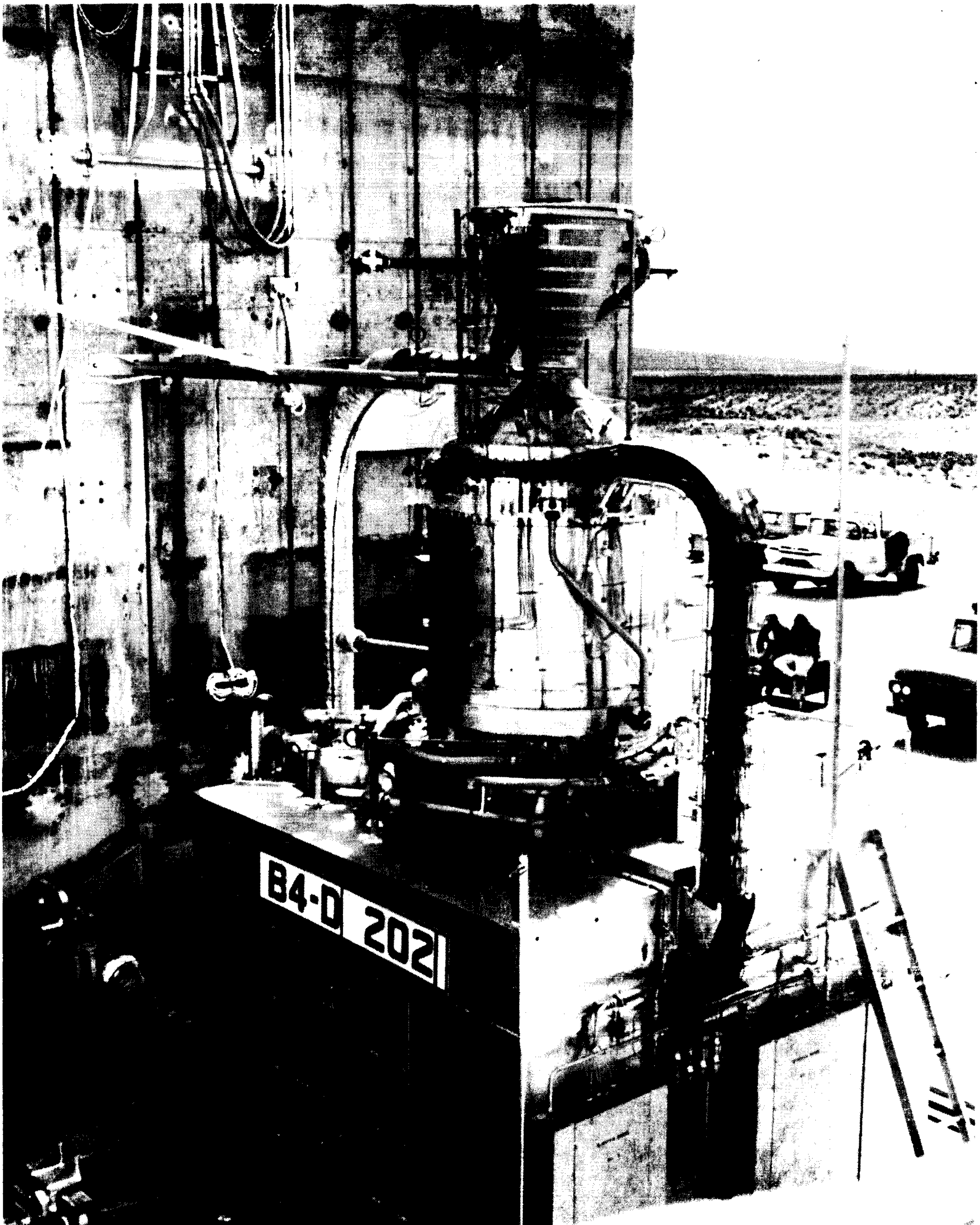
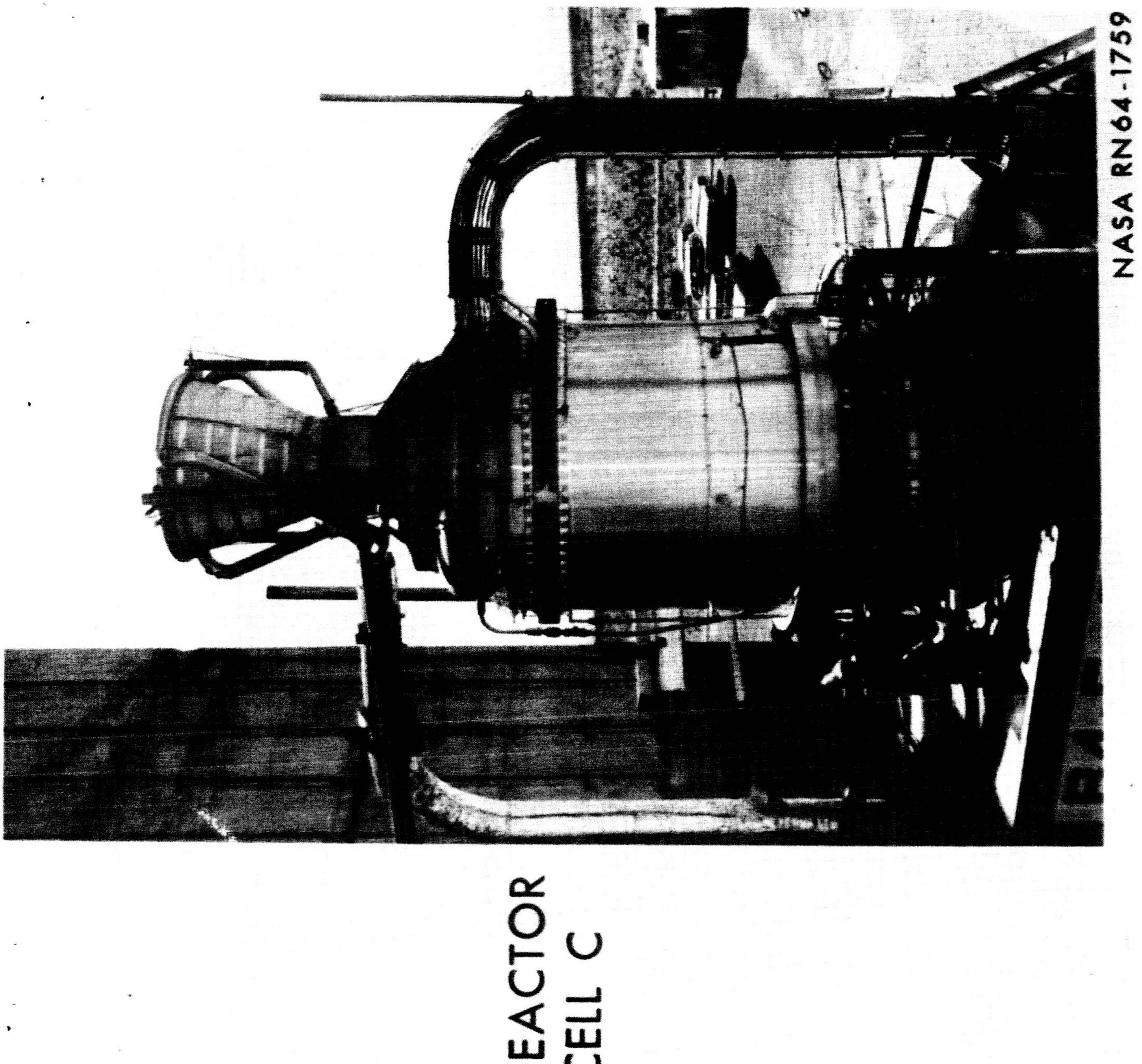


Figure III-16



KIWI-B4E REACTOR
AT TEST CELL C

Figure III-17

Figure III-18

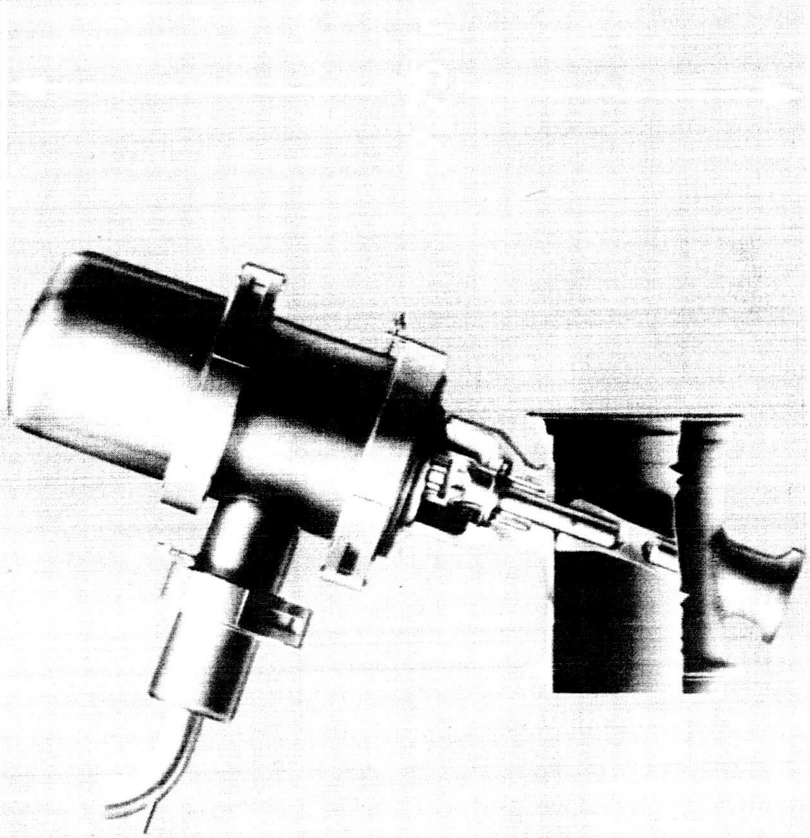
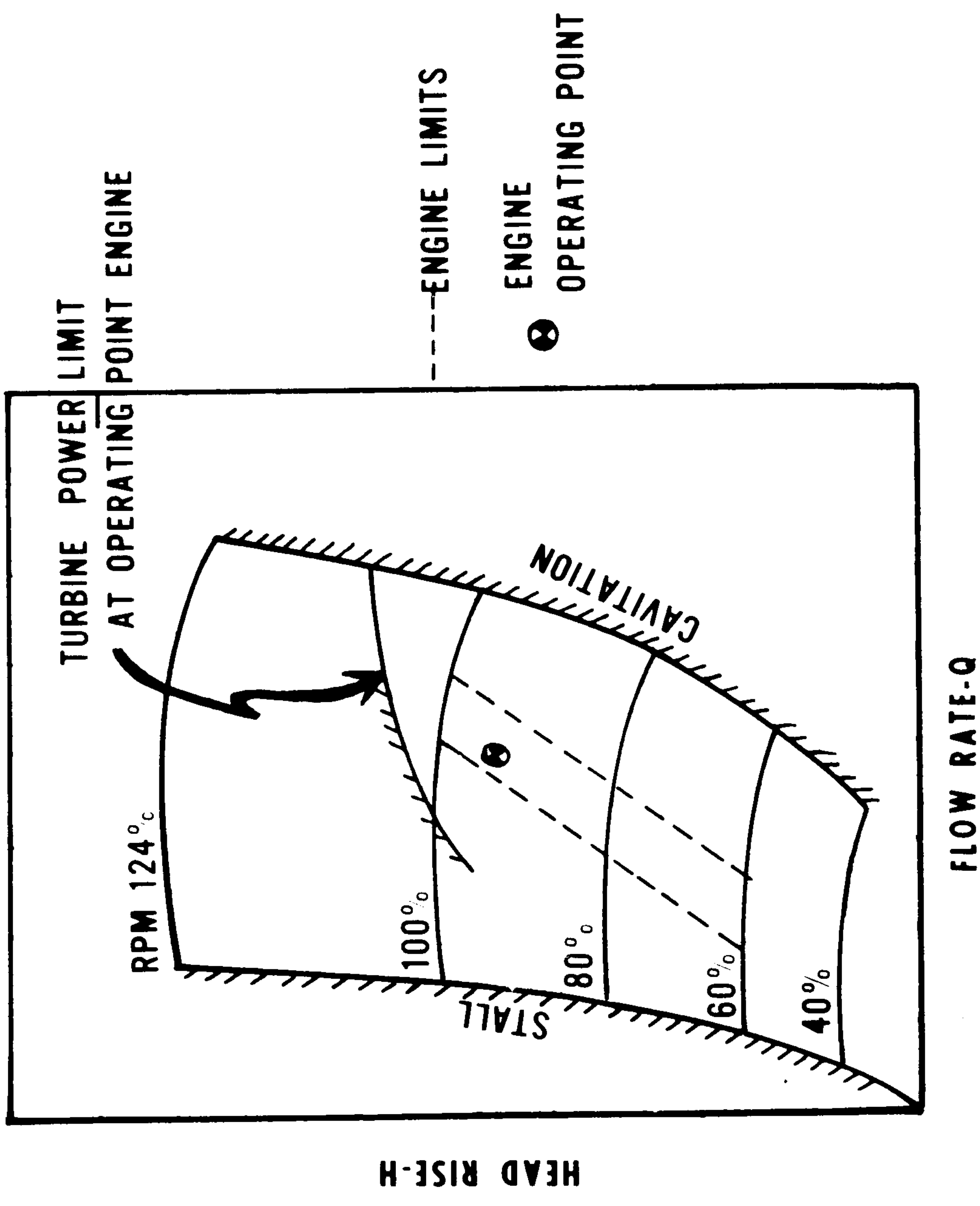


