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NUCLEAR PULSE SPACE VEHICLE STUDY

VOL. III - CONCEPTUAL VEHICLE DESIGNS AND OPERATIONAL SYSTEMS (U)

George C. Marshall Space Flight Center
Future Projects Office
National Aeronautics and Space Administration
Huntsville, Alabama

Contract NAS 8-11053

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Downgrading and Declassification

GENERAL ATOMIC

DIVISION OF

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JOHN JAY HOPKINS LABORATORY FOR PURE AND APPLIED SCIENCE

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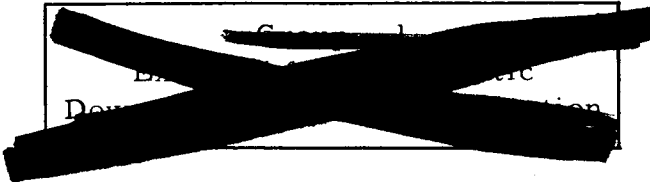National Aeronautics and Space Administration
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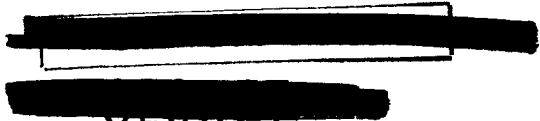Contract NAS 8-11053

Issued September 19, 1964

FOREWORD

The technical report on the Nuclear Pulse Space Vehicle Study performed under National Aeronautics and Space Administration Contract NAS8-11053 consists of four volumes:

Vol. I.   Summary Report (Secret);

Vol. II. Vehicle Systems Performance and Costs (Secret);

Vol. III. Conceptual Vehicle Designs and Operational Systems (Secret/Restricted Data);

Vol. IV. Mission Velocity Requirements and System Comparisons (Unclassified), prepared by General Dynamics/ Astronautics.

In addition to the technical report, a condensed summary of the study has been published as General Atomic Report GA-4891 (Secret).

The work reported in the present volume (Vol. III) was performed primarily by P. R. Shipps, Study Project Engineer, H. H. Amtmann, E. A. Day, C. V. David, T. Macken, W. E. Mooney, K. D. Pyatt, P. H. Sager, G. W. Stuart, T. Teichmann, M. Treshow, D. C. Weiss, and N. F. Wikner of General Atomic, and by B. Brown of General Dynamics/Astronautics and G. L. Getline of General Dynamics/Convair. The work was performed under the project direction of J. C. Nance, Project Manager, Nuclear Pulse Propulsion Project.

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CONTENTS

Figures

Tables

1. INTRODUCTION

The Nuclear Pulse Space Vehicle Study was divided into two phases: a general parametric phase during the first three months of the study and a specific-conceptual-systems phase during the last four months. Three major tasks were performed during the first study phase: the propulsion module (engine) characteristics were established and presented parametrically; performance data and approximate costs were determined for a broad range of mission velocities and payloads; and the operational characteristics and problems expected of nuclear-pulse-propelled vehicles were investigated.

During the second phase of the study, two specific conceptual designs of the propulsion modules, together with various over-all vehicle configurations appropriate to the variety of missions considered, were investigated. Considerable effort was also spent on determining performance, costs, operational characteristics, etc., of the specific conceptual vehicle designs. In addition, the determination of mission velocity requirements was performed during both study phases.

The more significant over-all results of the study--mission performance capability and mission costs--are reported in Vol. II. The work on mission velocity requirements, along with system comparisons between nuclear-pulse vehicles and other advanced space-propulsion systems, is reported in Vol. IV.

The present volume contains, for the most part, somewhat more detailed information which was either generated during this study or obtained from past or parallel study efforts. This information includes the complete-conceptual-vehicle design and weight, performance, and costing data which were necessary to determine the system performance and costs presented in Vol. II, as well as information on operations, ground support, and development planning. Owing to some of the detail provided on nuclear pulse unit designs and interactions, the present volume carries a higher classification, Secret/Restricted Data.

2. NUCLEAR-PULSE-PROPULSION-MODULE CHARACTERISTICS

The nuclear-pulse-propulsion concept considered in this study utilizes a series of nonisotropic nuclear explosions to impart periodic impulses to propel a space vehicle. The nuclear-pulse-propulsion concept can be briefly described as follows: A large number of nuclear devices (nuclear-pulse units), which are primarily carried in the propulsion module, are ejected successively to a specified point below the vehicle, where they are exploded. The pulse units are designed in such a manner that when they are exploded a large fraction of their mass is propelled toward a heavy circular disc (the pusher) which forms the base of the vehicle. Interaction of the high-velocity propellant with the pusher drives the pusher upward, and the shock-absorber system attached to the pusher transfers the momentum to the upper sections of the vehicle at accelerations tolerable to sensitive payloads.

Research on this propulsion-system concept has been carried out by General Atomic over the past six years, primarily under the auspices of the U.S. Air Force Special Weapons Center. It is from this theoretical and experimental background that the present level of confidence in the feasibility and potential performance of this propulsion system has been derived. Many different technical approaches have been studied, and a wide range of vehicle sizes have been considered and analyzed, with gross weights ranging from 200 to 10,000 tons and specific impulses (I_{sp}) varying from 1,600 to over 5,000 sec.

A major effort made in the past year (1963) under Air Force contract was a composite physics and engineering analysis of a 4,000-ton-gross-weight reference vehicle design. Parallel with the activities in the design analysis was the development of two-dimensional computer codes to determine pulse-unit characteristics, particularly in connection with a detailed analysis of the focusing and expansion of the propellant and their effects on velocity distribution and pressure on the pusher. Computer programs were also developed to study and optimize pusher shaping to hold compatible stress levels throughout the pusher under the varying loading conditions precipitated by errors in pulse-unit stand-off distance, axial deployment, and angular positioning.

The theoretical work was supported by an extensive experimental physics program to simulate and analyze the ablation phenomena associated

2

with the high-velocity plasma-pusher interaction and to develop techniques to measure ablation.

Finally, an experimental response program was carried on to simulate the pusher impulse and response characteristics in a regime almost identical to those to be experienced from nuclear explosions. Research on working samples of the pusher plate and of the toroidal shock-absorber system incorporating its attachments to the pusher has been performed by simulating the nuclear explosions with high explosives.

These programs have provided considerable insight into the nature of the physical phenomena involved in nuclear-pulse-propulsion systems and has led to the development of practical and acceptable solutions to the engineering problems presented in such propulsion systems.

The results of these integrated studies and experiments have been reported in "Technical Summary Report--Nuclear Propulsion Project (Project ORION)," Air Force Report RTD TDR 63-3006, Vols. I through IV.

2. 1. MODULE DESCRIPTION

The configuration of a typical nuclear-pulse-propelled vehicle with the propulsion module assembled with the payload section is shown in Fig. 2. 1. The payload section is considered to be a separate unit, since its size, weight, and composition are dependent on particular mission payload and personnel accommodations requirements (see Sec. 3), and hence is not included in this description of the propulsion system.

The design and functioning of the individual components and systems which make up the complete propulsion module are not presented in detail in this report, since this study was primarily concerned with the application of the system; most of the detail is presented in Vol. I of RTD-TDR-63-3006. A brief description of the components and their functions is given here, however, to provide a general understanding of the total propulsion-system operation.

The system can be considered as a three-mass, nonlinear, two-spring system. The three masses are the pusher, the intermediate platform (between the two shock-absorber stages), and the upper vehicle section, including the propulsion module body, the pulse-unit magazine section, and the payload. The two springs are the first-stage toroidal shock absorber and the second-stage piston-type shock absorbers.

INTERMEDIATE PLATFORM

1st STAGE SHOCK ABS.

PUSHER

2nd STAGE SHOCK ABS.

UPPER MODULE SECTION (BODY)

PAYLOAD SECTION

PROPELLANT MAGAZINES

PROPULSION MODULE
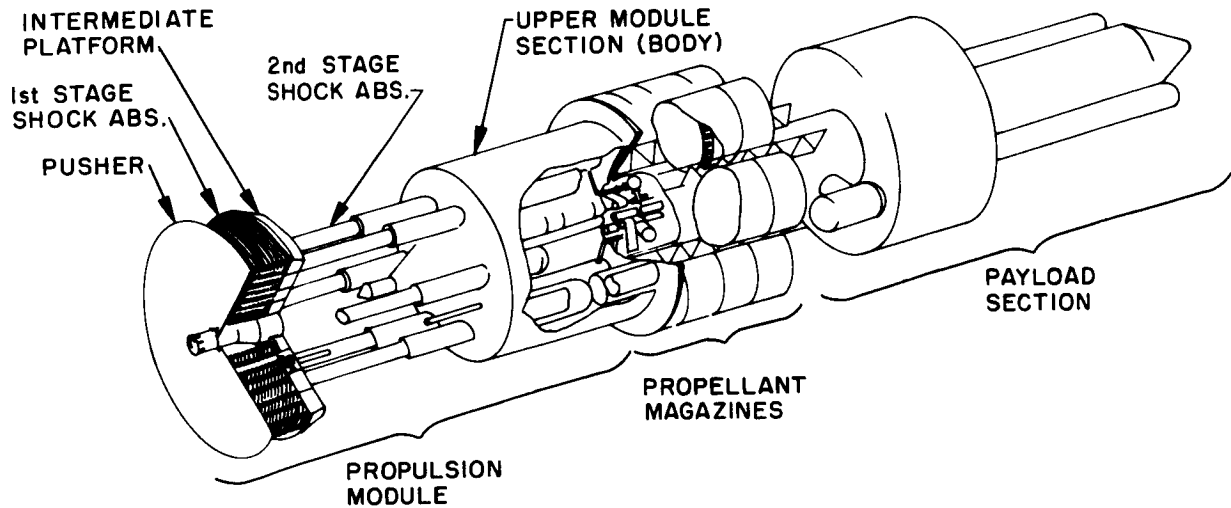
Fig. 2.1--Typical nuclear-pulse vehicle

2.1.1. Pusher Plate

The pusher plate is a relatively thin steel (USS T-1 or similar) plate designed to intercept the propellant and transfer its momentum through the shock-absorber system to the upper vehicle section. The pusher-plate design developed in an earlier design study is shown in Fig. 2.2.

The pusher plate is designed with a varying thickness to provide a mass distribution which matches the radial distribution of the propellant impulse. This matching is required to attain uniform acceleration over the plate area and thereby minimize bending stresses.

A hole is provided in the center of the pusher to permit penetration by the first-stage shock-absorber protection tube. This tube also serves to distribute an antiablation oil over the plate between pulses.