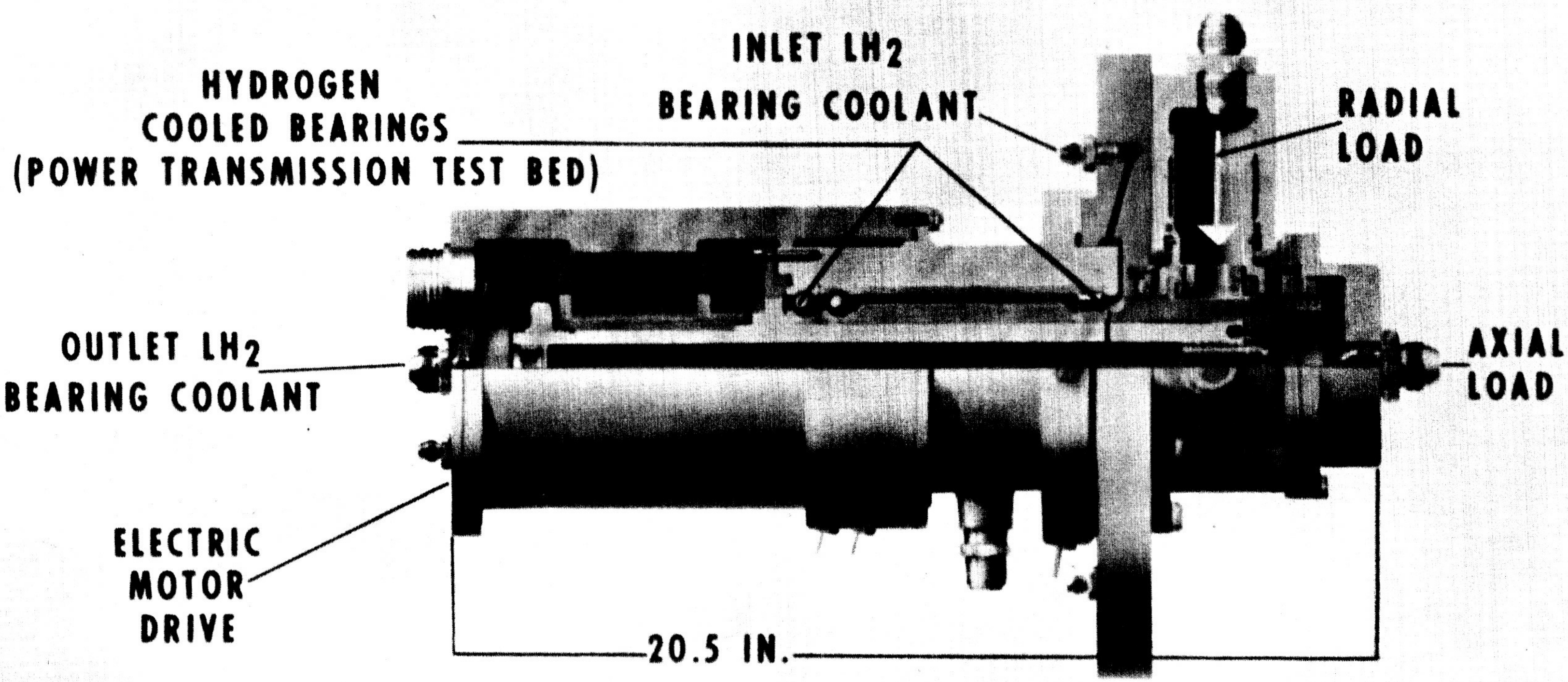
Figure III-19



TURBOPUMP OPERATING MAP

Figure III-20

NERVA BEARING TESTER



TEST SPEED (RPM)________24,000
AXIAL LOAD (LB)________2,000
RADIAL LOAD (LB)________2,000

NASA NPO 64-670

Figure III-21

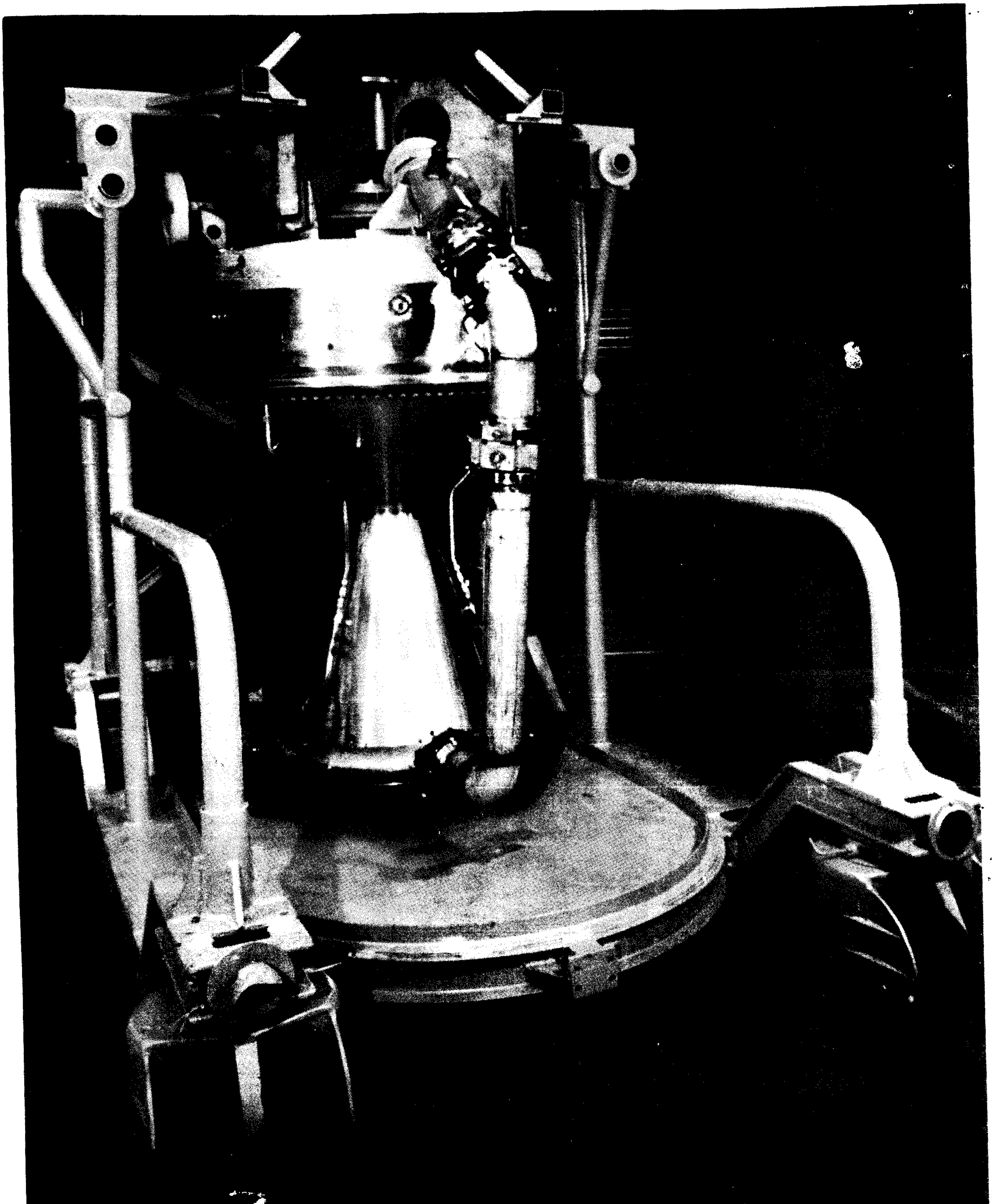


Figure III-22

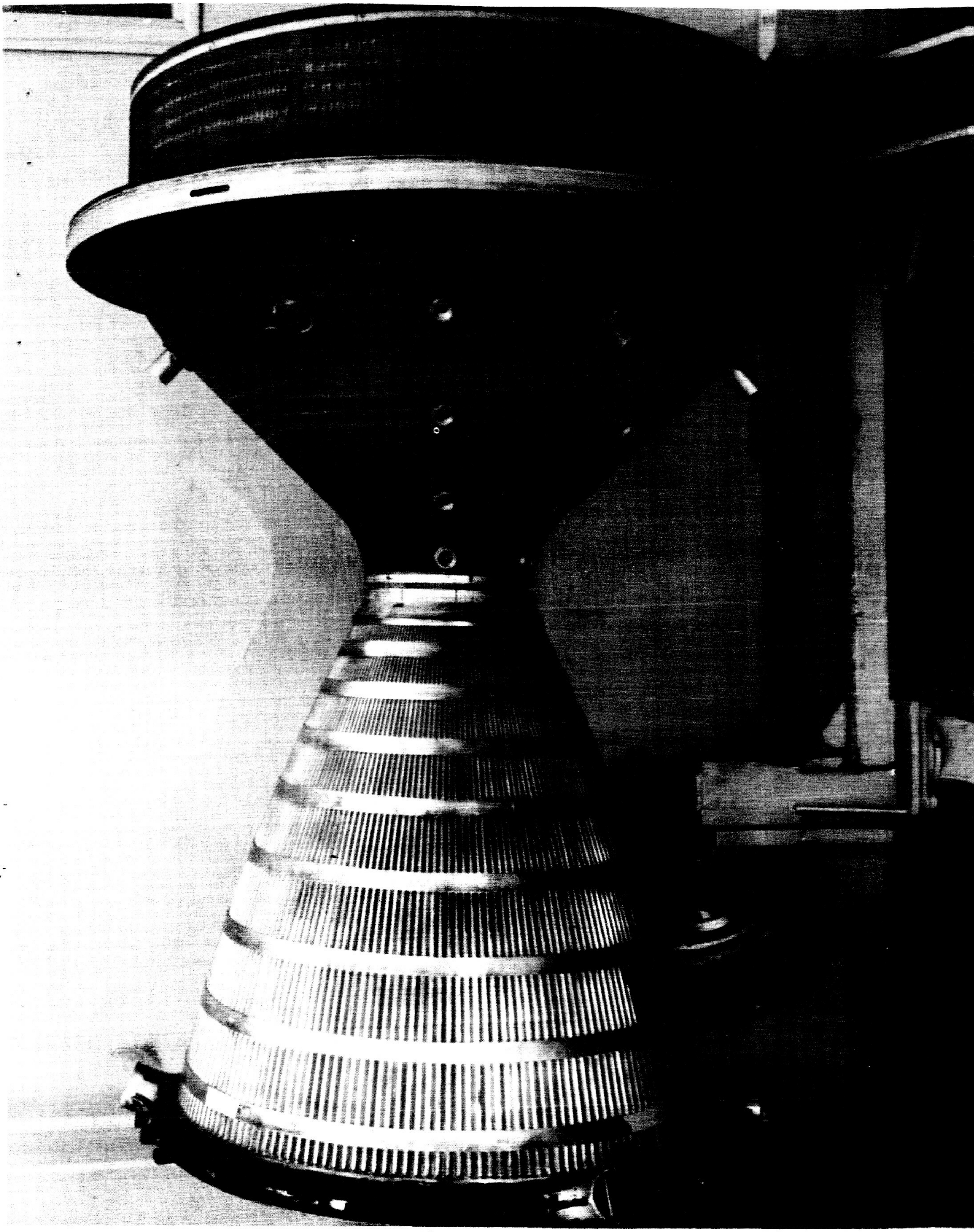


Figure III-23

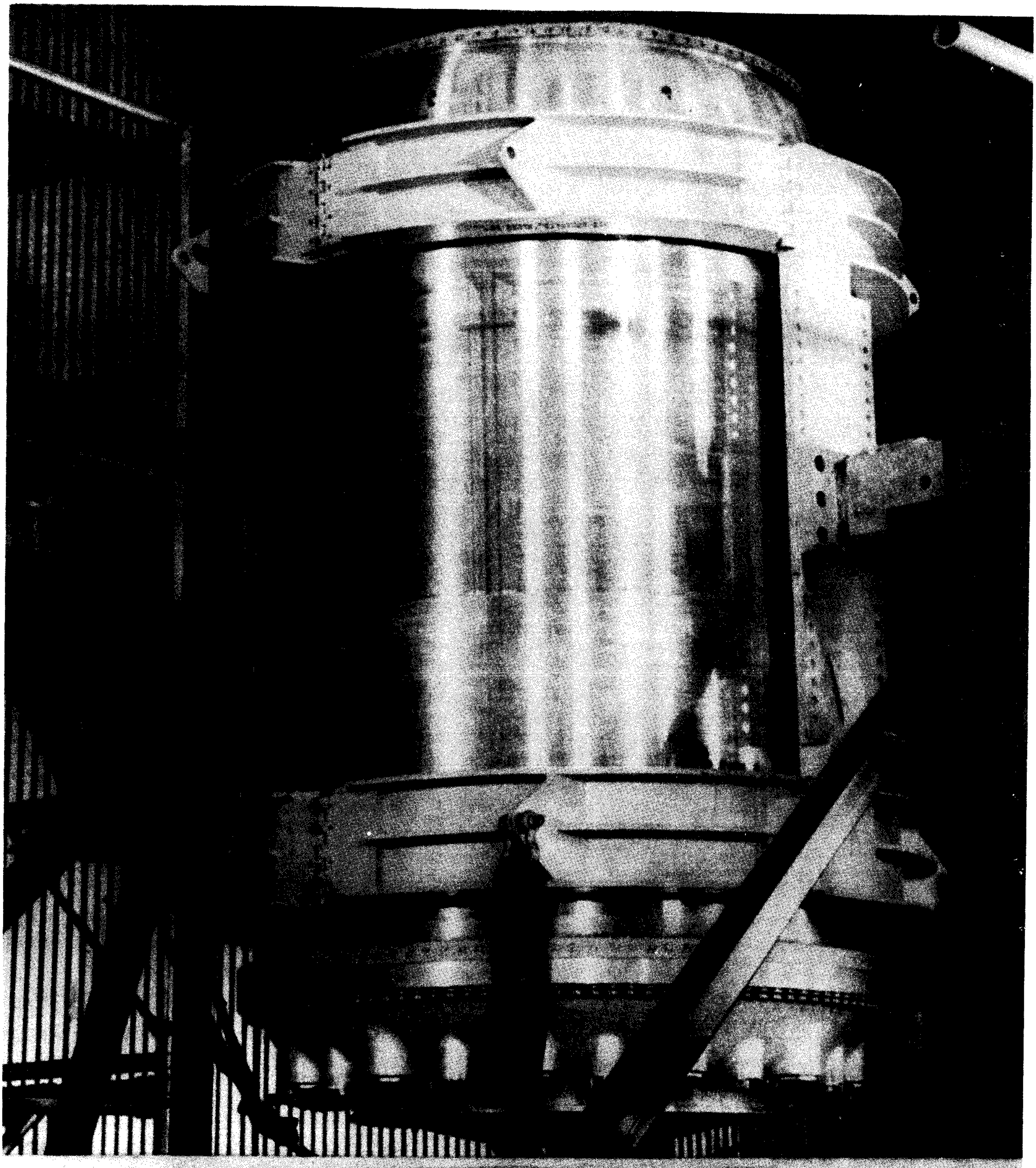


Figure III-24

NERVA THRUST STRUCTURE

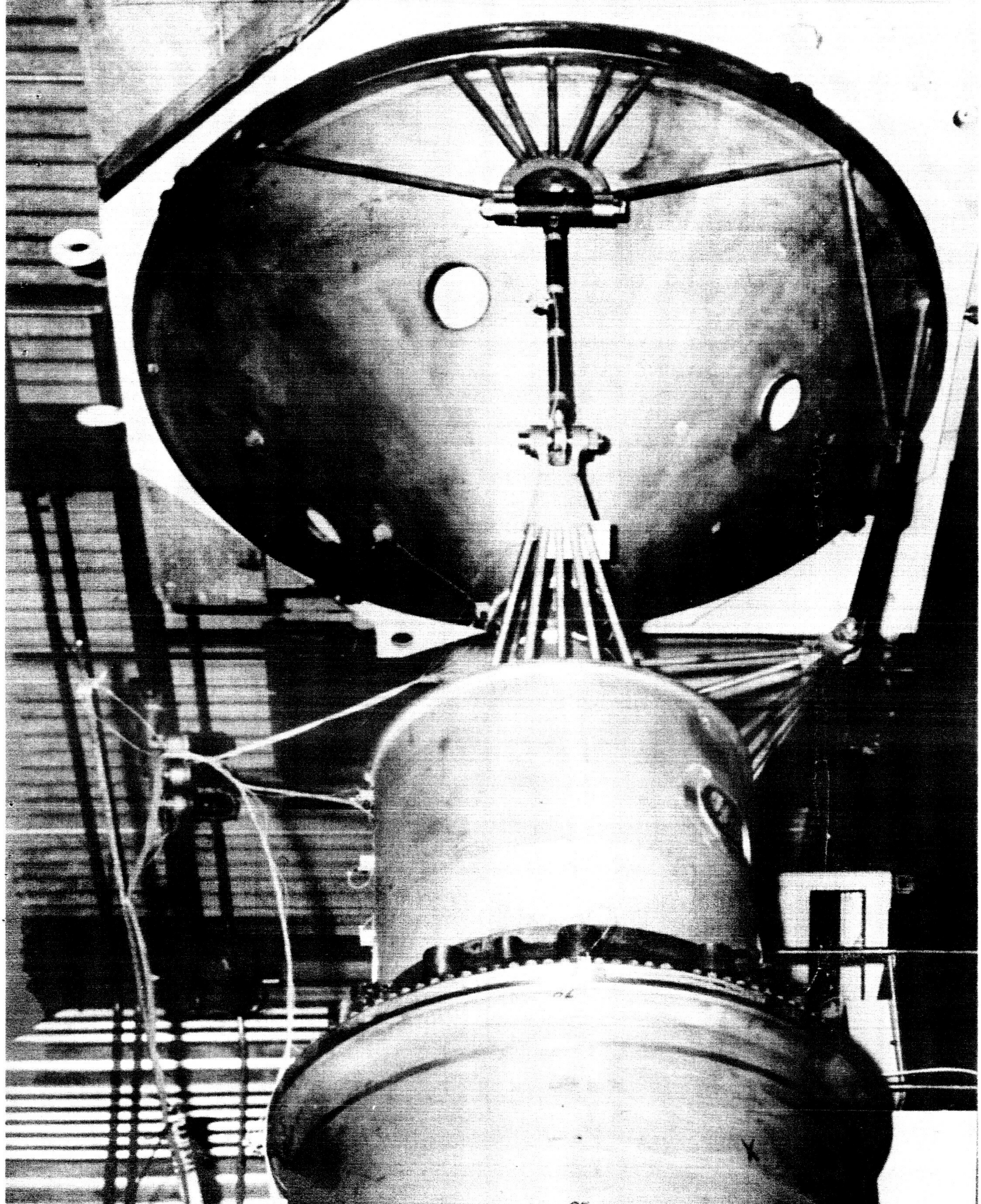
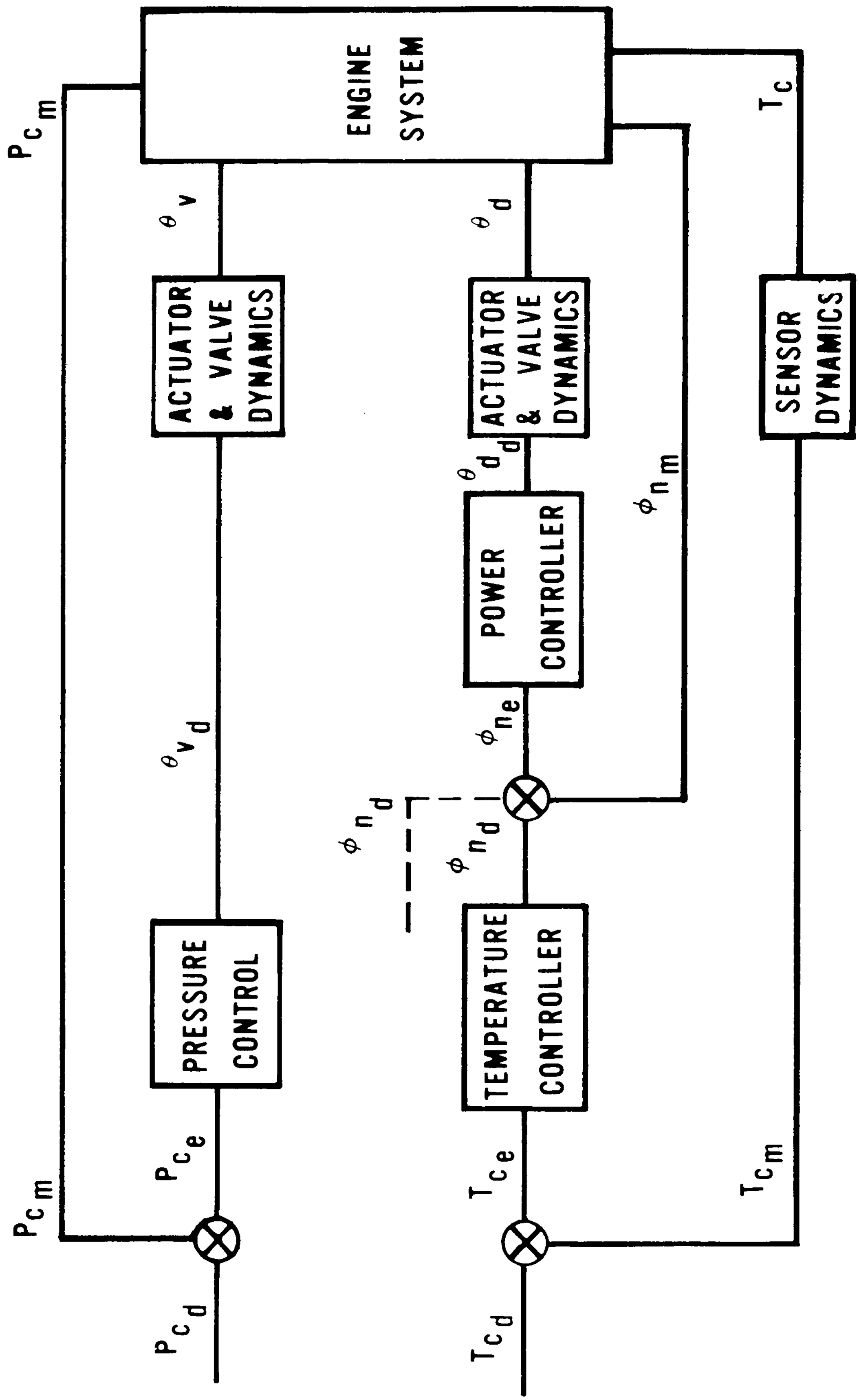


Figure III-25

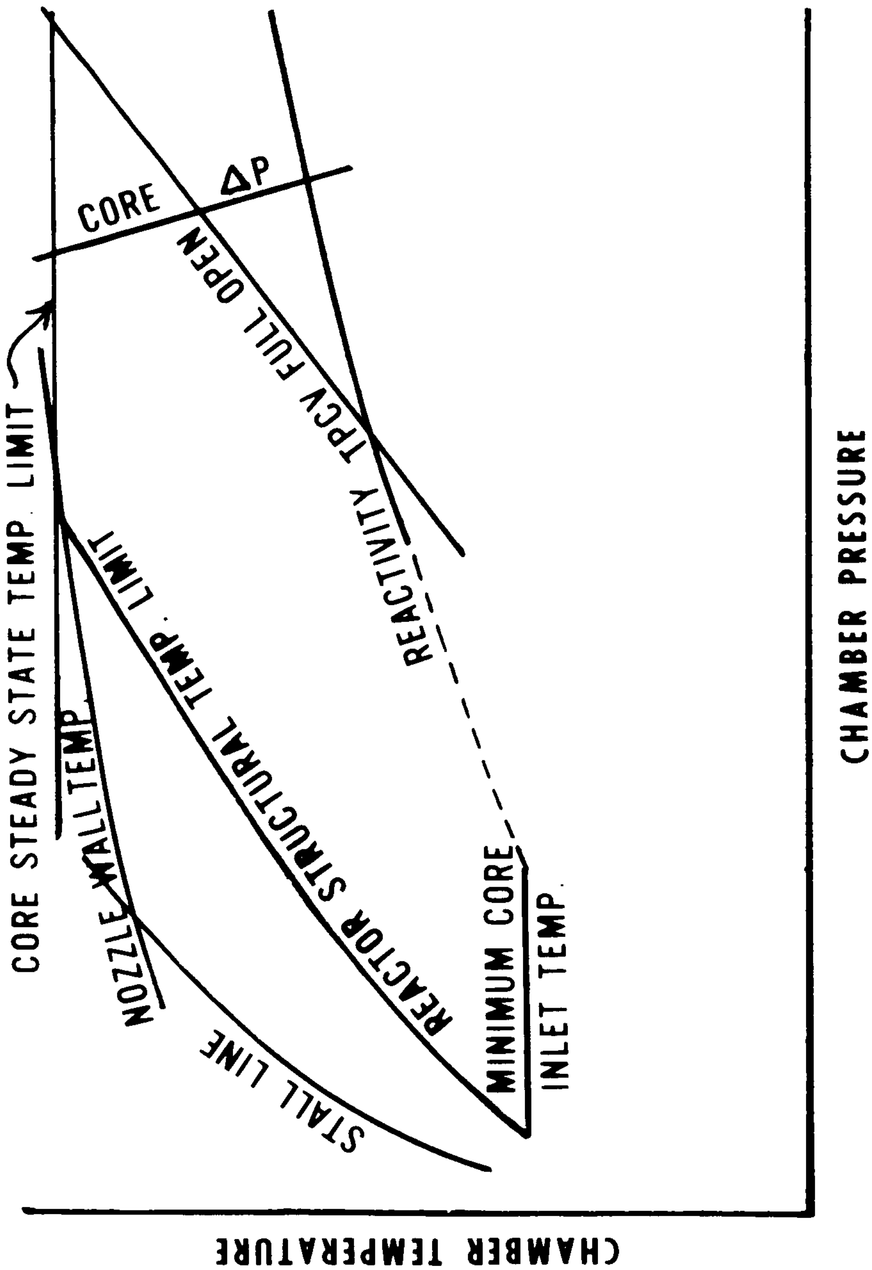


NUCLEAR ROCKET ENGINE CONTROL SYSTEM

Figure III-26

ACTUATOR DEVELOPMENT TEST

Figure III-27



ENGINE STEADY STATE OPERATING MAP

Figure III-28

ENGINE COMPONENTS TEST STAND

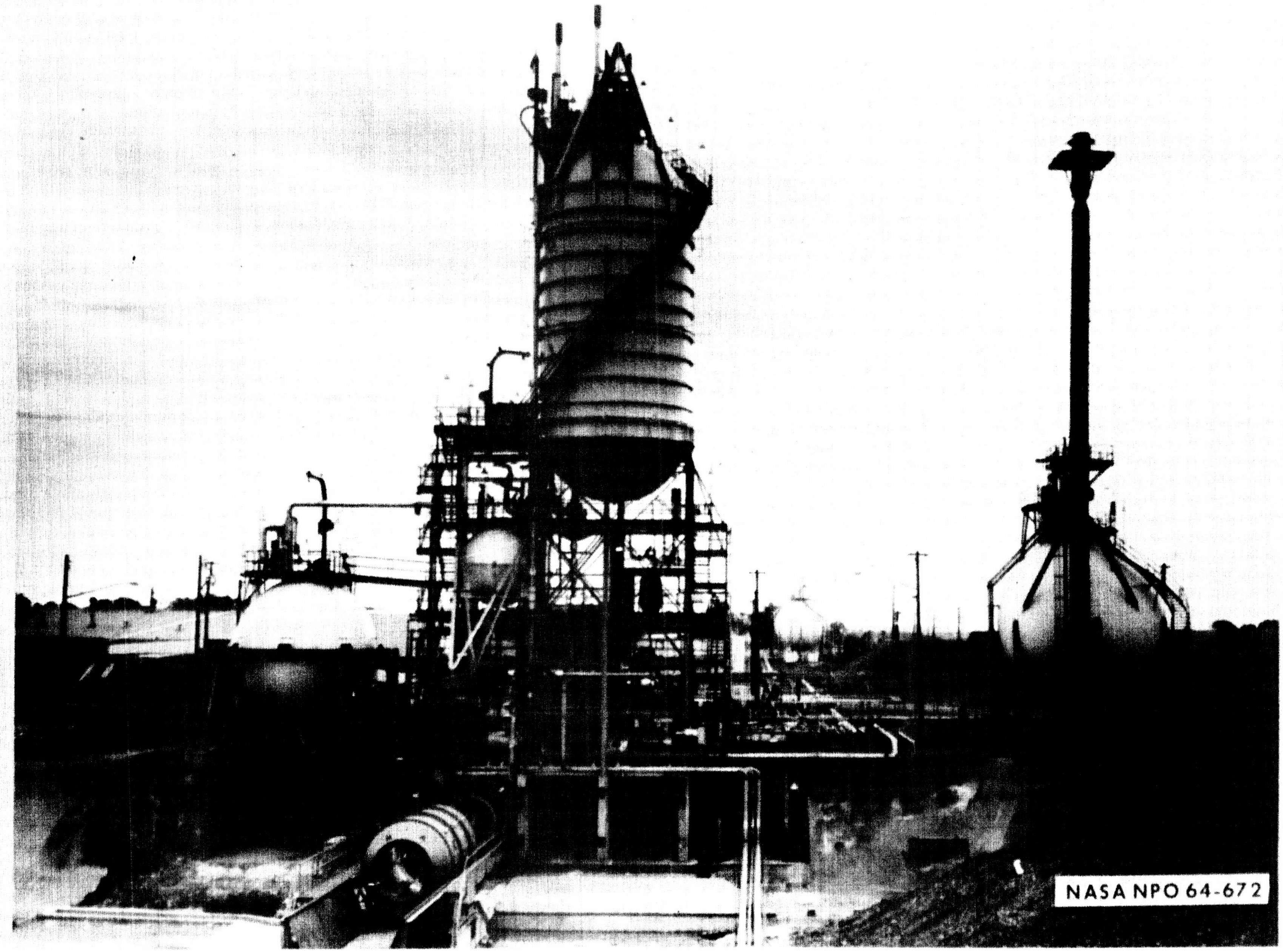


Figure III-29

REACTOR TEST CELL A AT NRDS

NASA NPO64-395

Figure III-

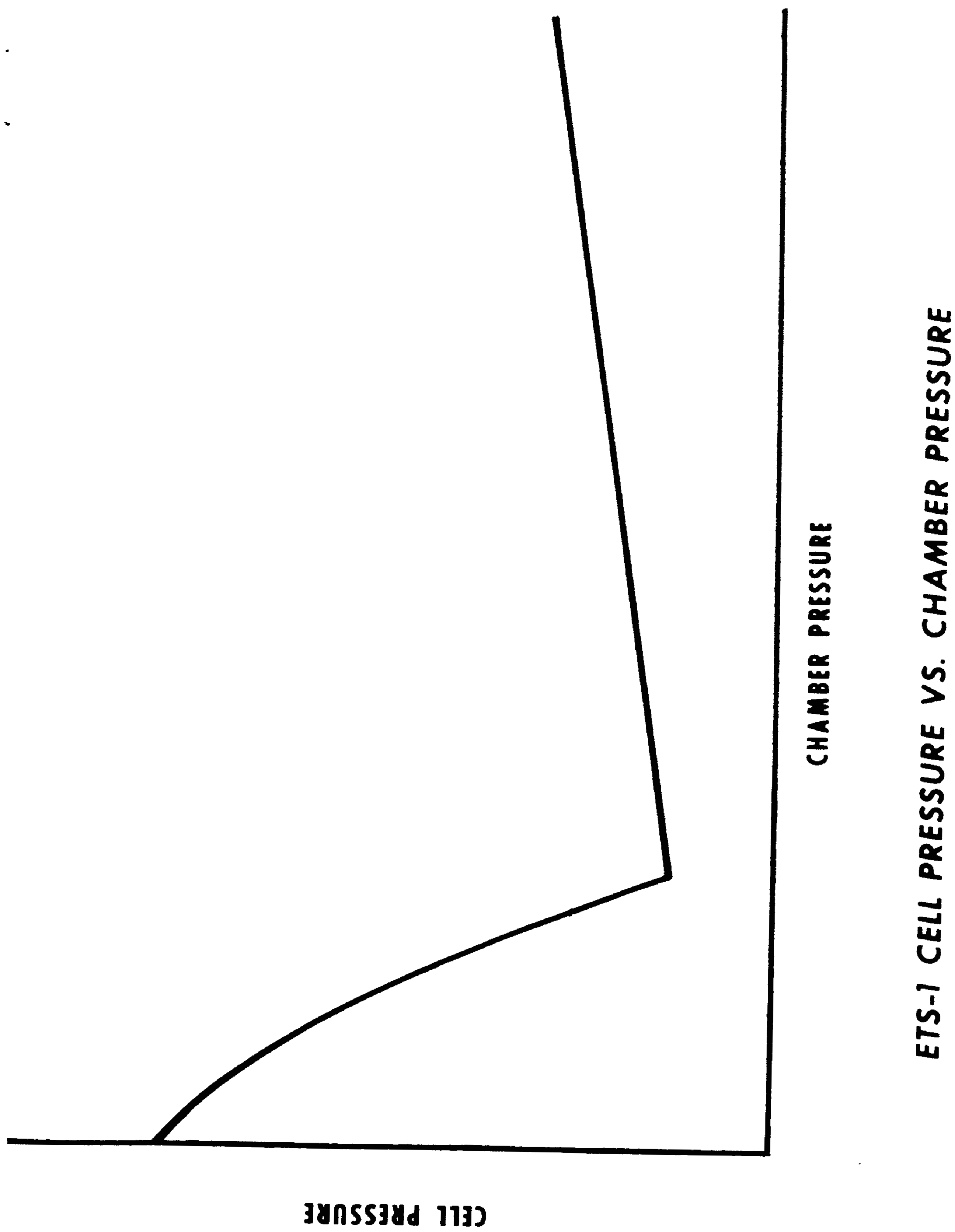