The hot propellant interacting with the pusher would cause extensive ablation of the lower pusher surface if some means of protection were not provided. The results of experimental investigation of the ablation problem have resulted in a method for preventing such ablation. For the normal pressure, velocity, density, etc., of the propellant interacting on the pusher, a layer of carbonaceous material such as oil (~6 mil thick) is applied to the bottom surface of the pusher between pulses. Since the oil must be applied as a continuous coating, spray nozzles built into the walls of the central protective tube extending through the pusher are arranged to direct the spray upward to cover the complete surface of the pusher. The main oil supply is stored in

4

(19) CONCENTRIC GROOVES FOR
FIRST-STAGE TORI ATTACHMENT

B B

86' DIA

3.75'

12" DIA

NEUTRAL
POSITION

$\frac{3}{16}$R

1"

$\frac{3}{4}$

SECTION B-B
FIRST-STAGE SHOCK-ABSORBER
TYPICAL ATTACHMENT
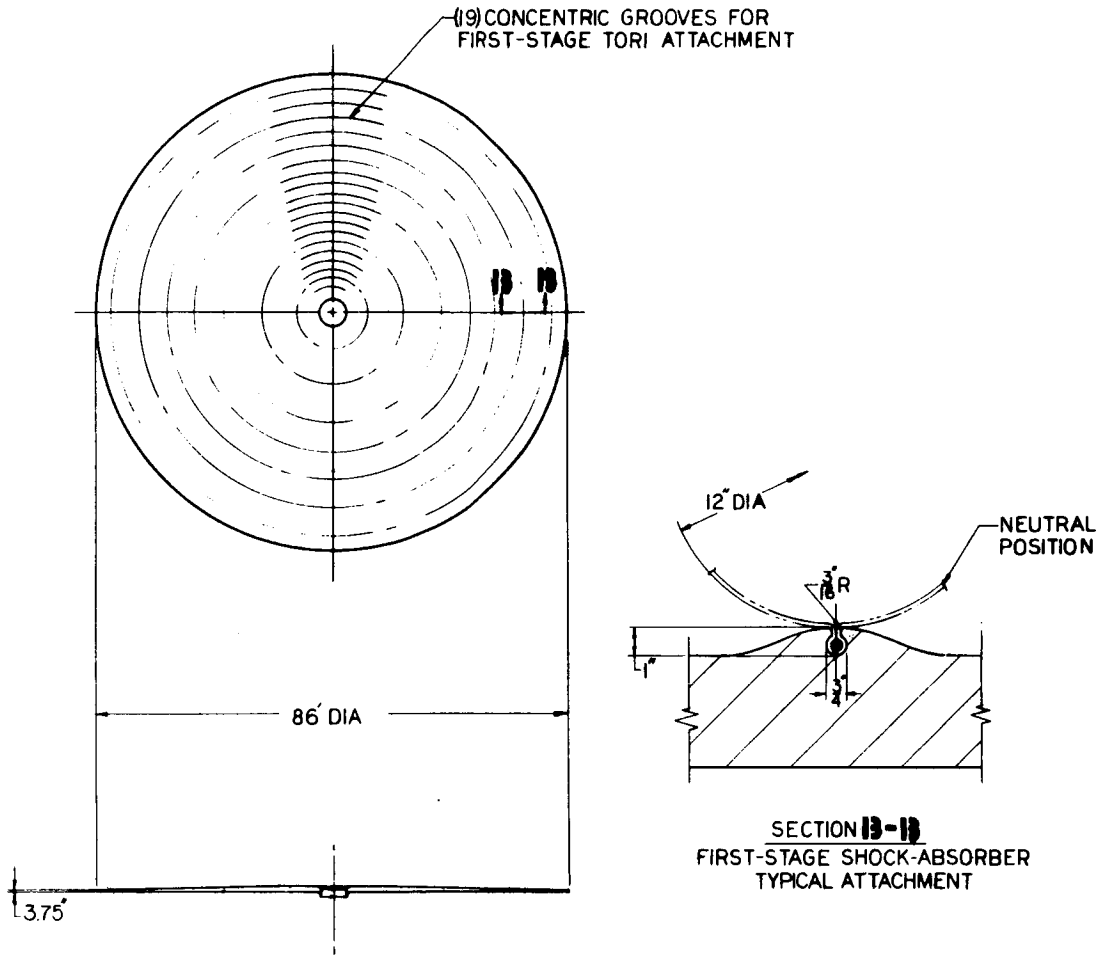
Fig. 2.2--Design of a large pusher plate

the body section of the propulsion module. Pipelines and automatic valving connect the oil supply to the channels in the walls of the shock-absorber protective tube.

Due to the extremely high accelerations (on the order of 50,000 g) which the plate experiences, special provisions must be made for attaching the first-stage shock absorber to the plate. The type of attachment used is a function of shock-absorber design, material, and characteristics and of the characteristics of the impulse load. One attachment design which has proven satisfactory is shown in Fig. 2.2.

2.1.2. Shock-absorber System

The shock-absorber system transfers the momentum of the pusher to the upper vehicle section in such a way that the peak accelerations are within the tolerable limits of module, payload, and crew. The first stage of the shock-absorber system may be considered analogous to a tire on an airplane landing gear; it reduces the peak acceleration from the impulse on the pusher to values which are compatible with the relatively rigid structures of the second-stage shock absorber, which, like a landing-gear strut, further reduces acceleration to a few g's.

The first-stage shock absorber consists of a series of concentric, multilayer, gas-filled tori which connect the pusher to an intermediate platform between the two shock-absorber stages. The gas pressure in the tori when the pusher is in its neutral position is approximately 100 psi. When the tori are fully compressed or extended, the first-stage shock absorber travels approximately 0.4 times its height in the neutral position. The double-acting second-stage shock absorbers connect the intermediate platform to the body section of the module and are arranged either in a single concentric row for the smaller-size modules or in double concentric rows for the larger sizes. In a typical operation, the first-stage shock-absorber system oscillates through 4-1/2 cycles between pulses, whereas the second-stage shock absorber oscillates through one half-cycle (Fig. 2.3.). There is some energy dissipation in both stages, although the shock-absorber system as a whole is considered to be nondissipative. The entire shock-absorber system is synchronized with the pulse-unit firing, but it is double-acting in that provision is made to decelerate it and return it to its neutral position should it travel beyond that position owing to failure of a pulse unit to explode. The average vehicle acceleration is a function of the loading condition of the vehicle, while the relationship between average and peak accelerations is a function of the shock-absorber design characteristics. A typical acceleration curve for Mode I operation at start-up is shown in Fig. 2.4. The solid line is the acceleration profile
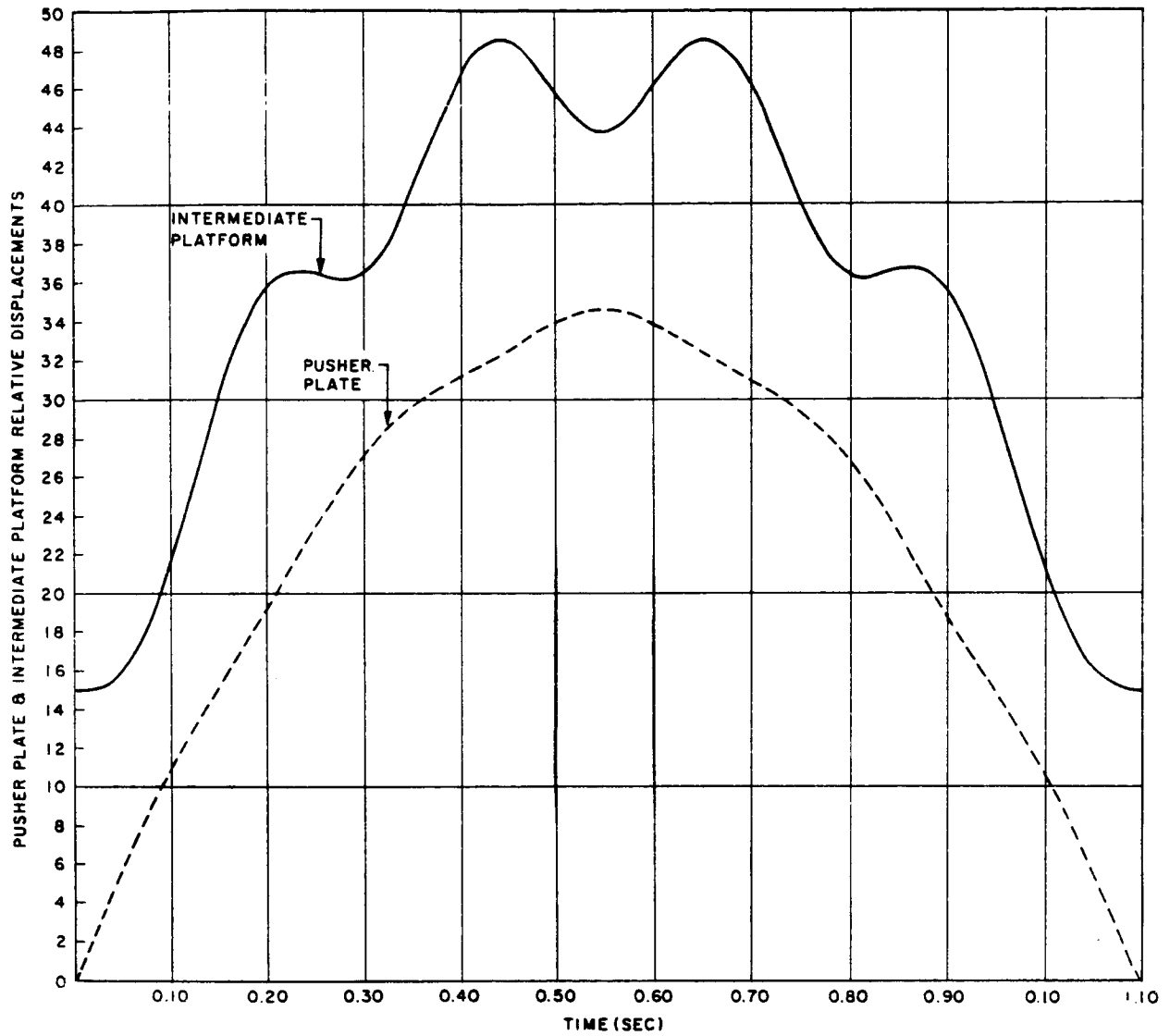
Fig. 2. 3--Pusher and intermediate platform displacement
with respect to the vehicle

for start-up at an effective thrust-to-weight ratio (F_e/W_o) of 1.25. The dotted curve shows a typical maximum average design acceleration of $F_e/W_o = 3.1$, which would represent the maximum acceleration conditions prevailing at the end of a mission with zero payload and light vehicle structure. The average acceleration will normally be well below 3.1 g, the degree depending on specific mission requirements. It will be noted that negative accelerations are experienced during the shock-absorber damping periods (15 to 20) after shutdown and in the event of a pulse-unit misfire.
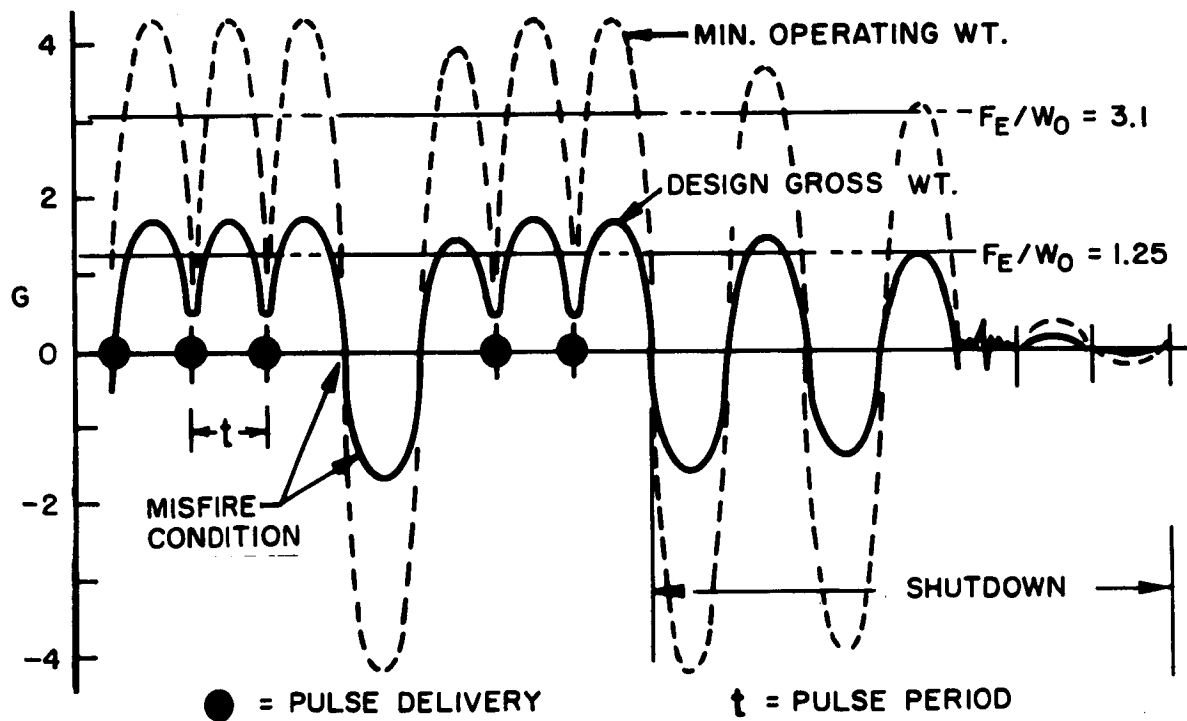
Fig. 2.4--Typical nuclear-pulse-vehicle acceleration profile

For a Mode III operation (orbital start-up), in which the module is boosted into orbit and loaded with the desired mass for the mission, typical values of start-up average accelerations are approximately 0.3 to 0.5 g, while final average accelerations are approximately 3.0 g or less.