Figure III-31

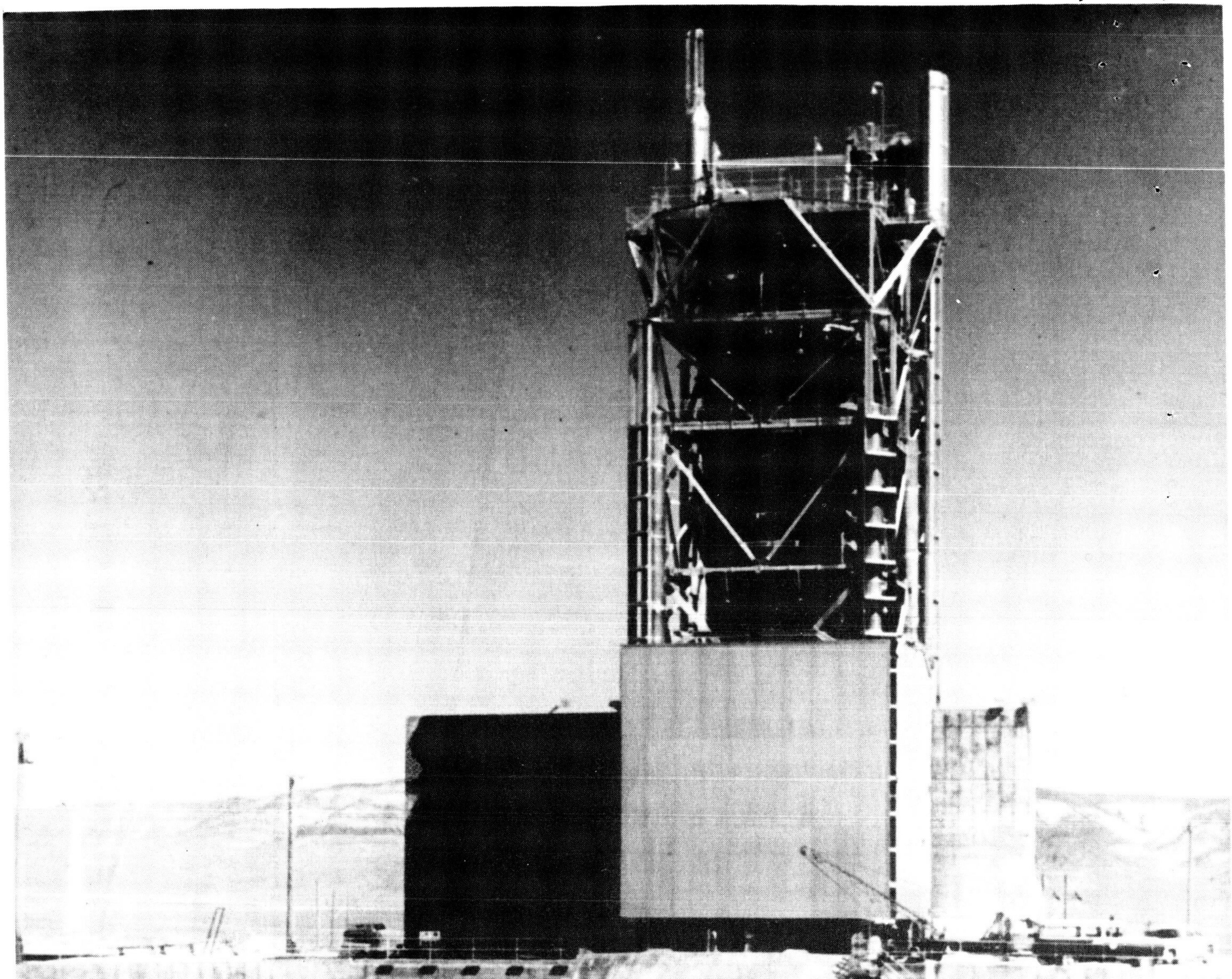


Figure III-32

EFFECTS OF DESIGN CHANGES ON REFLECTOR CONTROL

Δ		CONTROL EFFECT
CORE DIAMETER	↑	↓
PRESSURE	↑	↓
VOID FRACTION	↑	↓
FUEL FRACTION	↑	↑

Figure IV-1

VARIATION OF TURBOPUMP WEIGHT WITH PUMP DISCHARGE PRESSURE

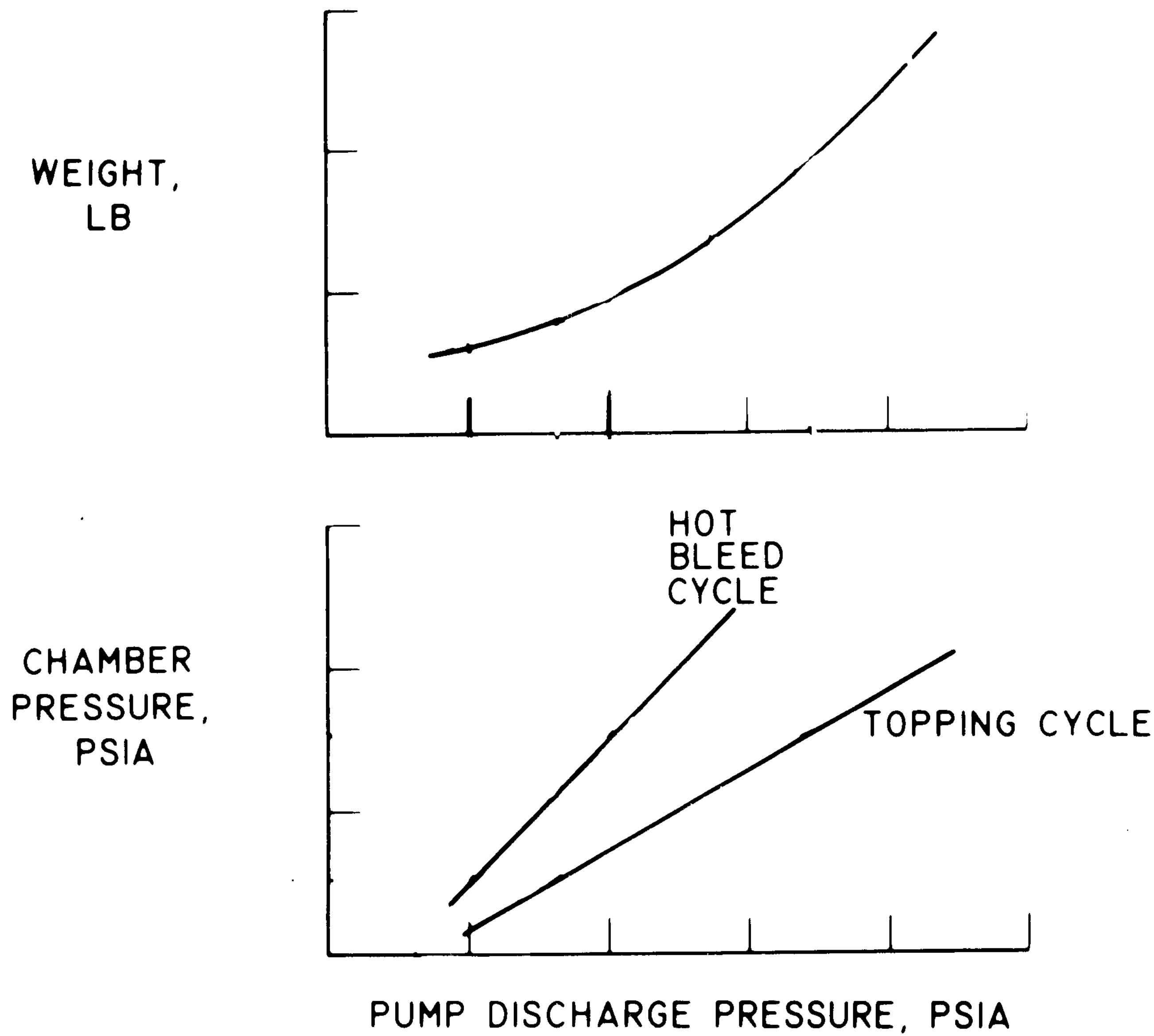


Figure IV-2

VARIATION OF ENGINE WEIGHT WITH CHAMBER PRESSURE