2.1.3. Body Section

The body section of the propulsion module incorporates the basic structure of the upper part of the second-stage shock absorbers and accommodates the pulse-unit storage compartment and the pulse-unit loading and delivery system (see Fig. 2.5). The ejector-gas mixing tank, admission valve, piping, and other associated equipment are also located in the body section. The lower conical section contains storage tanks for gases, coolants, and antiablation oil. A deflector cone protects the end and the interior of the ejection tube from the plasma that expands through the hole in the center of the pusher. The lower conical structure incorporates a protective shield against the propellant plasma deflected by this cone.

The pulse units are accommodated on several floors (depending upon the number of units required). The pulse units are designed to be supported on tracks in a series of conveyor channels from which the units are progressively fed by the conveyor system to the loading mechanism surrounding the central delivery tube. The pulse-unit conveying and loading arrangement provides for two types of pulse units--a normal full-yield unit and a half-yield unit. The half-yield units are required for initial start-up and for restart after any interruption of the propulsion sequence when the pusher is in its neutral position with zero velocity. The loading mechanism aligns the pulse units with the ejector tube prior to expulsion.

2.1.4. The Pulse Unit

The energy for the propulsion of the nuclear-pulse vehicle is provided by the pulse unit; it converts the energy released by a nuclear explosion into a well-focused cloud of high-velocity propellant vapor. As shown in Fig. 2.6, the pulse unit consists of the propellant, channel filler, radiation case, nuclear explosive device, delivery case, and fusing and firing mechanism.

When the nuclear device is exploded, the channel filler absorbs the radiation emitted and rises to a high temperature. The radiation case serves to contain the energy released by the explosion so that more energy is absorbed by the channel filler than is emitted into the solid angle it subtends relative to the source. The high pressure achieved in the heated channel filler then drives a strong shock into the propellant, which vaporizes the propellant material and accelerates it toward the pusher.

The specific impulse of the propulsion module is directly proportional to the fraction of the mass of the pulse-unit propellant intercepted by the pusher. In order to maximize this fraction, it is necessary
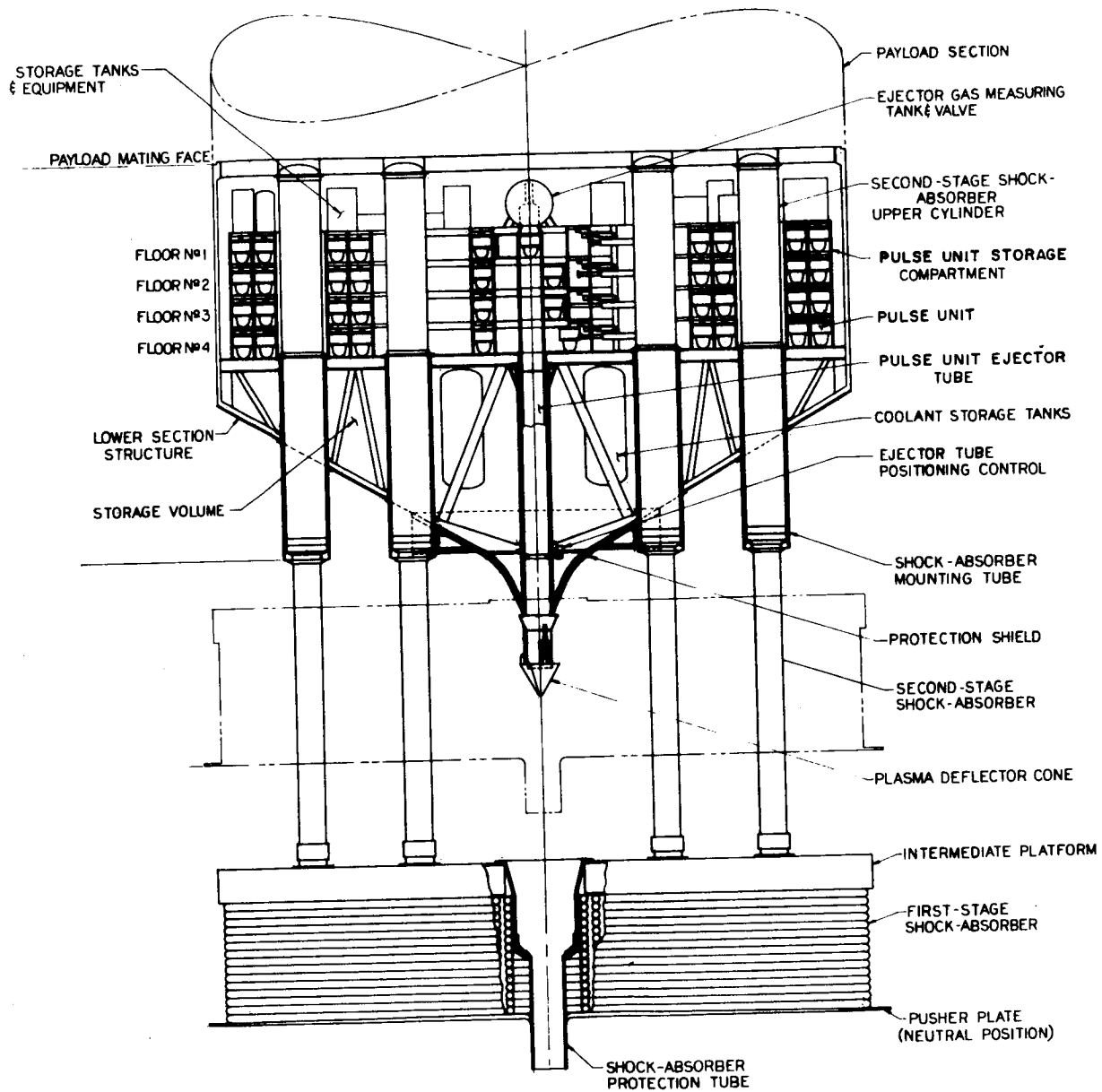
Fig. 2.5--Propulsion-module section layout

to control the manner in which the propellant expands and to ensure that
its mass forms a large fraction of the total mass. The degree to which
the expansion is controlled by shaping and the degree to which the
expanding vapor remains focused is proportional to the propellant
diameter-to-thickness ratio. This ratio should be large. But if the
diameter of the pulse unit is to be kept to a reasonable size, the pro-
pellant must be made from a high-density material. Furthermore, the
thickness and opacity of the channel filler should be such that the
radiation does not penetrate the propellant.

PROPELLANT

CHANNEL FILLER

RADIATION CASE

NUCLEAR DEVICE

DELIVERY CASE

FUSING & FIRING
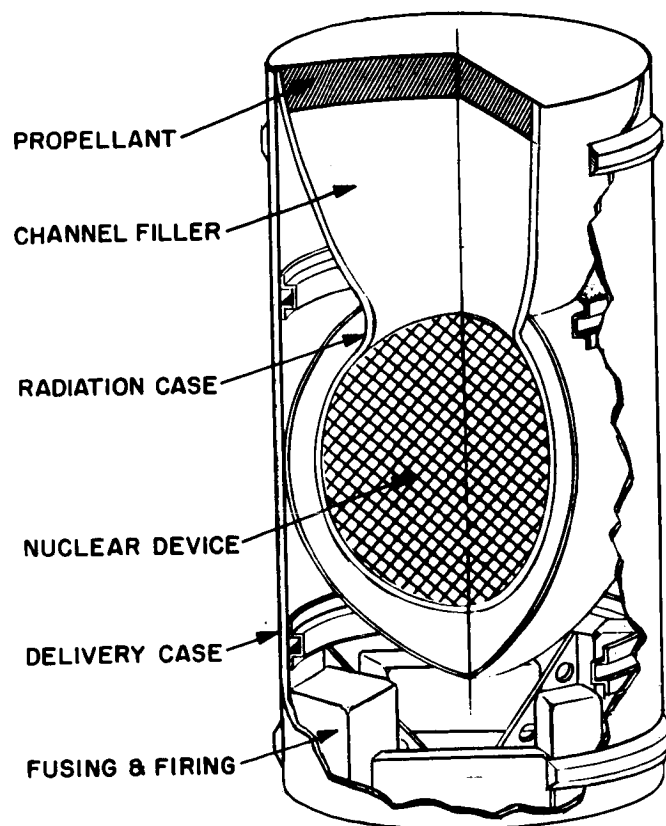
Fig. 2. 6--Pulse-unit design for a 10-m propulsion module

The propellant and channel filler also serve as a shield to pro-
tect the engine and upper vehicle from the neutron and high-energy
gamma radiation produced by the nuclear explosion. This sets a lower
limit on the thickness of the propellant and channel filler for a particular
design.

An analysis of these requirements has led to the choice of tungsten for the propellant material and beryllium oxide for the channel filler, with a tungsten—beryllium oxide mass ratio of approximately 4:1.

The radiation case around the nuclear device is required to be sufficiently optically thick to contain the radiation until the source has released all of its energy. For this purpose a high-opacity material such as uranium is the best choice. Limitations on the velocity to which it is desirable to accelerate the propellant are discussed in Vol. II of RTD TDR-63-3006.

Given the average velocity, the pulse-unit design problem is largely one of devising a system that will place the largest possible fraction of the propellant mass within the solid angle intercepted by the pusher. Distribution of the propellant mass within the intercept angle is also critical in that this determines not only the impulse per unit area as a function of the radius of the pusher, but also the peak pressure that will be applied to the pusher as a function of time.

2.1.5. External Propellant Magazines

The basic propulsion module of this study, as previously discussed, has an internal propellant storage capacity of 900 pulse units plus the associated fluids. These units are carried in "ready-to-fire" racks that transport the units to the delivery tube breech at the required rate of approximately one per second. These 900 units are more than enough for any sustained propulsion period desired of most missions, but not enough for the complete mission propulsion requirements. Hence, additional propellant is carried in expendable magazines which are so arranged that the pulse units are transported during coast periods down tubes from the "stacked" magazines to the racks within the propulsion module. The external "stacked" arrangement of the magazines can be seen in almost any of the over-all configurations of this report.

The expendable propellant magazines carry the pulse units only, not the fluids, and are coupled to the delivery tubes at installation. When emptied, they are expended during the next coast phase. Should a magazine not come free when it is to be expended, it may be retained at no penalty other than that due to its mass during subsequent maneuvers. The fluids that accompany the pulse units transported in the magazines are carried in tanks typically located in the payload spine (see exploration configurations) or in the propulsion module itself.

2.1.6. Control System

An integrated control system is required to ensure the proper synchronization of the pulse-unit firing sequence and mechanical operations of the engine components (Fig. 2.7). Since the large masses involved operate under conditions where the recycling period is in the range of 0.80 to 1.5 sec, the majority of propulsion-system components will be monitored and controlled through a computer system to ensure high reliability. A schematic diagram displaying the location of and inter-relationship between major components and systems is shown in Fig. 2.8.

The principal function of the control system is the sequencing of the pulse-unit ejection and detonation. Excess gas from the shock-absorber system, supplemented by a gas-generation system as required, is utilized to eject the pulse units. This high-pressure gas system, which is activated by a signal from the propulsion-system control computer, admits a measured quantity of gas into the ejection tube via an admission valve and propels the pulse unit through the ejector tube. After being ejected out the muzzle (at a velocity of ~450 ft/sec), the pulse unit passes through the first-stage shock-absorber protection tube, which extends through the center of the pusher, and travels on to the prescribed detonation point below the pusher. The pulse unit remains unarmed until it is within a specified distance of the detonation point, whereupon it is armed.

Detonation is initiated upon a signal from the engine-control computer only when all conditions of synchronization between the pusher, the shock-absorber system, and the pulse unit are satisfactory. The tolerances on these conditions are stringent, but the system is so arranged that there is no possibility of an inadvertent explosion of a pulse unit too close to the vehicle. If for some reason the pulse unit fails to detonate on signal, it is automatically disarmed and is subsequently destroyed by a separate signal when it has reached a point well out of range of the vehicle.

2.1.7. Flight Control

Various methods of flight control during propulsion have been considered in previous nuclear-pulse-propulsion studies. One method of directional control was by fine adjustments of the position or attitude of the pulse units at detonation, which requires no auxiliary propellant. In this study, however, a chemical-rocket, lateral-thrust attitude-control system is used. Its size is relatively small, since the basic propulsion system, for most over-all configurations at least, is stable and partially self-correcting for pulse misalignments that are within tolerances.
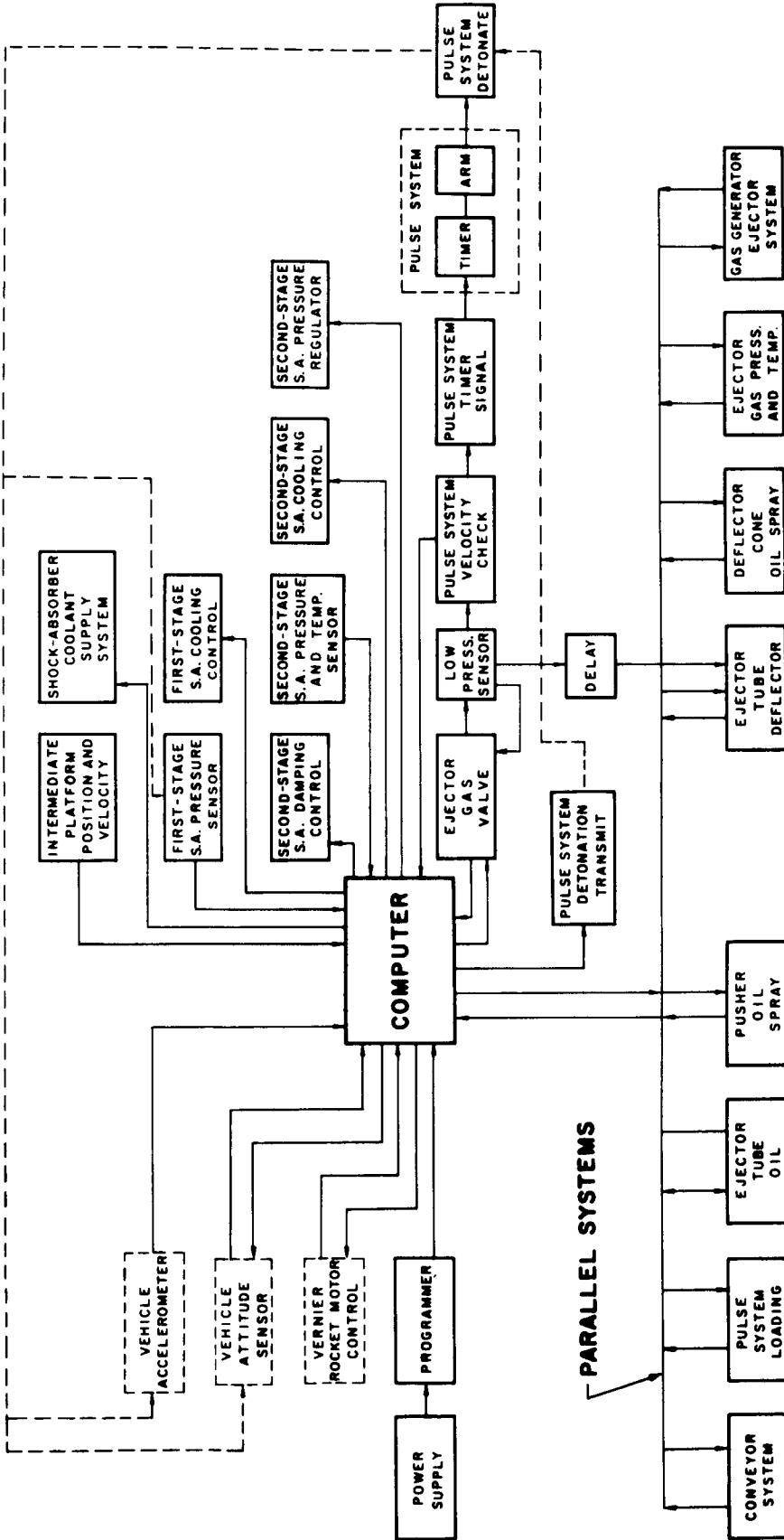
Fig. 2.7--Propulsion-system-control block diagram
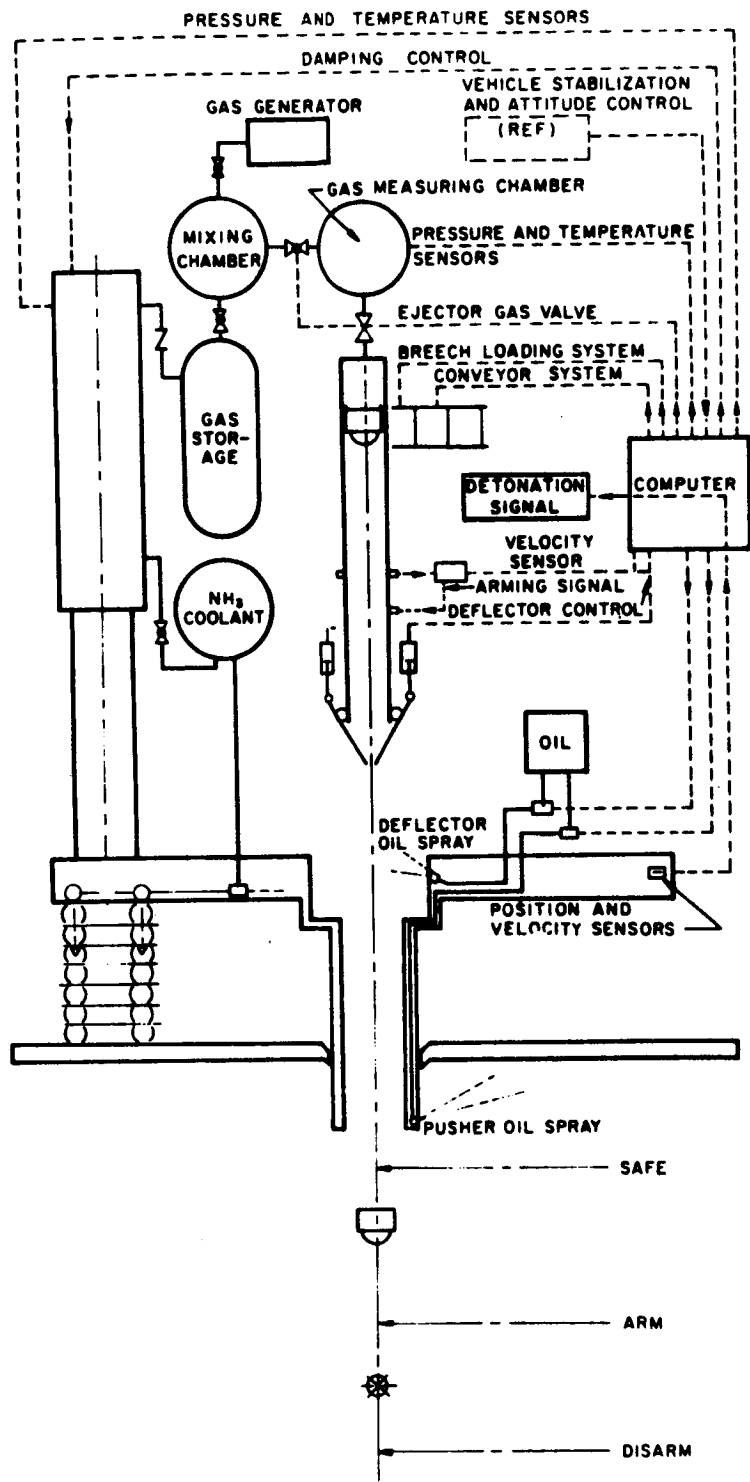
Fig. 2.8--Principal elements of propulsion system control

15

Although this system is located in the payload section of the systems considered in this study to attain a long moment-arm, its description is included with the propulsion-module description because its function is so closely related to the operation of the propulsion module.

Preliminary estimates have been made of the amount of attitude correction required for various sizes of propulsion modules and over-all configurations. The four rocket motors used in the flight-control system are positioned near the nose of the over-all vehicle; they are 90 degrees apart and thrust slightly aft of normal to the longitudinal axis. Simple "bang-bang" motor control is assumed, with the input suitably damped to take advantage of the propulsion module's self-correcting tendency. The amount of propellant required, based on the use of relatively low specific impulses appropriate for easily storable liquids (or in one case, the monopropellant H_2O_2), varied from about 0.5 to 1 percent of the nuclear-pulse propellant consumed during the same propulsion period. For the performance calculations of this study, therefore, chemical control propellant (and its tankage) was provided, and it was presumed that the quantities consumed would be equal to 1 percent of the nuclear-pulse propellant.

2.1.8. Access for Maintenance and Repair

The nuclear-pulse-propulsion module has sufficiently low activation which permits manned access to most of the propulsion module minutes after shutdown. Manned access to the pusher itself is permitted within hours after shutdown. Further, the interior of the propulsion module is widely accessible owing to the nature of the propellant and the propulsion system. Therefore, for manned exploration vehicles, simple access provisions are made from the personnel compartments into the propulsion module through airlocks in the payload spine that connects the two. The main structural compartment of the propulsion module is unpressurized (except that a low pressurization may be desirable for satisfactory operation and lubrication of mechanisms); therefore, pressure suits are required unless selected areas are temporarily pressurized for occupancy.

Because of the above indicated accessibility to the propulsion-module mechanisms, the vehicle system of this study can carry a considerably larger allowance of spares and repair equipment (3,400 kg for the 10-m exploration vehicles; 16,000 kg for larger 20-m vehicle) than indicated in comparable studies. It may be found advantageous to carry even more spares, since on the relatively long exploration missions there is ample time and ample access to perform maintenance and repair tasks with these propulsion systems.