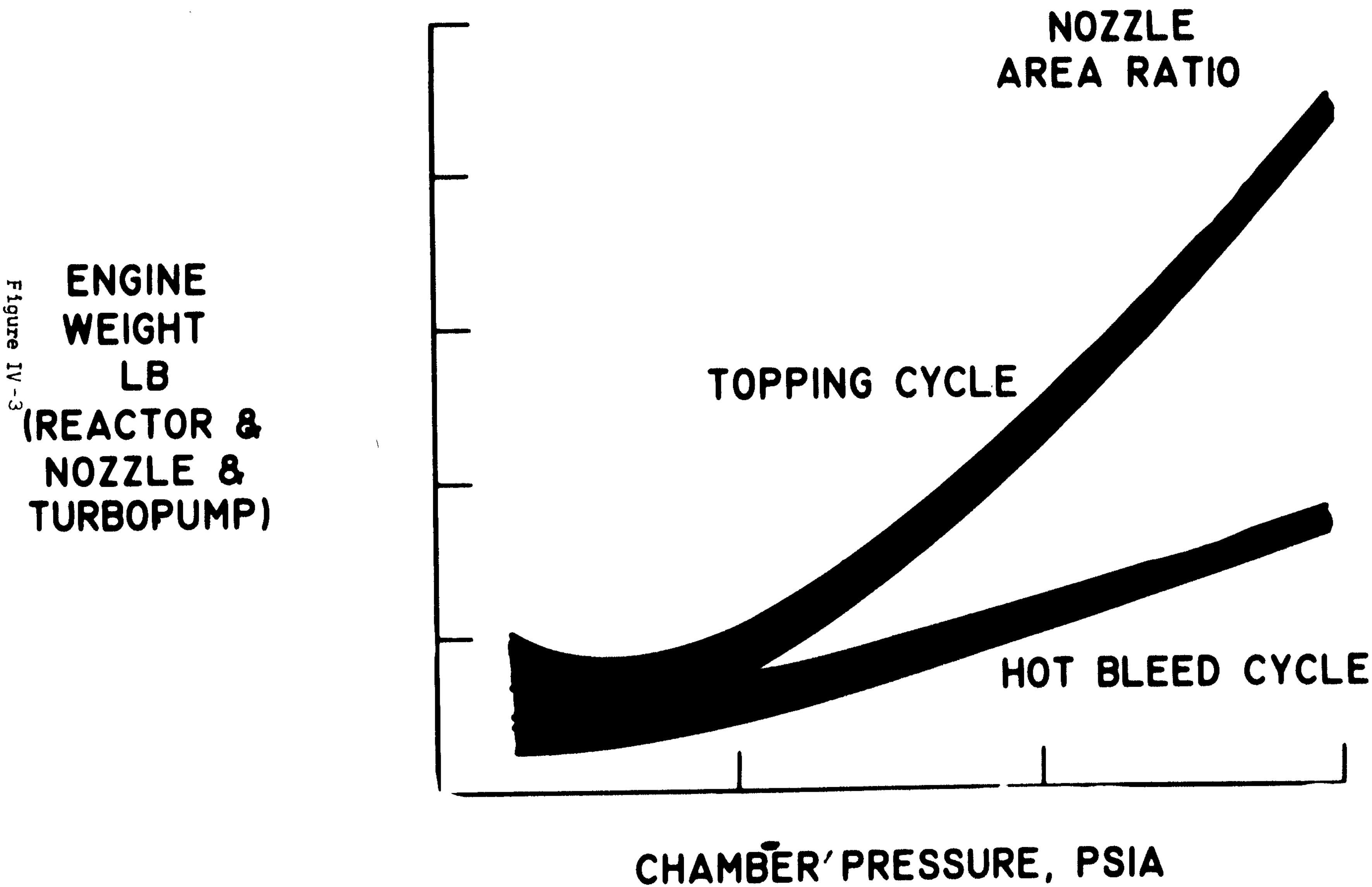


Figure IV-3

HONEYCOMB CRITICAL ASSEMBLY

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Figure IV-4

HYDROGEN HEAT TRANSFER FACILITY

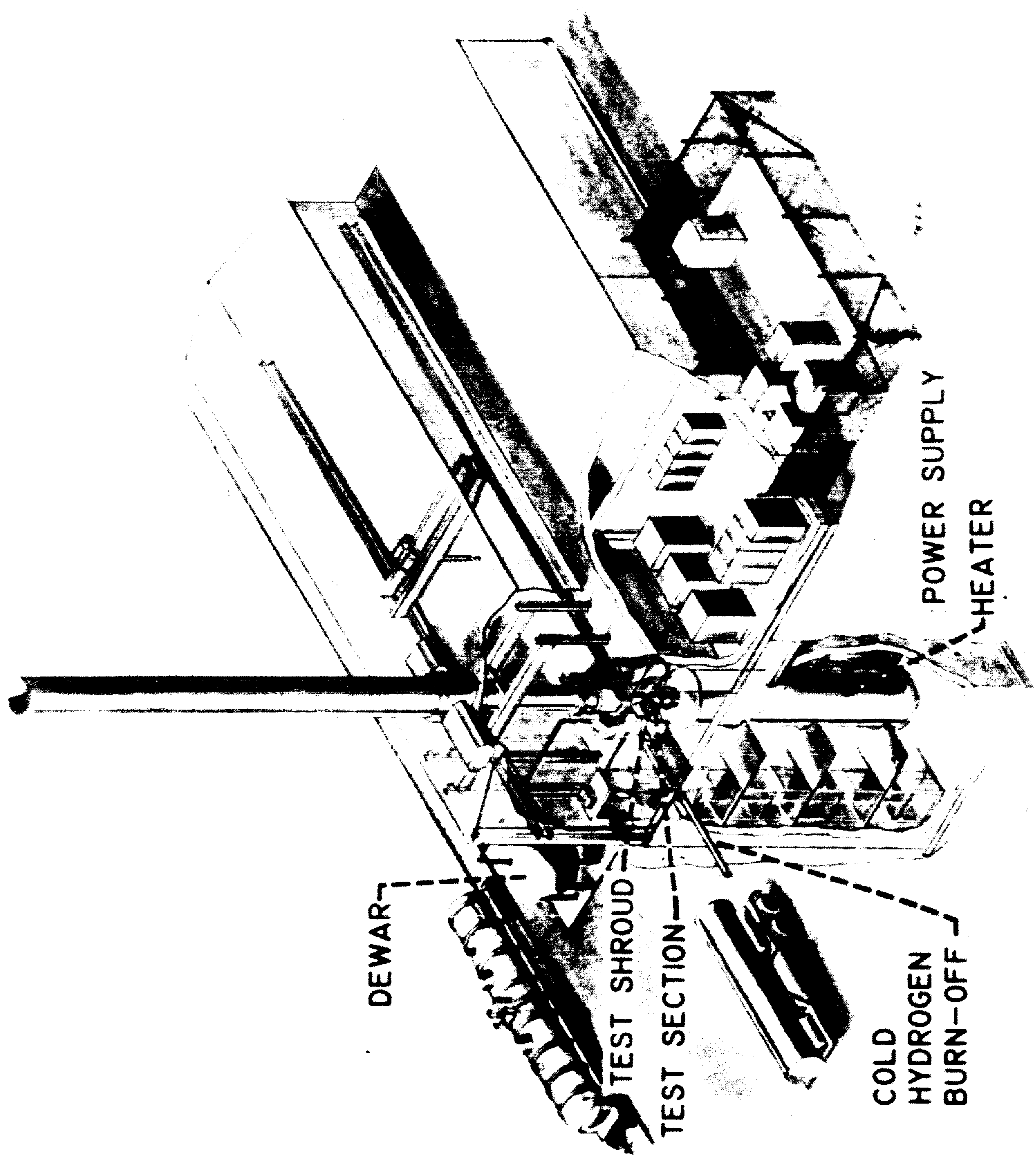


Figure IV-5

TEST AREA SECTION HYDROGEN HEAT TRANSFER FACILITY

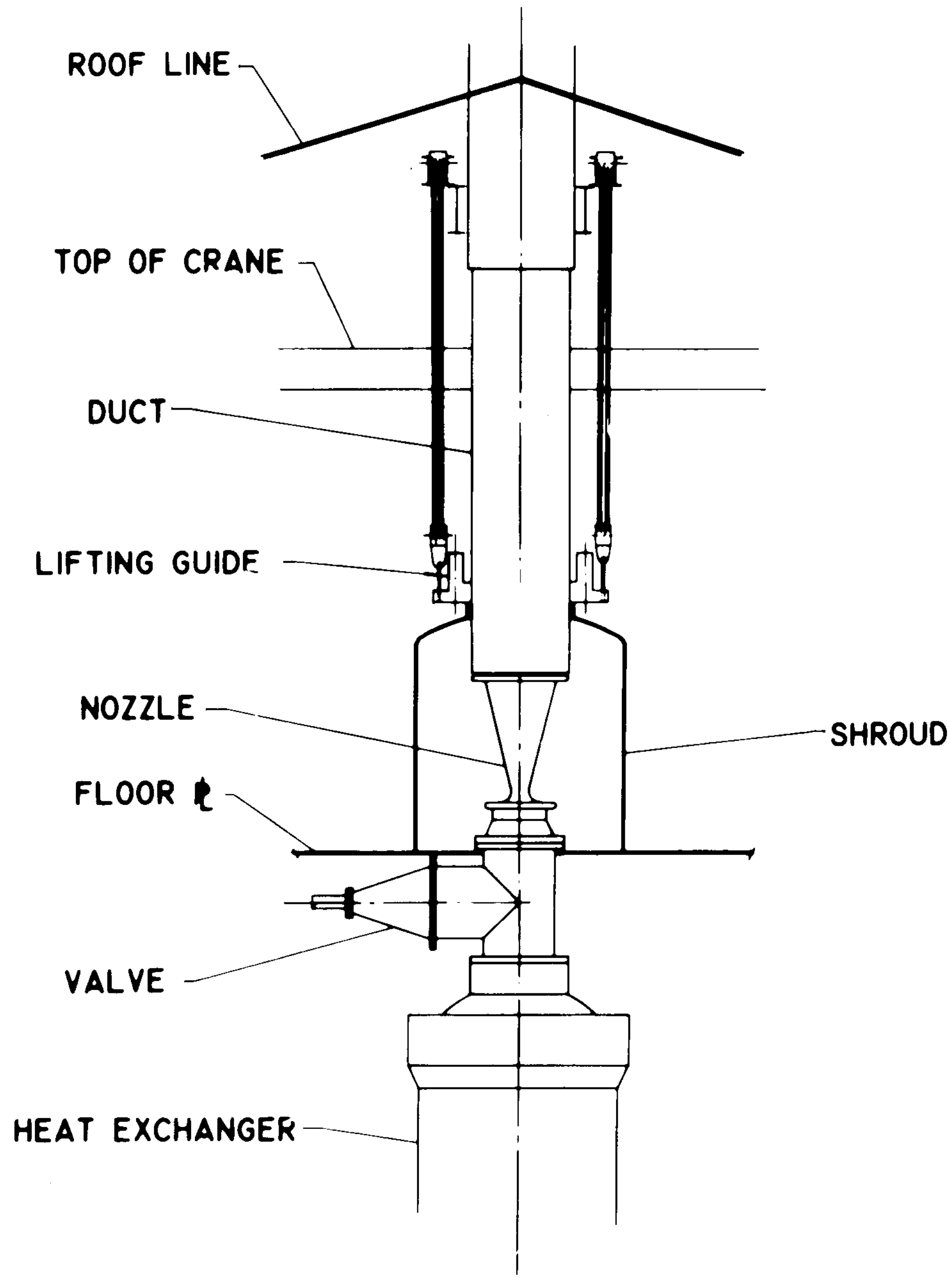
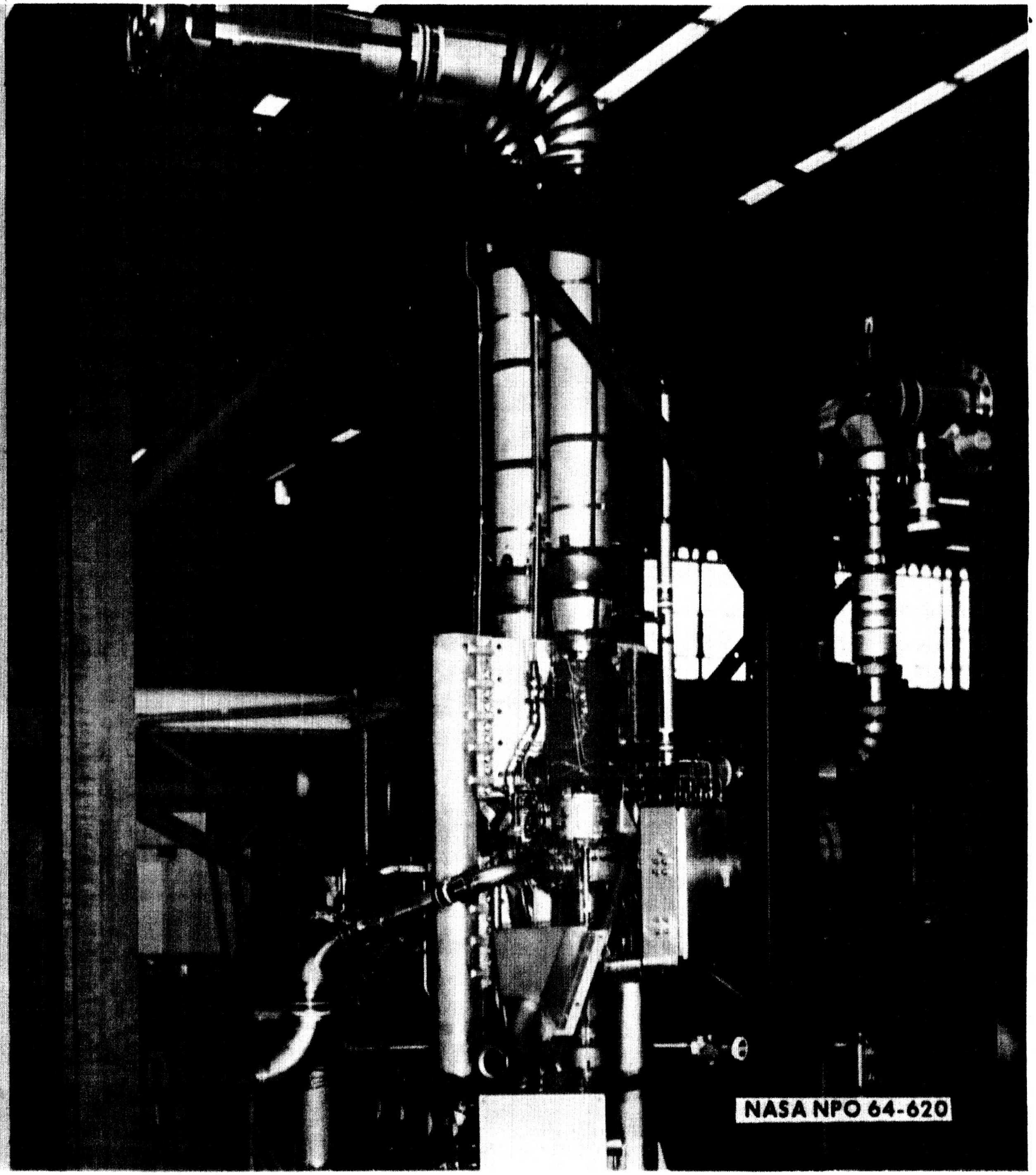
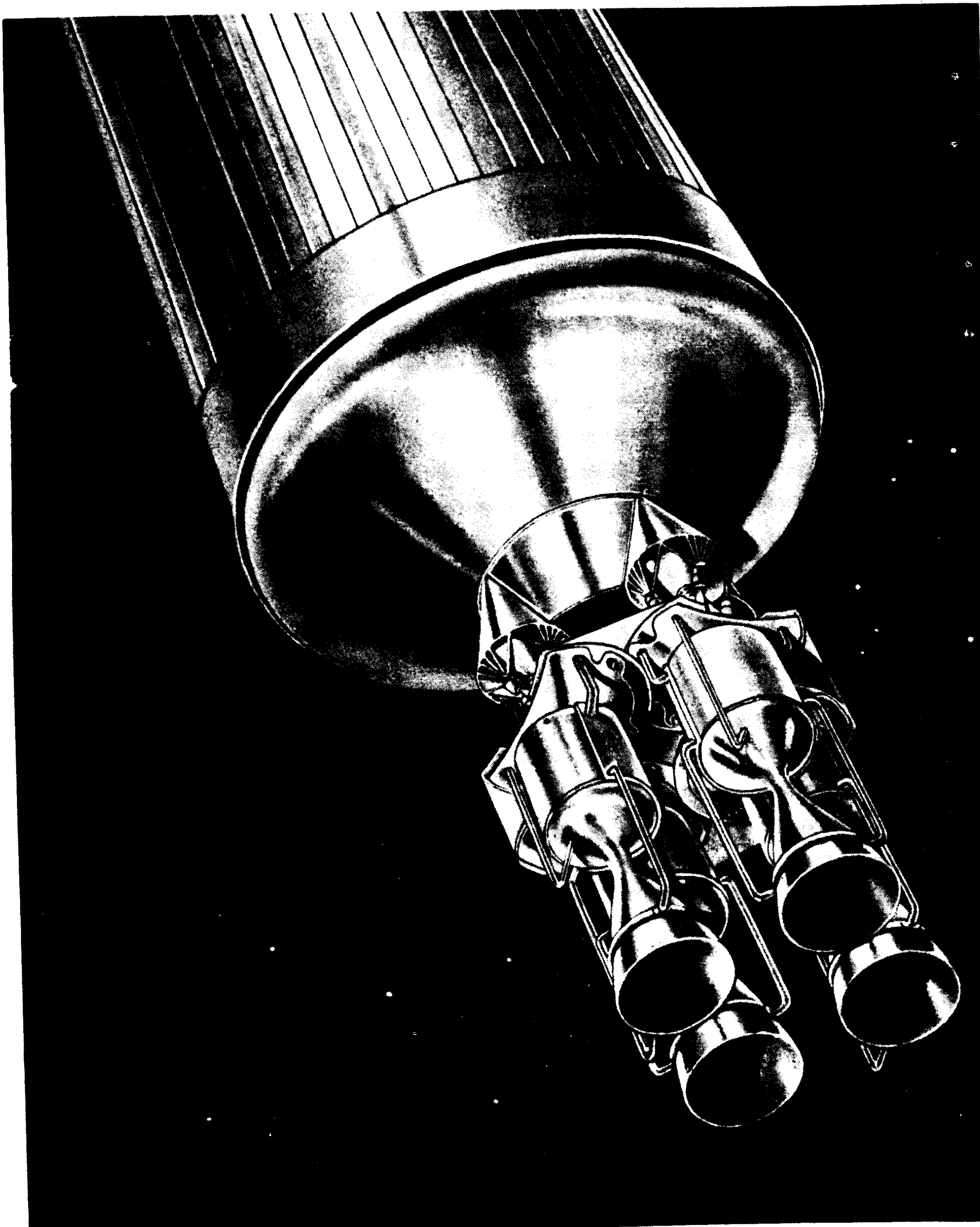