2.2. PARAMETRIC DATA

Previous design studies on nuclear-pulse-propulsion modules have been concerned with specific design points for rather large modules (e.g., thrusts of 44×10^6 newtons). In order to derive system data with which propulsion modules of more appropriate size could be selected, the specific impulse, thrust-to-weight ratio, and physical size of the nuclear-pulse-propulsion module were generated on a parametric basis.

2.2.1. Specific Impulse

The specific impulse was generated as a function of effective thrust, as indicated in Fig. 2.9. The nominal (center) curve was based on the performance derived for a previous detail design (at 44×10^6 newtons) and a number of less rigorous investigations at lower thrusts. In generating this curve, the relationships between the fundamental characteristics of the pulse-unit design, the pusher interaction phenomena, and the vehicle performance requirements in terms of payload and mission velocity were used. These relationships for the most part were optimized for a nominal mission requirement of 75,000 ft/sec. Further development of this important optimization is currently proceeding under a separate contract.
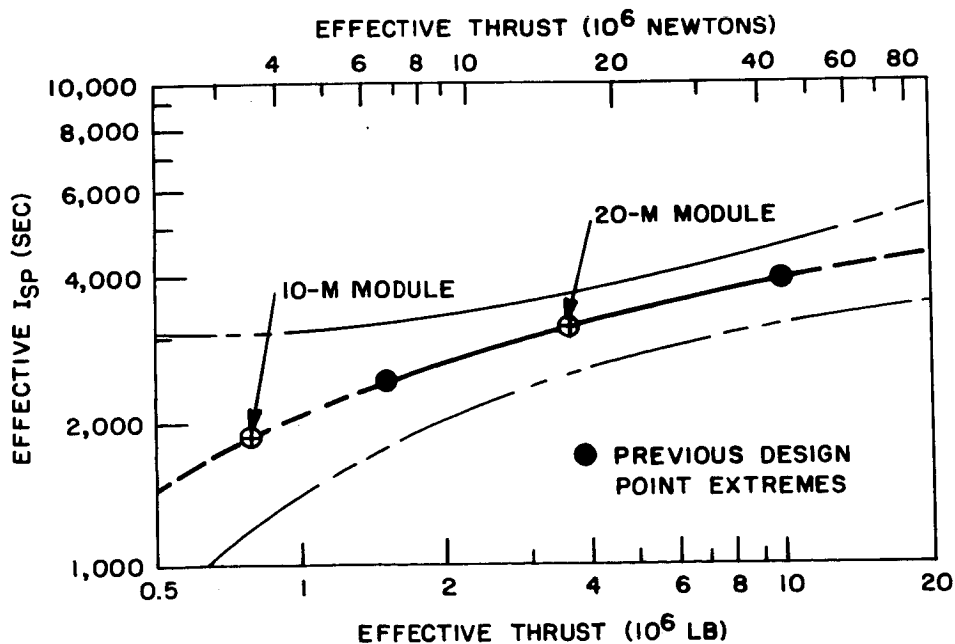
Fig. 2.9--Propulsion-module specific impulse

In addition to the nominal specific impulse (solid line), an uncertainty band is shown to represent optimistic and pessimistic limits. In view of the fact that these data were generated on the basis of a 75,000 ft/sec mission velocity, it may be that performance figures are conservative for more demanding missions.

The impulse delivered to the pusher is obtained by calculating the momentum carried by the propellant plasma as it expands into the solid angle defined by the pusher relative to the point of explosion. The tangent of this angle is $\alpha_e = R_p/L$, where R_p is the pusher radius and L is the stand-off distance from the point of explosion to the pusher.

The specific impulse of the pulse unit itself, defined here as the pulse-system specific impulse, I_{ps}, involves only the mass of the bare pulse unit:

$$I_{ps} = \frac{M(\alpha_e)}{M_{ps}} \frac{V(\alpha_e)}{g} \quad ,$$

where $M(\alpha_e)$ = mass contained within the angle defined by α_e,

$\quad V(\alpha_e)$ = average velocity directed perpendicular to the pusher within the angle defined by α_e,

$\quad M_{ps}$ = mass of the bare pulse unit.

In order to determine the propulsion system specific impulse, a correction factor must be applied to I_{ps}, taking into account such considerations as the additional mass allowance for antiablation oil, coolant and working fluids for the shock absorber and other subsystems, pulse-unit ejection gas, pulse-unit hardware for loading and handling, and an allowance for spare pulse units to replace faulty, mispositioned, or misfired pulse units. An estimate of these requirements was determined as a function of thrust, as presented in Fig. 2.10. (In order to facilitate reading of the values for the nonpropulsive expendables, the curve for these is plotted a factor of 10 higher than the curve for the pulse unit and total mass.)

The combination of these mass allowances as a fraction of the mass of the pulse unit is defined as ϵ, where

$$\epsilon = \frac{\epsilon \text{ mass of nonpropulsive expendables}}{M_{ps}} \quad .$$

Thus, the effective specific impulse for the propulsion system is

18

$$I_{sp} = \frac{I_{ps}}{1 + \epsilon} \quad .$$

Given the quantity I_{ps}, the effective engine I_{sp} can be defined by accounting for other masses expended during one pulse period. The numerical value of ϵ is approximately 0.2 for most systems which have been studied.
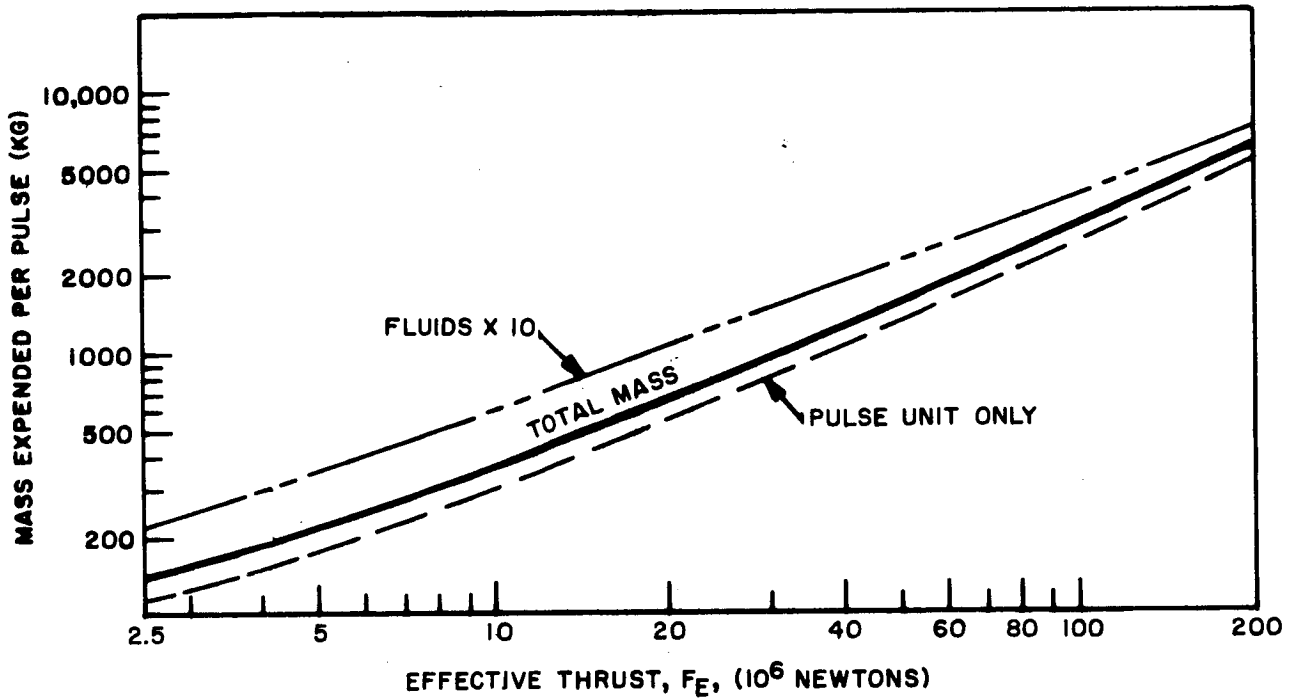
Fig. 2.10--Over-all propellant composition

2.2.2. Start-up and Shut-down Expendables

The only significant mass loss in a cycle of propulsion module operation, not otherwise accounted for, is the loss of shock-absorber gas in preparing first for operation and then for the longer periods of coast. It is currently assumed that the cylindrical second-stage shock absorbers remain pressurized at all times and also serve as coast-period storage containers for most of the gases required by the first-stage (flexible tori) shock absorber.

At the beginning of a coast period after a period of operation, the first-stage shock absorber is pumped down to a low pressure (compressing the gas in the second-stage cylinders), then vented as required to compress the tori to about one-third their neutral height. At this compressed level the flexible tori are enclosed within a meteoroid protection skirt that extends below the intermediate platform so that it reaches the retracted pusher. The relatively vulnerable tori are thus protected from smaller particles during the usually long coast periods.

Prior to engine start-up after a period of coast, the extra gas in the second-stage cylinders is released into the first-stage tori, and additional gas from on-board storage is used to bring the complete shock-absorber system up to operating pressures. The estimated amount of additional gas required, including the replacement of minor leakage from the second-stage cylinders during coast, is 25 percent of the capacity of the complete shock-absorber system. This quantity, converted to mass expended per start-up cycle as a function of effective thrust, is shown in Fig. 2.11.
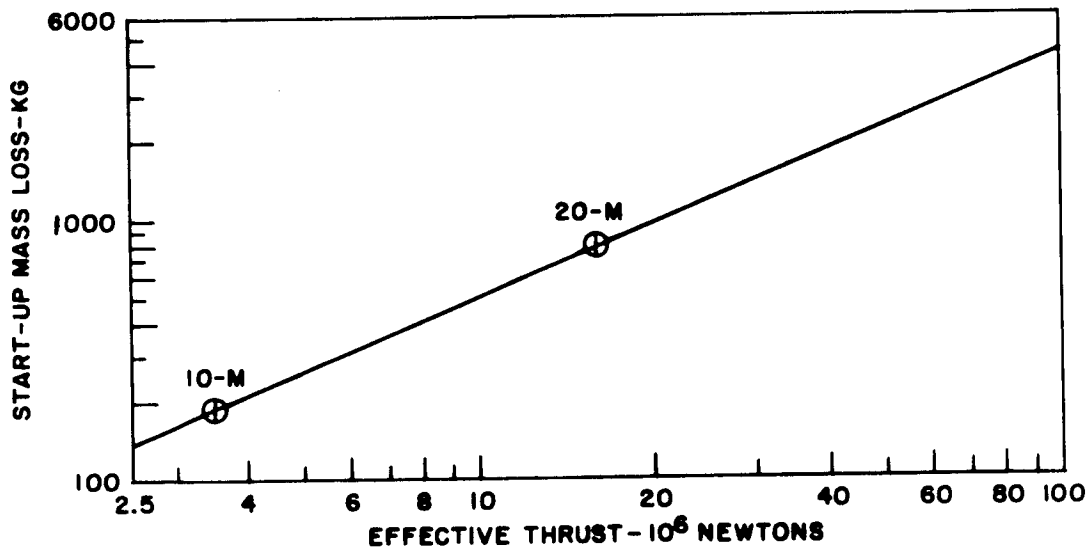
Fig. 2.11--Estimated mass loss per propulsion start-up cycle

The start-up mass loss is small enough--between one and two times the mass of a single pulse unit--that it does not excessively degrade the propulsion efficiency even when only small velocity increments are required. Even for course corrections requiring but tens of feet per second velocity increment so that only one pulse unit is required, the effective specific impulse exceeds that of the best chemicals.