Figure IV-6

LIQUID HYDROGEN
FEED SYSTEM FOR
HIGH-POWER
REACTOR TESTS





PLUM BROOK REACTOR FACILITY

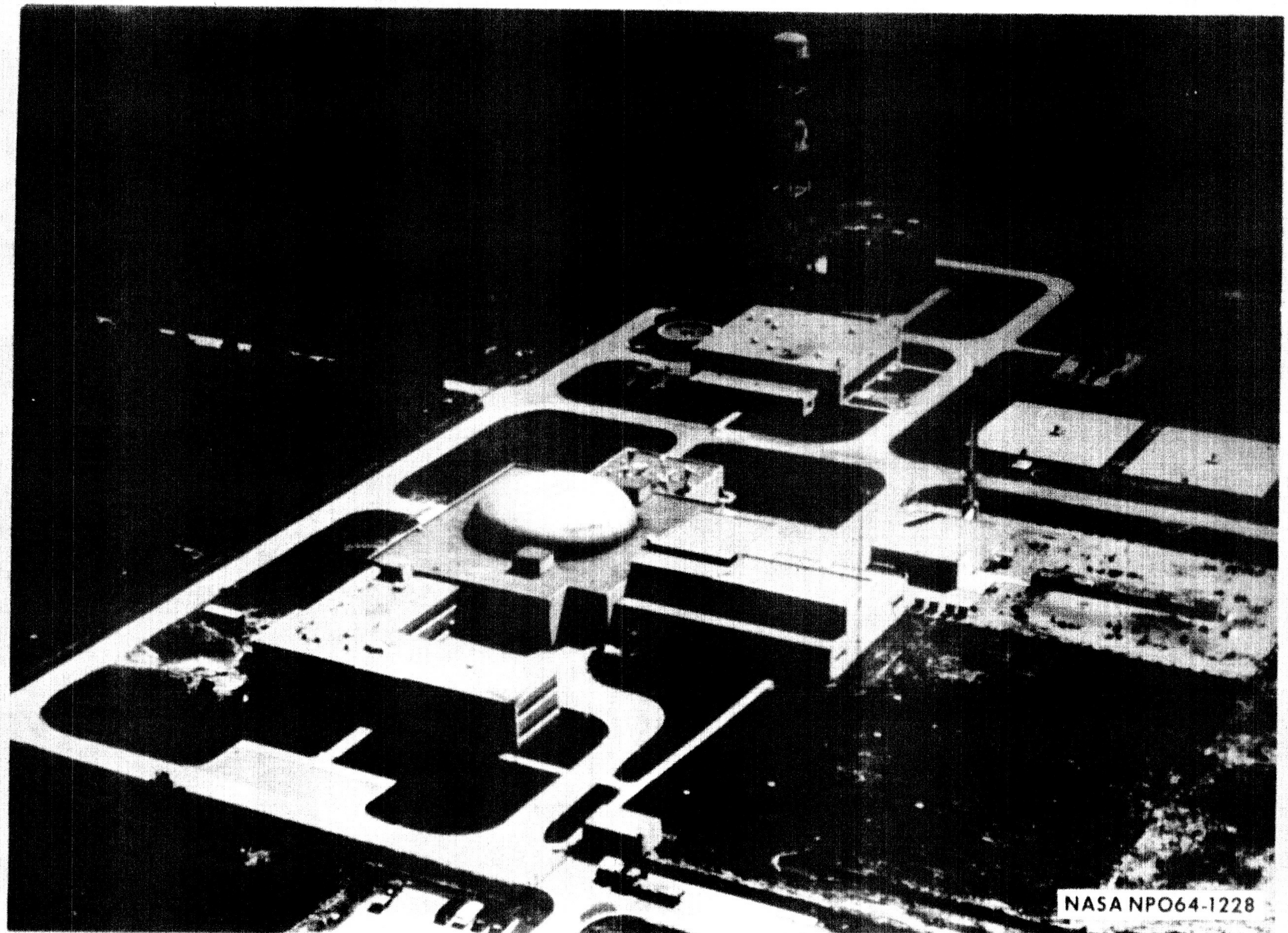


Figure IV-9

MAJOR RADIATION EFFECTS TESTING FACILITIES IN THE UNITED STATES

FACILITY	FAST FLUX (N/CM^2-SEC)	THERMAL FLUX (N/CM^2-SEC)	REMARKS
MTR	2.6×10^{14}	3.8×10^{14}	HORIZONTAL AND VERTICAL TEST HOLES; MAXIMUM TEST HOLE SIZE APPROX. 4 IN. ID
ETR (IDAHO)	3×10^{14}	4×10^{14}	VERTICAL TEST HOLES; MAXIMUM TEST HOLE SIZE APPROX. 9 IN. ID
ATR (IDAHO)	4×10^{14}	$4\text{-}5 \times 10^{14}$	VERTICAL TEST HOLES; MAXIMUM TEST HOLE SIZE APPROX. 9 IN. ID
CTR (FORT WORTH)	3.6×10^{9}	1.5×10^{11}	EXPERIMENTS PLACED UPON MOVABLE PALLETS NO BASIC SIZE LIMITATIONS FOR EXPERIMENTS; GAMMA HEATING RATE OF APPROX. 6 WATTS/LB MAXIMUM DOES NOT MEET SPEC. FOR SOME CONTROL COMPONENTS.
ASTR (FORT WORTH)	1×10^{10}	5×10^{11}	EXPERIMENTS MOVABLE; HYDROGEN MAY BE USED FOR EXPERIMENT COOLANT; NORMALLY ONLY ONE EXPERIMENT RUN AT A TIME.
PBRF	$1\text{-}4 \times 10^{14}$	$2\text{-}9 \times 10^{14}$	HORIZONTAL AND VERTICAL TEST HOLES; MAXIMUM TEST HOLE SIZE 11.75 IN. ID; EXPERIMENTS INSERTED AND REMOVED WHILE REACTOR OPERATING.

Figure IV-10

PNEUMATIC ACTUATOR TEST CAPSULE

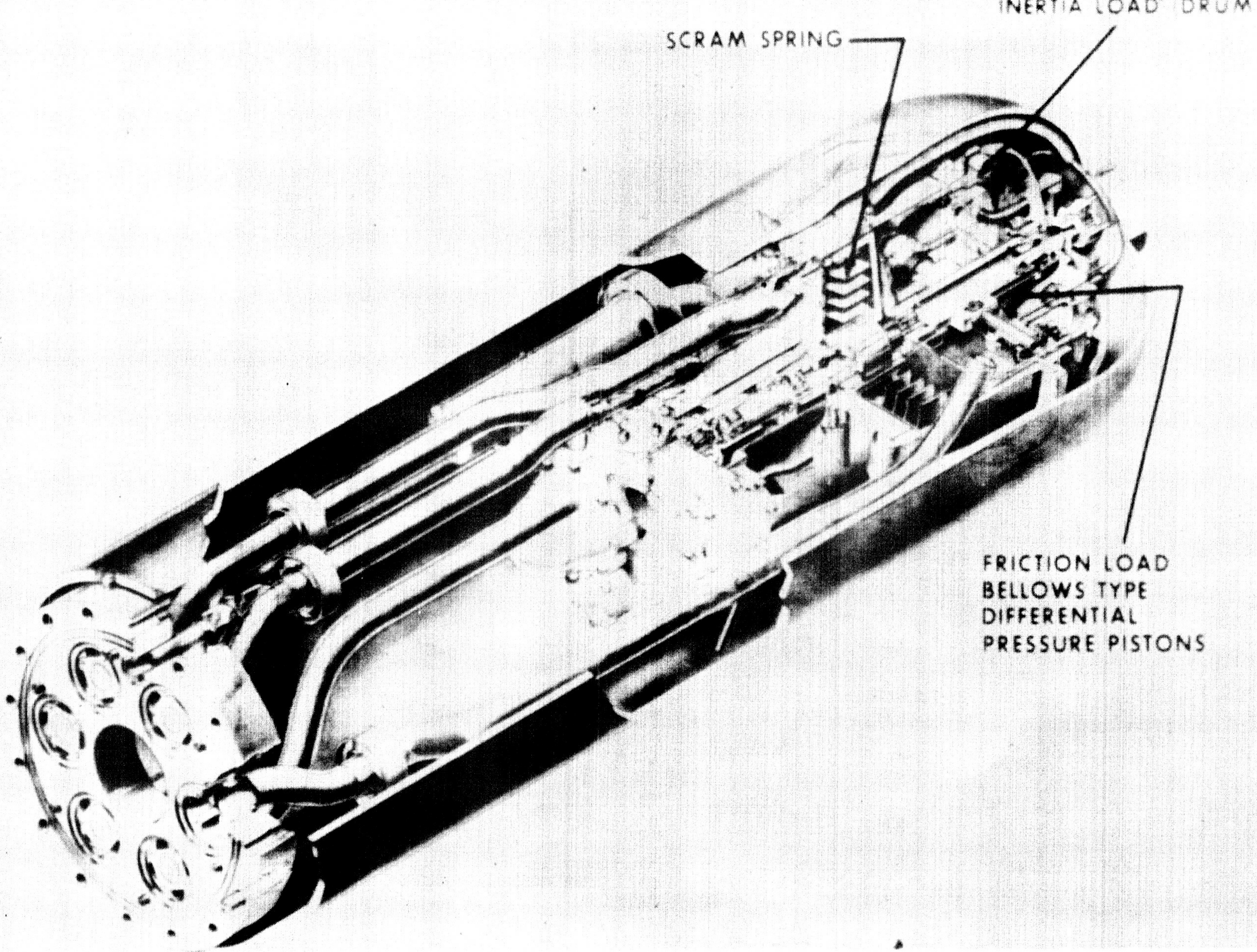
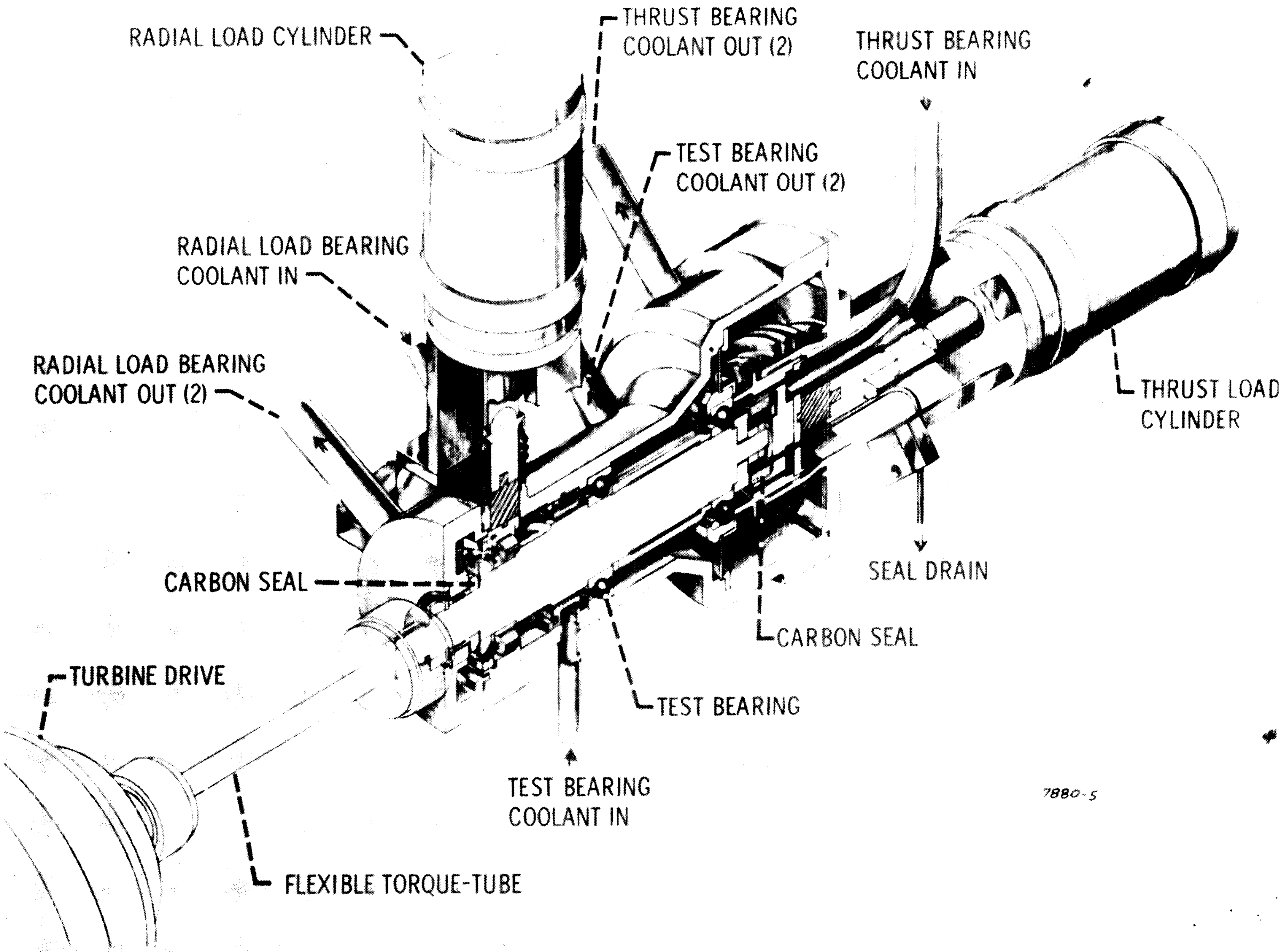


Figure IV-11



PRE-PILE BEARING TEST RIG

Figure IV-12

(a) Wall heat flux, 3.49 Btu per square inch per hour.

(b) Wall heat flux, 7.58 Btu per square inch per hour.

(c) Wall heat flux, 15.85 Btu per square inch per hour.

Figure IV-13

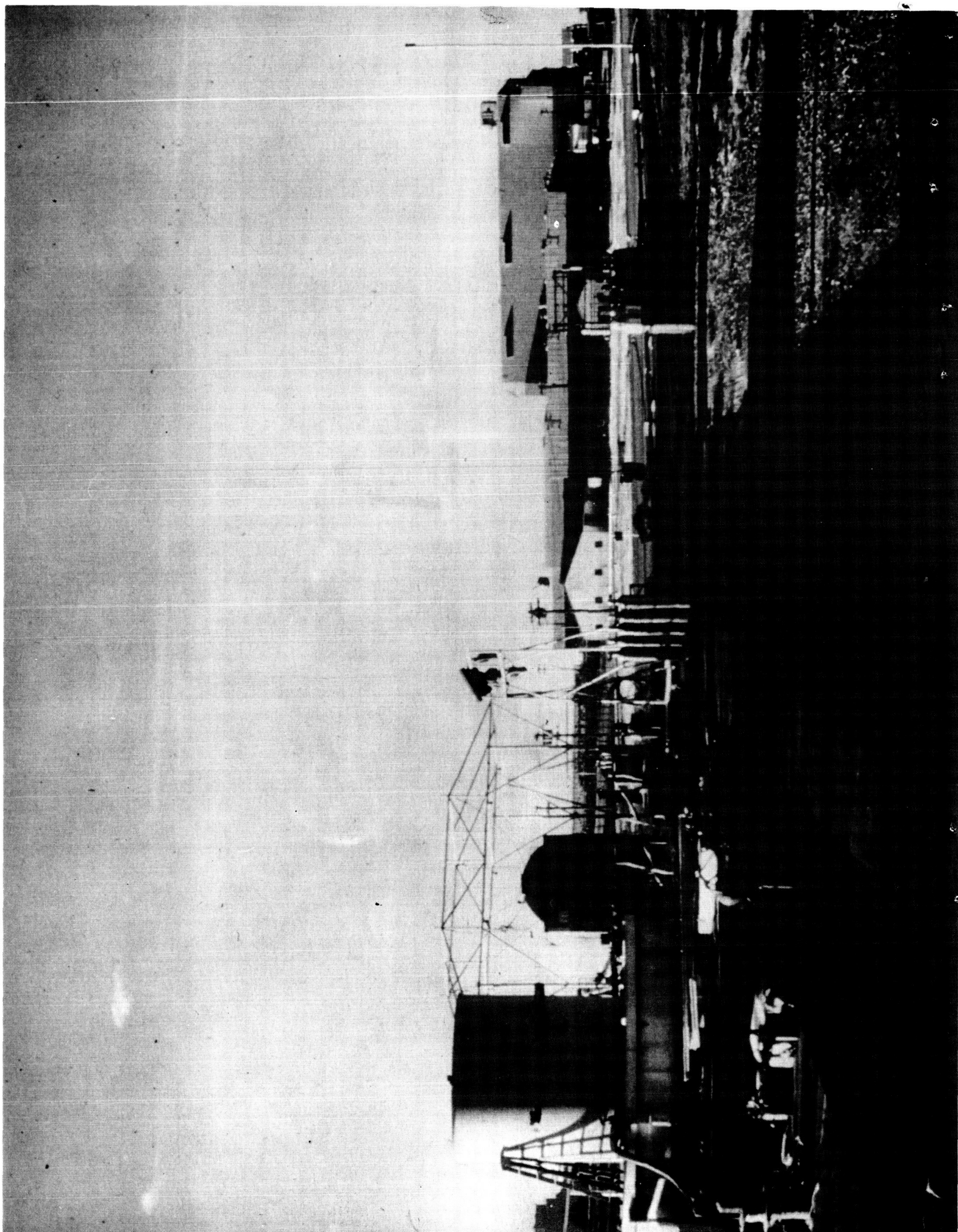


Figure IV-14

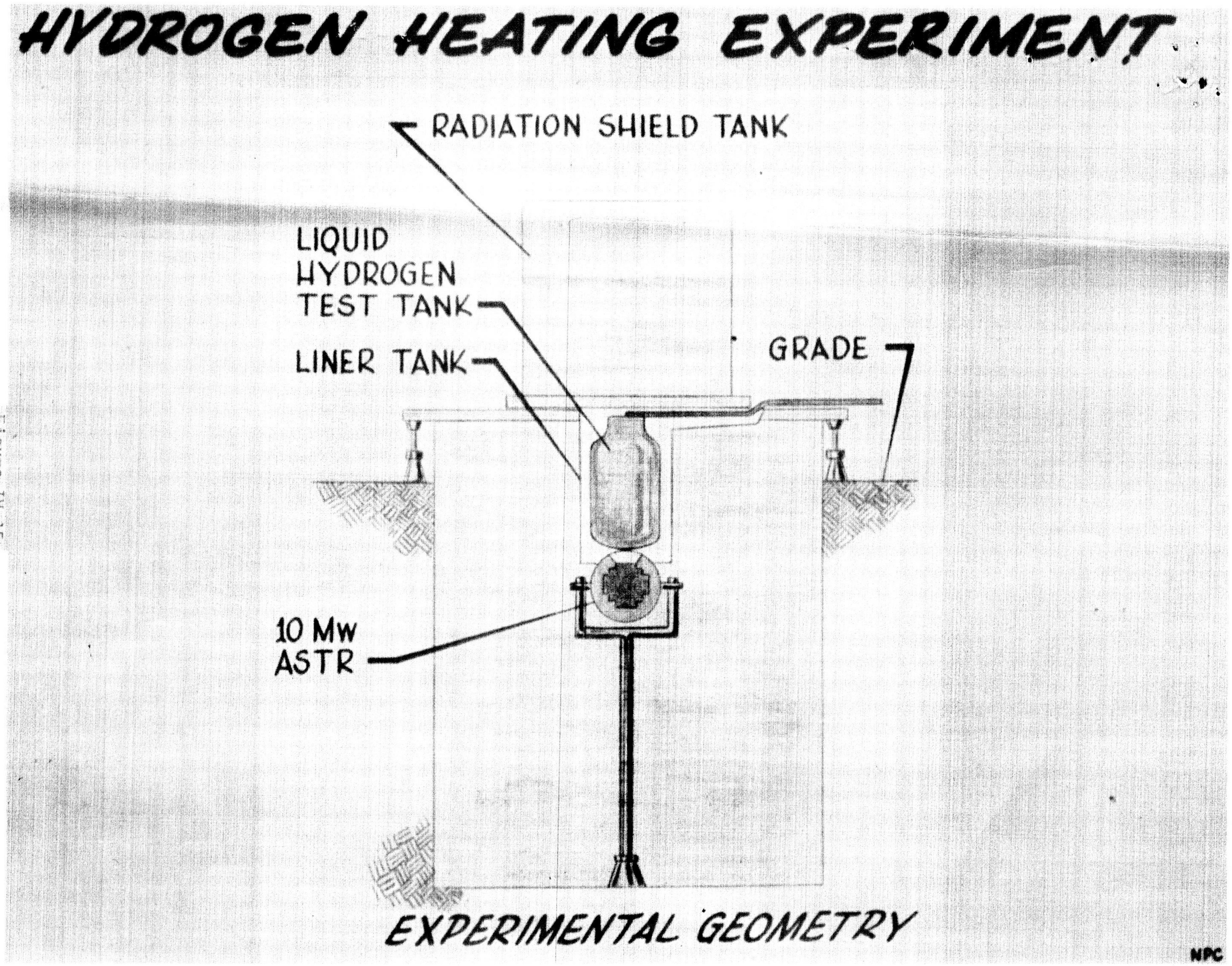

HYDROGEN HEATING EXPERIMENT
RADIATION SHIELD TANK
LIQUID HYDROGEN TEST TANK
LINER TANK
GRADE
10 Mw ASTR
EXPERIMENTAL GEOMETRY
NPC