At this point in the design of the propulsion modules, there appears to be no constraint on the number of times the propulsion operation is started up or shut down other than tankage capacity for the necessary gases. In fact, occasionally exercising the propulsion module in making minor course corrections may be less detrimental than the otherwise longer periods of inactivity.

2.2.3. Propulsion-module Effective Thrust-to-weight Ratio

The predicted effective thrust-to-weight ratio, F_E/W_E, at the time of this study, is shown as a function of effective thrust in Fig. 2.12. The nominal (center) curve was based on design analyses of a number of points in the range of 6×10^6 to 50×10^6 newtons. At 44×10^6 newtons the design point for the most detailed analysis is shown for reference. It will be noted that there is a very slight decrease in F_E/W_E with increasing F_E. This reflects the belief that the smaller modules can be built with somewhat proportionally lower pusher and module structure weight. An additional contributing factor is that the optimum average velocity increment per explosion is assumed to decrease slowly with decreasing thrust.
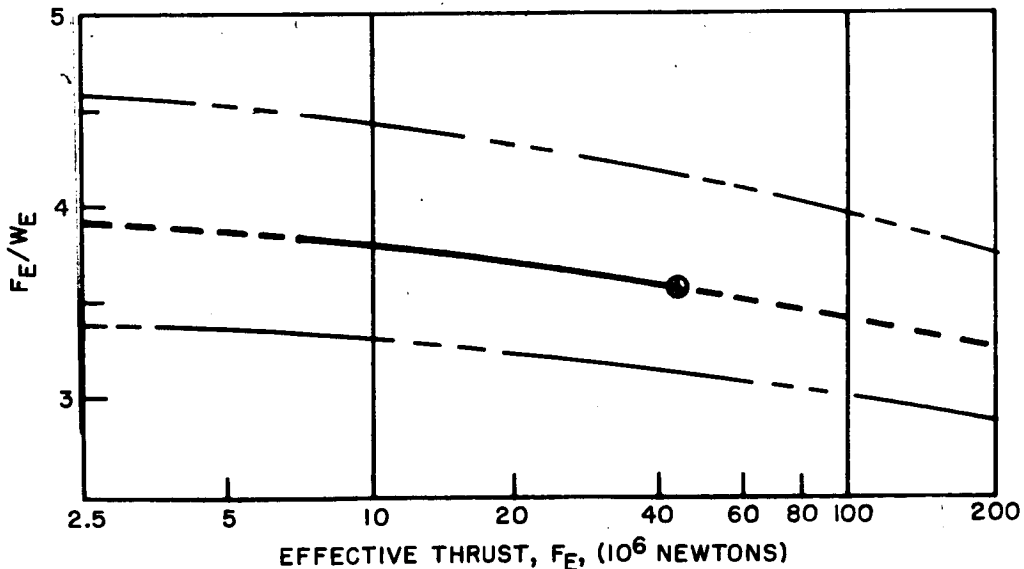
Fig. 2.12--Propulsion-module effective thrust-to-weight ratio (F_E/W_E)

The propulsion-module weight considered in deriving this thrust-to-weight curve is the "dry" weight, which includes all elements

21

of the module: pusher, shock-absorber system, upper structure, control system, auxiliary power supply for the module, shielding for the equipment (where applicable), and tankage and pumping system for operating the coolants, pusher antiablation oils, and other fluids.

For these data, the module structure (less the pusher, shock-absorber system, and miscellaneous equipment) was designed for Mode I operation, and therefore the design is based on the loads associated with a fully loaded vehicle at launch. This results in somewhat higher structure weights for Mode I than are necessary for Mode II or Mode III operations, where the propulsion module is partially or completely empty during the suborbital flight.

The propulsion-module weights include storage capacity for approximately 900 pulse units plus the fluids to be expended with the units. In some rare instances it may be necessary to increase the propellant storage capacity to permit an excess of the 900 pulse units to be consumed in one continuous acceleration period without the delay to restock the propellant storage facilities from expendable propellant magazines. For such a storage increase, the basic module is lengthened to increase its capacity, which increases its dry weight by 0.075 times the mass of the additional pulse units.

For pulse-unit requirements in excess of the capacity of the basic module, which would usually be encountered in the lower thrust-to-weight condition prevailing in operational Modes II and III, expendable magazines may be used. These pulse-unit magazines would have less sophisticated and less stringently operated pulse-unit handling systems than the basic module. The weight of the magazines is currently estimated at 0.05 times the mass of the pulse units contained in them.

The greatest single weight component is the pusher, and any potential weight reduction would naturally center on this item. Development of the basic design of the pulse unit may contribute to significant pusher weight reduction by means such as variation of pusher diameter, impulse distribution, and peak pressure.

A preliminary investigation of the potential design changes has been made, and the two dashed curves shown with the center curve of Fig. 2.12 reflect optimistic and pessimistic estimates of module-weight variation. The highly significant influence of module weight requires additional study, particularly with respect to the smaller module sizes and specific mission applications. This is believed to be particularly true when it is necessary to make rather specific and definite comparisons

with other systems and to meet or to recommend particular operational techniques which, in turn, would influence the propulsion-module weight. This is considered to be an important area for future work.

2.2.4. Propulsion-module Over-all Dimensions

The over-all dimensions of nuclear-pulse-propulsion modules over a wide thrust range are shown in Fig. 2.13. The basic module length (L_E) includes sufficient storage volume within the module to accommodate approximately 900 pulse units and the tankage for the coolants required to support the operation of the module during the firing of the pulse units. For additional propellant capacity within the basic module (as opposed to storage in separate magazines), the length ΔL of 0.03 L_E is added to the basic length for each additional 500 pulse units desired.
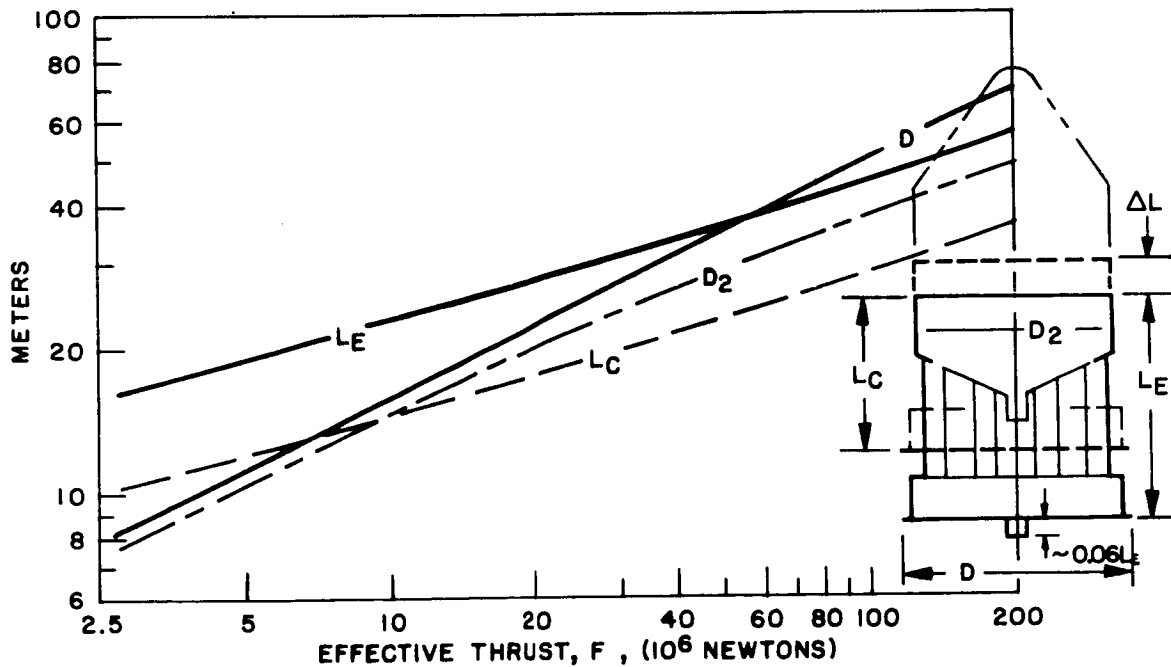
Fig. 2.13--Propulsion-module over-all dimensions

The vehicle upper structure, payload compartment, and any expendable propellant magazines are to be located above the basic module envelope, in the general region shown by the phantom lines in Fig. 2.13.

The over-all length of the module is based on the assumption that a telescopic, double-piston, second-stage shock-absorber system is used. This length increases approximately 1.2 times L_E if the currently considered longer diaphragm-type shock-absorber system is used. For the specific conceptual designs of the 10-m or 20-m modules described in this report, the longer shock absorbers were used, resulting in a greater over-all module length. Length L_C represents the engine with the shock absorbers compressed. The shock absorbers remain compressed during lofting to above the atmosphere (operational Mode I or II) and during boosting to orbit (operational Mode III).

For lofting or boosting, it is assumed that the pusher is supported or clamped to the booster by the lofter interstage structure. The interstage structural fairing extends around the pusher to hold the body section of the propulsion module to which booster forces are transmitted. The compatibility of the propulsion module and booster is discussed in Sec. 4.

2.2.5. Pulse-unit Yield

The yield of the nuclear-pulse unit is one of the more uncertain parameters of the propulsion system design. The efficiency with which the energy of the current-design nuclear device can be applied is certainly subject to improvement. Figure 2.14 shows a band plotted to give a semiquantitative indication of the yield required as a function of propulsion-module thrust. A previous design point is shown to indicate the point from which the data were scaled.

2.2.6. Formulas for Propulsion-module Characteristics

It is desirable to express the characteristics of the current-design propulsion modules in formulas that will approximate their performance. Such simplified formulas are applicable to broad-scope studies making gross comparisons between competitive propulsion systems. In this context it should be remembered that the following reflects current nuclear-pulse-propulsion modules as defined for this report, not the advanced systems presented in Sec. 2.5.