Figure IV-15

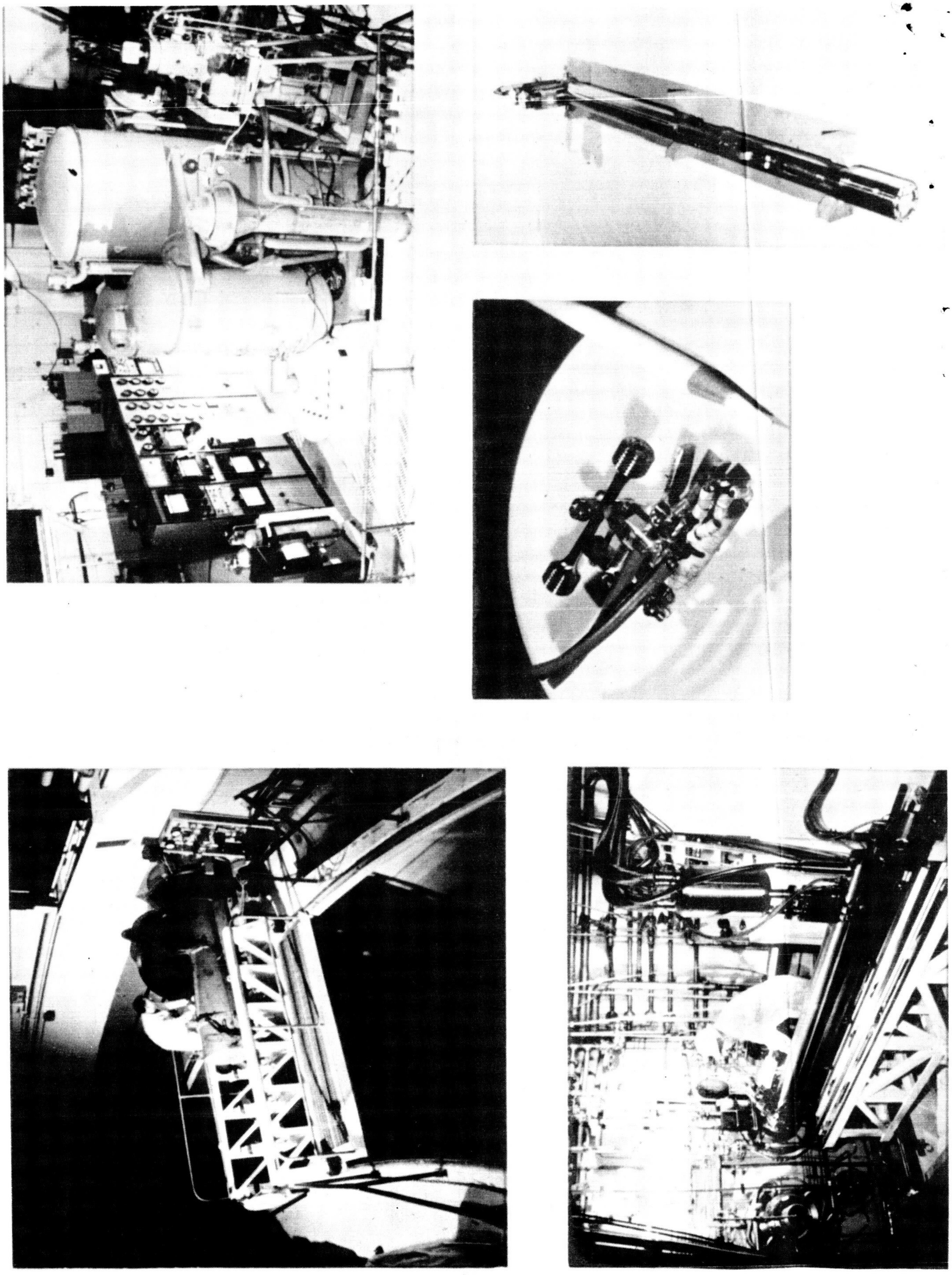


Figure IV-16

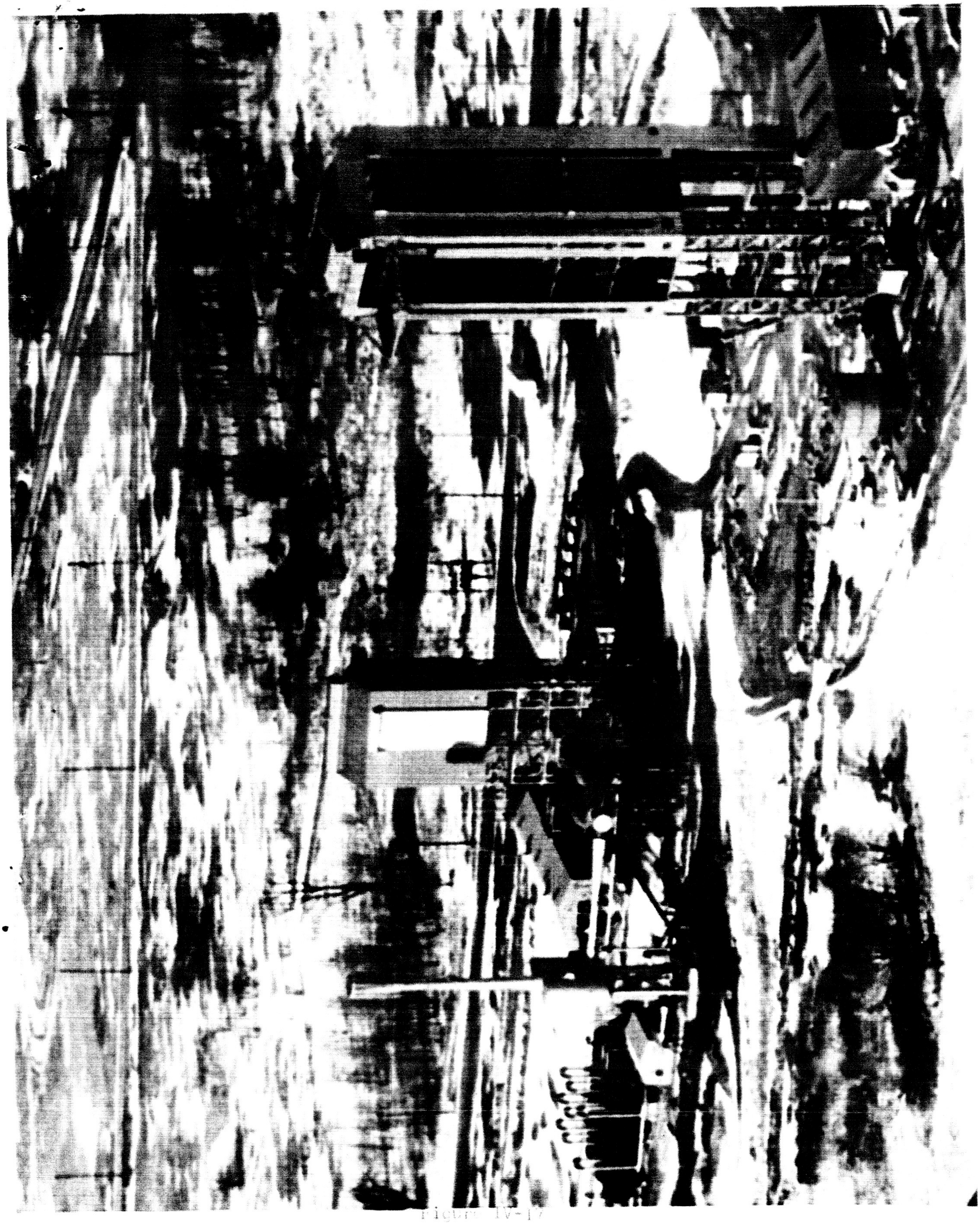


Figure IV-17

NUCLEAR ROCKET SYSTEMS SIMULATION EXPERIMENT

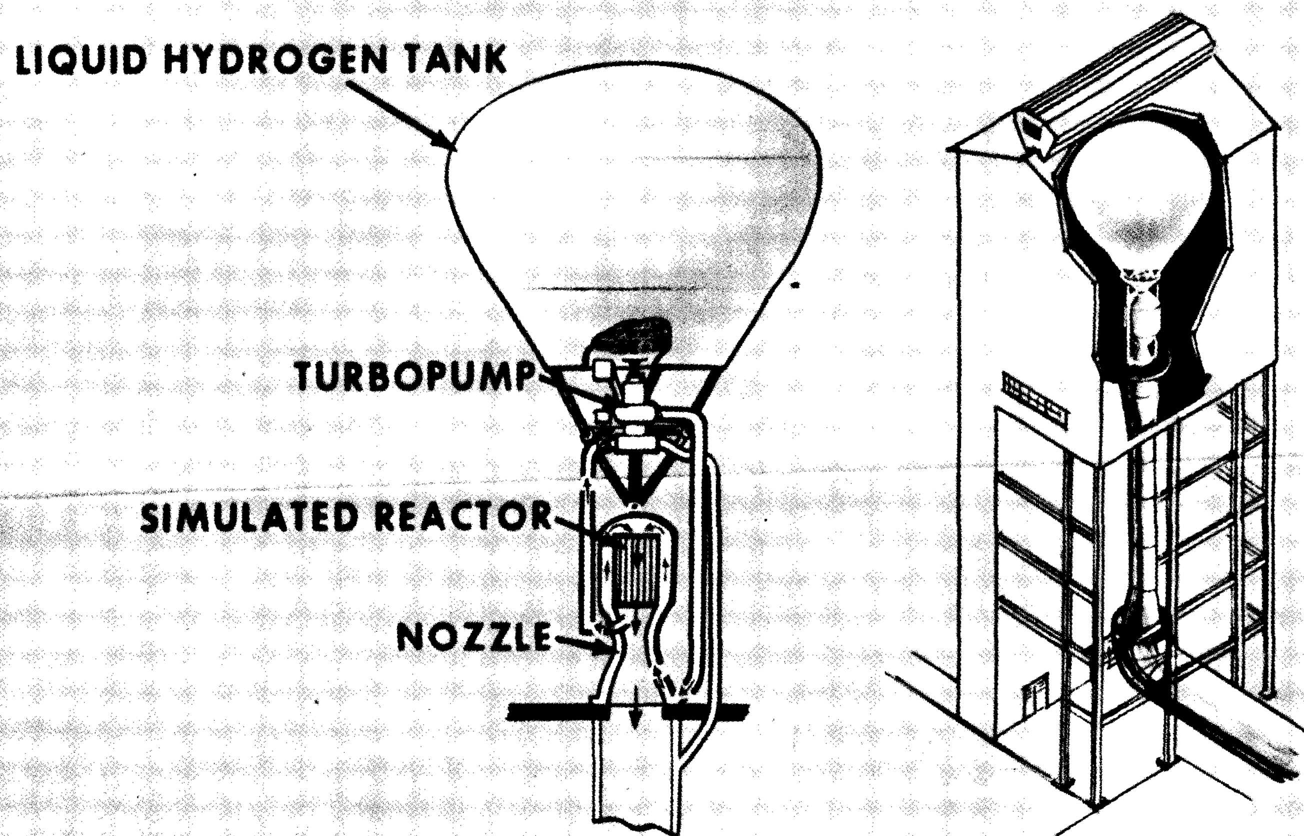


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Figure IV-18

NRX-A1 REACTOR

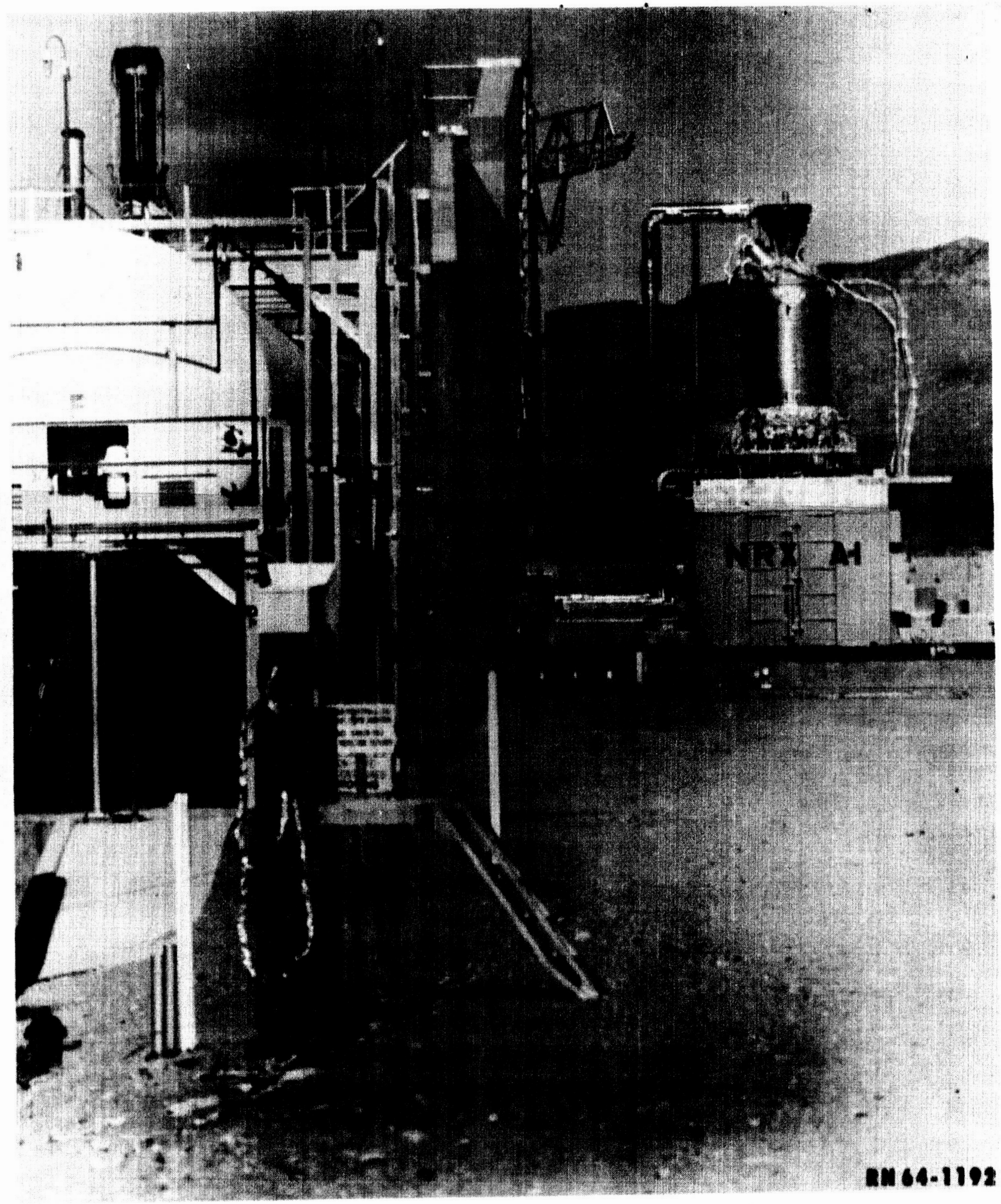

Figure III-15

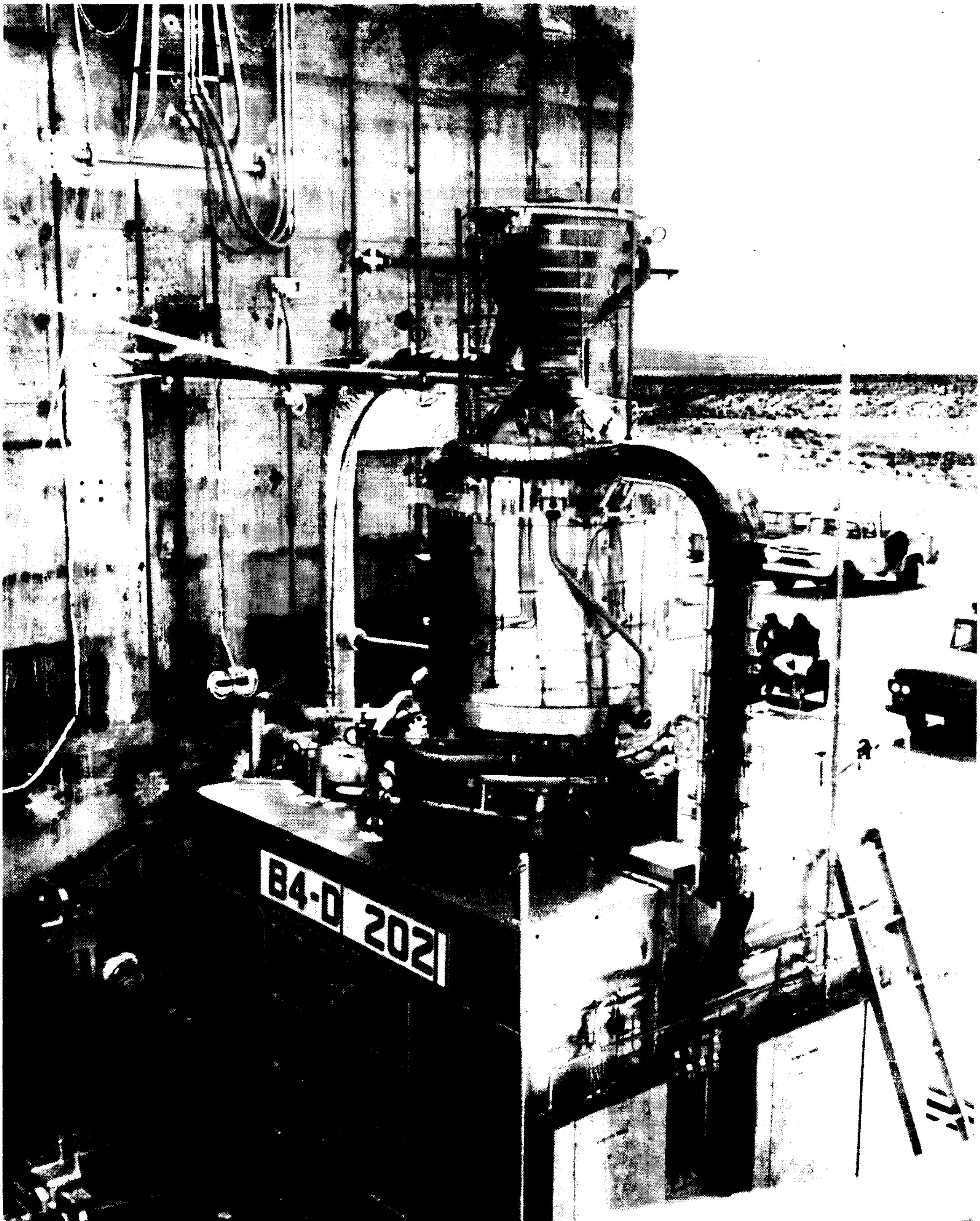

Figure III-16