Based on current understanding of the propellant-pusher interaction, scaling laws have been developed which relate the amount of ablation expected during the interaction to properties of the ablative material that is used to protect the pusher and to properties of the stagnating propellant gas. A semianalytic model of the expansion of the propellant cloud has also been developed for the behavior of pulse units, based on the extensive numerical calculations. The model allows an analytic description of the properties of the propellant cloud to be related to pulse-unit design parameters. Such a model, backed up by continuing detailed study of the pulse unit needed to keep the various

24

parameters in touch with reality, can be used to relate the pulse-unit design to propulsion-module design.
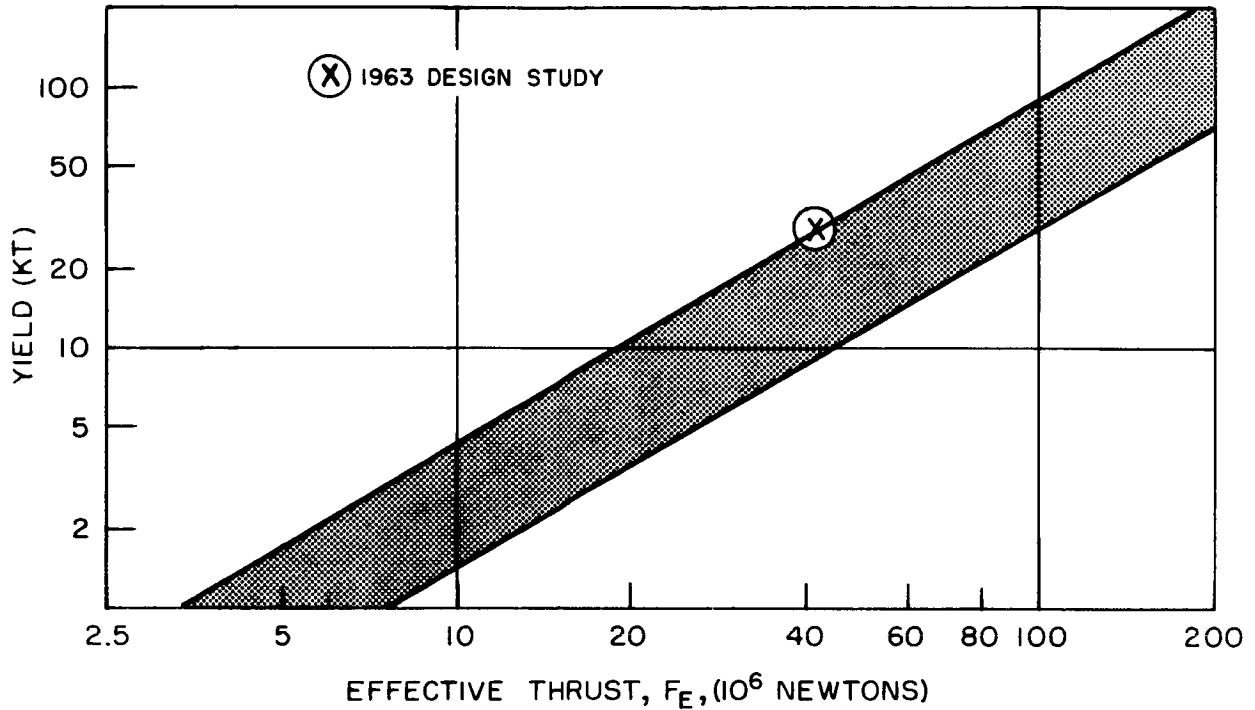
Fig. 2.14--Nuclear pulse-unit yield range

Using the model and the scaling laws for the propellant-pusher interaction, propulsion-module optimization studies can be conducted to obtain maximum specific impulses for a given module mass and module thrust-to-weight ratio. A "first-cut" at such a study has been completed, using a number of simplifying assumptions. The propulsion module was considered to be of the same type as the basic designs discussed in this report. The effective specific impulse, I_{sp}, the yield required, Y, and the propulsion-module thrust-to-weight ratio, F_E/W_E, were obtained as functions of the effective thrust, F_E:

$$\frac{1}{I_{sp}} = \frac{C_1}{F_E(1 + C_3 F_E^{1/3})} + \frac{C_2(1 + C_3 F_E^{1/3})}{F_E^{1/3}} \quad ,$$

25

$$\frac{F_E}{W_E} = 3.6 \ ,$$

$$Y = C_4 F_E^{4/3} \ ,$$

where I_{sp} = effective specific impulse, in <u>seconds</u>,

F_E = effective thrust, in <u>newtons</u>,

W_E = propulsion-module (engine) weight ($M_E g$, where mass is in kilograms),

Y = yield required, in kilotons,

$C_1 = 5.30 \times 10^2$,

$C_2 = 4.32 \times 10^{-2}$,

$C_3 = 2.83 \times 10^{-3}$,

$C_4 = 9.3 \times 10^{-10}$.

These formulas are currently considered to be valid over a range in thrust of

$$3 \times 10^6 < F_E < 2 \times 10^8 \ .$$

2.3. REFERENCE DESIGNS

Using the foregoing parametric data on the propulsion module, parametric vehicle-system-performance data were generated (Sec. 3, Vol. II). These data indicated that a relatively low-thrust ($\sim 3.5 \times 10^6$ newtons) module can perform manned Mars or Venus exploration missions with a comfortable margin when operating from earth orbit. Since a 10-m-diam module produces the required thrust and is of a size and weight compatible with the diameter and orbit delivery capability of the Saturn V, this design point was selected as a reference design for the study.

In order to provide for capability for missions involving very high payloads and/or high velocity increments, a larger 20-m-diam propulsion module was also selected as a reference design. The weight and size of this module are beyond the capability of the Saturn V for one-piece orbital delivery in Mode III, but it is anticipated that either an adequate booster will be available or large-scale orbit-assembly operations will be feasible by the time such a system is required.

2.3.1. 10-meter Propulsion Module

The following principal characteristics of the reference design 10-m module selected for this study were derived from the parametric data presented in Sec. 2.2.:

$$F_E = 3.5 \times 10^6 \text{ N}(786,000 \text{ lb}) ,$$

$$I_{sp} = 1850 \text{ sec},$$

$$W = 90,946 \text{ kg } (200,500 \text{ lb) dry.}$$

The weight is for the basic module only, without payload support spine and magazine and external payload support structure.

The reference design 10-m nuclear-pulse-propulsion module is illustrated in Fig. 2.15. The basic module is comprised of the pusher plate, first- and second-stage shock absorbers, and body section. An upper structure is provided to support the payload and pulse unit magazines.
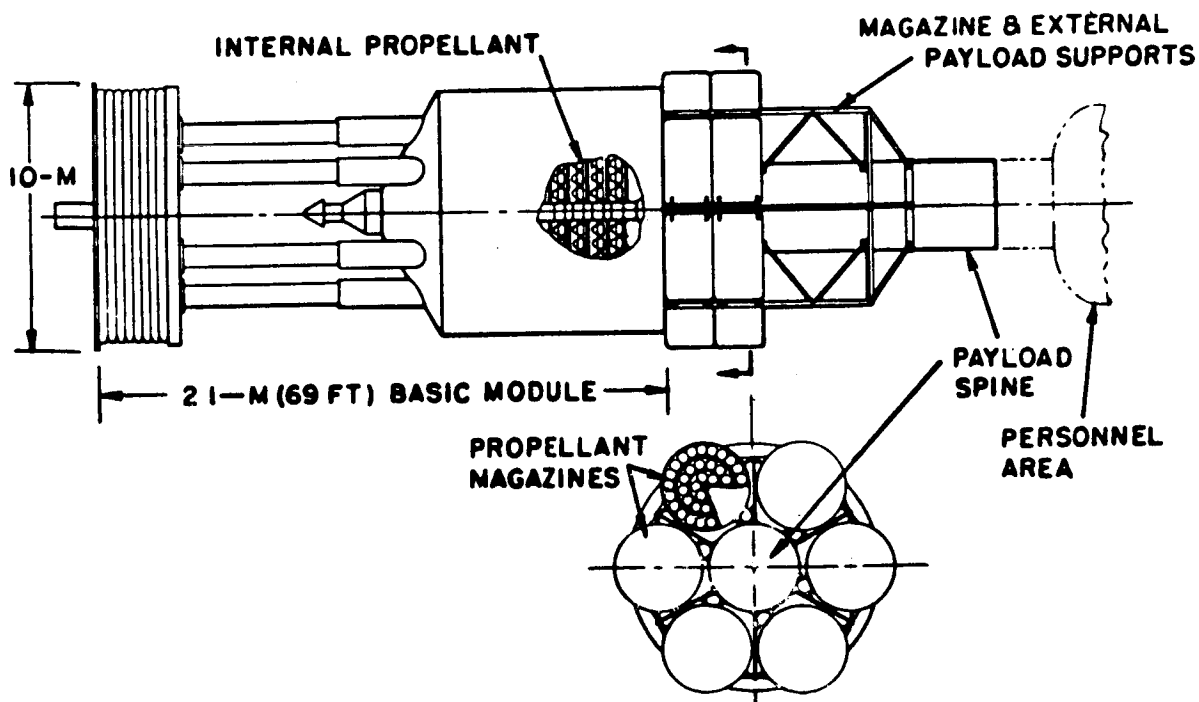
Fig. 2.15--Study configuration of the 10-m nuclear-pulse-propulsion module

The general features and operating characteristics of this design are similar to those for nuclear-pulse-propulsion modules in general, described in Sec. 2.1. For this design, a 141-kg (311-lb) pulse unit is used. In order to achieve the nominal thrust of 3.5×10^6 N, they are ejected and exploded at a period of approximately 0.86 sec.

The body section of the propulsion module accommodates 900 pulse units, while each of the magazines accommodates 92 units (90 plus 2 spares). The magazines are jettisoned in pairs after transfer of the pulse units to the body-section storage during a coast period.

The basic module is 21 m (69 ft) long when the shock absorbers are in the neutral position. A minimum spine length of 12 m is used, with additional length being provided if required to accommodate additional magazines and external payload. The spine and magazine support structure are not considered a part of the propulsion module or included in its dry weight.

2.3.2. 20-meter Propulsion Module

The reference design 20-m module selected for more advanced missions has the following characteristics, which were also derived from the data presented in Sec. 2.2:

$$F_E = 16 \times 10^6 \text{ N } (3,600,000 \text{ lb}) ,$$
$$I_{sp} = 3150 \text{ sec} ,$$
$$W = 358,000 \text{ kg } (789,400 \text{ lb}) \text{ dry.}$$

The 20-m module is similar to the 10-m configuration, except that it utilizes two concentric circles of shock absorbers instead of a single one in order to provide for distribution of the larger pulse load. The module has (as its designation implies) twice the diameter of the 10-m version, and it is 31 m (102 ft) in length, or approximately 1-1/2 times as long as the 10-m module.

Each pulse unit weighs 450 kg (993 lb); the units are ejected at a period of approximately 0.87 sec.

As with the 10-m design, 900 pulse units are accommodated in the basic module. Additional pulse units are accommodated in magazines with a capacity of 163 units (160 plus 3 spares) per magazine.

2.4. ADVANCED-VERSION PROPULSION MODULES

The propulsion modules of this study are rather austere and inefficient in comparison with the apparent potential of nuclear-pulse propulsion; there is a pronounced growth potential. The designs of this study have been kept in a performance regime in which, after six years of study and experimentation, the system interactions are reasonably well understood. The development of more advanced systems with noticeably improved capability and system economics can be predicted.

If it is assumed that ablation as it is now understood is the dominant physical phenomenon limiting the specific impulse that can be achieved, and that the ablation scaling laws are applicable to average velocities greater than 2×10^7 cm/sec, then a maximum I_{sp} can be calculated on the basis of these assumptions. For pulse units with the same degree of propellant focusing that has already been achieved by computer simulation, the ablation-limited maximum I_{sp} for a propulsion module with a thrust of 4.5×10^7 newtons is approximately 10,000 sec. If the pulse-unit design can be improved to the point where one-half the mass of the pulse unit is intercepted by the pusher, then the ablation-limited maximum I_{sp} for the same module is 16,000 sec.

Given several years of further research and investigation into the problems associated with pulse units and propulsion-module design, it seems quite likely that specific impulses of about 20,000 sec and module thrust-to-weight ratios of 3 or larger can be obtained for nuclear-pulse-propulsion modules having a thrust of 4×10^7 newtons or greater.

From the above considerations, it is predicted that nuclear-pulse-propulsion modules for the post-1980 era will have an effective specific impulse between 10,000 and 20,000 sec. The same propulsion modules will have thrust-to-weight ratios between 3 and 4.

The payload delivery capability and system economics of vehicles using such advanced propulsion modules are given in Sec. 6 of Vol. II.

Several means of noticeably improving nuclear-pulse-propulsion capability are available in addition to the large "second generation" vehicles discussed above. Probably the most important of these is the redesign of the pulse unit for use with the smaller (~10-m) class of vehicles. With pulse-unit redesign, orders-of-magnitude improvements in fission-product release (near-fusion or total-fusion energy source) as well as improvements in specific impulse can be attained, opening up additional operational regimes.

Other means of improvement lie in the design of the propulsion module, particularly the pusher and shock-absorber system. Here, advanced material properties, or alternative arrangements involving new combinations of materials, may permit higher propellant velocities which, in turn, provide correspondingly higher specific impulses.

Economic improvements for nuclear-pulse propulsion are also closely tied to the design of the pulse units. As stated in Sec. 8, the costs per kilogram for propellant are subject to reduction by factors of 2 to 4 through pulse-unit redesign, with the larger reduction applying to the smaller modules, such as the 10-m module of this study.

3. COMPLETE VEHICLE CONCEPTUAL DESIGNS

A complete nuclear-pulse-propelled vehicle may be considered to be composed of the propulsion module plus an upper vehicle. The upper vehicle, at the minimum, carries the payload and a guidance and control system and protects the payload. The propulsion module, as described in the preceding section, provides, in rocket terminology, the equivalent of an engine, tankage, plumbing, and thrust structure.

During the first half of the study, which was in part directed to defining the significant module parameters, the upper vehicle was defined very simply. For the early parametric performance calculations, the upper vehicle was considered to have a cone or an ogive configuration and its weight was a function of the propulsion module weight and the weight of payload. The weight of the upper-vehicle structural shell with a guidance and control system was approximated to be 10 percent of the propulsion-module weight, with an additional payload support increment approximated to be 5 percent of the weight of the payload carried.

For the specific conceptual designs generated during the last half of the study, the upper-vehicle structures were defined in much greater detail. They were designed to meet the specific requirements of the missions and the propulsion system. But, as would be expected, many of the requirements stemmed from the needs of mission personnel. Therefore, the mission personnel accommodations are discussed prior to examining the various conceptual vehicle designs.

3.1. PERSONNEL ACCOMMODATIONS

Two requirements for personnel accommodations appear most important in shaping the configuration of a nuclear-pulse-propelled vehicle. First and most obvious is the need for living—work quarters and an ecological system that will satisfy personnel needs for periods up to one year or more. Second is the requirement for protection from the propulsion system's nuclear radiation, which intermeshes with the consideration of protection from normal galactic radiation or from solar-flare radiation.

The living space and ecological requirements for exploration missions have been studied by various concerns and agencies over the past few years. For this study, the latest work on ecological systems by General Dynamics/Astronautics[*] was used either directly or as a basis for extrapolations. The radiation protection requirements for nuclear pulse propulsion are conversely different than those for other nuclear systems. Therefore, this subject was considered in greater detail.

3.1.1. Personnel Shielding Requirements

One of the major differences between nuclear pulse and most other nuclear propulsion systems is the relative freedom from activation from pulse propulsion. Except during the actual propulsion periods (typically lasting only 5 to 15 min per major maneuver), there is almost negligible radiation from any part of the vehicle except the pusher. Even the pusher radiation levels permit personnel access within a few hours after operation.

Radiation levels during the nuclear pulse-**unit** explosion, which are discussed in Section 5, are such that all personnel <u>must</u> be in shielded compartments during the explosions. The amount of shielding required is shown in Fig. 3.1. These data were computed early in the study program and have subsequently been found to be somewhat conservative. The 7 to 10 percent shielding weight reductions currently indicated were not integrated, however, into the vehicle performance calculations. The effect of such a weight reduction can be seen in the inert-mass sensitivity data of Vol. II.

The curves of Fig. 3.1 show the amount of shielding required on the bottom of the powered flight station to keep the radiation dose (from propulsion) per mission to 50 rem. Three curves are for the hydrogenous shielding material required for 1,000 to 10,000 pulses per mission. The estimated requirement for 55 g/cm^2 of lead or equivalent at the upper surface of the bottom shielding was considered to be essentially independent of the thrust or number of pulses. The required sidewall and top shielding to resist scatter radiation (for typical designs having but little structure or other masses to cause sidewall scatter) was estimated to be 25 g/cm^2 of hydrogenous material.

The vertical lines through the shielding curves represent the thrust levels of the 10-m and 20-m propulsion modules of the study.

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[*]<u>A Study of Manned Interplanetary Missions</u>. Vol. II—Weight and Payload Report, General Dynamics/Astronautics, Report AOK 64-006-5, 1964.
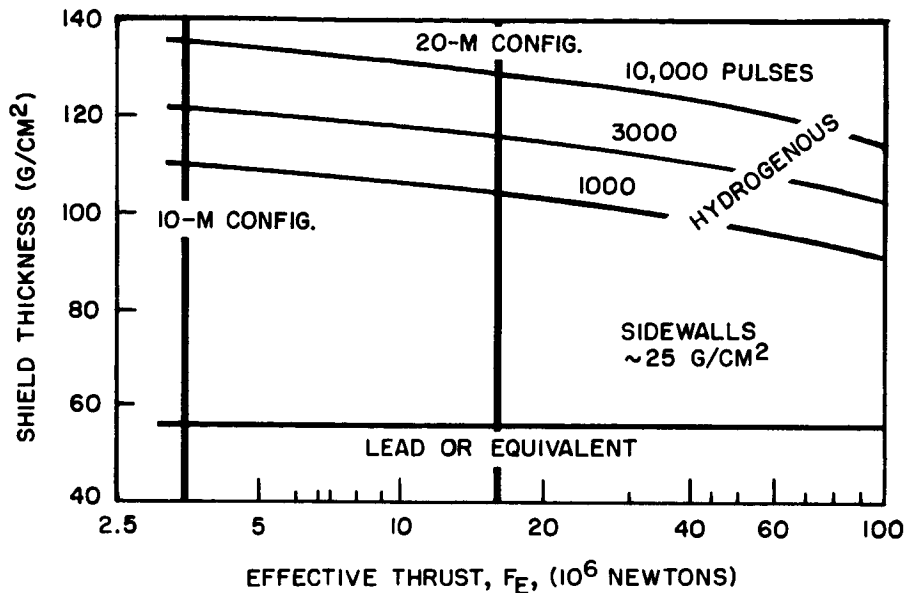
Fig. 3.1--Shielding requirements for bottom
of powered flight station

Most missions for the 10-m configuration require less than 3,000 pulses, so a bottom shielding requirement of 120 g/cm² of hydrogenous material was used for most conceptual designs.

Shielding of 25 g/cm² or more was thus provided on all sides of the powered flight station. According to recent studies of planetary exploration requirements, this shielding is sufficient to attenuate the probable solar-flare radiation to dose values similar to the 50 rem allowed for propulsion. Thus, the total mission dose from both sources was expected to be about 100 rem.

The location of the powered flight station and personnel accommodation was strongly influenced by the shielding for propulsion requirement in order to minimize the shielding weight. The need for ready access between the powered flight station and living quarters and the assumed requirement for an emergency escape capability were also factors in locating and shaping the personnel compartments.

Figure 3.2 shows schematically the major factors that influenced the design. The cylindrical spine extending below the personnel compartment is the structural tie between the personnel compartment and the propulsion module. It serves several purposes. In addition to maximizing the distance from the pulse radiation sources as shown, it

33

ACCESS DOORS PREFERRED
IN TOP OR SIDEWALL

ESCAPE CAPABILITY DESIRED
FOR COMPARTMENT CONTENTS
(PERSONNEL AND ESSENTIALS)

MINIMIZE STRUCTURE OR
MATERIALS ADJACENT TO
SIDEWALL SHIELDING

MINIMIZE ANGLE FOR
SCATTER TO SIDEWALLS

REDUNDANT ACCESS TO
LIVING AND WORK SPACES
DESIRED

MINIMIZE MATERIAL MASS
LOCATED TO CAUSE
SCATTER TO SIDEWALLS

MAXIMIZE ATTENUATING MASSES
BETWEEN COMP'T AND PULSE
SOURCE

MAXIMIZE DISTANCE
FROM PULSE SOURCE

Fig. 3.2--Factors that influence the location of the shielded
powered flight station

provides sufficient personnel separation from the vehicle's center of
gravity to permit an acceptable radius of rotation for artificial g, it fits
within the circle of "stacked" propellant magazines or externally carried
payload, it provides an easy internal access to the propulsion module,
and it provides a convenient large area for repair and maintenance
equipment and spares storage. These features will be illustrated in the
over-all designs discussed subsequently.

The relatively large amounts of hydrogenous shielding material
required for the bottom of the powered flight station prompted consid-
erations of using dual-purpose material. Since the shielding is required
during final propulsion operation, it cannot be consumed for some other
purpose prior to the final nuclear-pulse maneuver. The use of shield
material as propellant for emergency escape, when provided, was con-
sidered acceptable since if such a propellant is used, the vehicle is then
abandoned. Two kinds of escape propellant were considered desirable:
solid-propellant rocket motors to rapidly initiate separation and storable
liquid propellant for post-escape maneuvers. Figure 3.3 shows dia-
grammatically the extent that these propellants were integrated into the
shielding for the escape vehicles or exploration missions. The quanity
of propellant provided for post-escape maneuvers gave the shielded
escape vehicle a ΔV capability of ~2,000 fps.
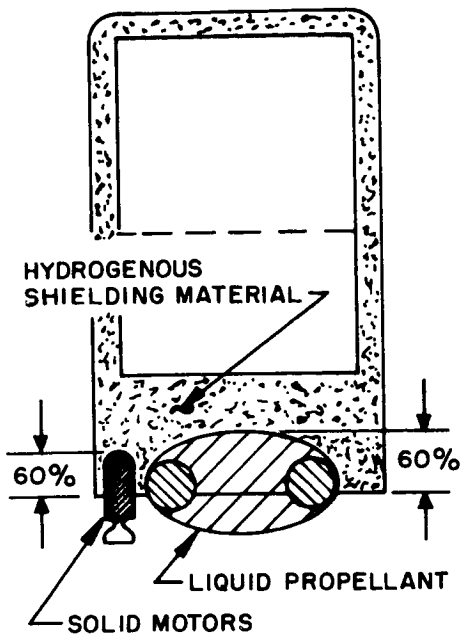
PROPELLANT REQUIREMENT:

  1. SOLID PROPELLANT ESCAPE MOTORS
  2. STORABLE LIQUID FOR POST-ESCAPE
    MANEUVERS

ASSUMPTION:

    SELECTED PROPELLANTS HAVE
    $\geq$ 2/3 SHIELDING EFFECTIVENESS
    OF SINGLE-PURPOSE MATERIAL

TENTATIVE APPLICATION:

    PROPELLANTS ~60% BURIED AS
    SHOWN.
    2/3 PROPELLANT MASS IS DEDUCTED
    FROM SHIELDING MASS REQUIREMENT

Fig. 3.3--Shielding-abort propellant combination used for the
combined powered flight station—escape vehicle

3.1.2. Powered Flight Station—Escape Vehicle

For exploration missions it is considered desirable to have an
emergency escape capability. Escape would be in order should a cata-
strophic failure of the basic vehicle be imminent (an uncontrolled fire,
explosion, etc.) or should propulsion fail completely while on a cata-
strophic trajectory. Escape would not necessarily be the appropriate
action, however, in the event of propulsion failure alone; staying with
the more completely equipped hulk appears logical until such time as
departure is most advantageous.

The dual requirements of the powered flight station, since it
also serves as a solar storm cellar, cause it to be equipped for con-
tinuous occupancy for as long as two weeks and therefore it must have
rather complete accommodations. Hence, it is not a large penalty to
put the emergency ecological and power supply system (both considered
a requirement in any event) and a 90-day portion of the supplies in an
unshielded portion of the escape vehicle. This was done for the escape
concepts of this study.

Figure 3.4 shows a cutaway view of the powered slight station—
escape vehicle designed for the 8-man exploration vehicles. The escape
vehicle is 3 m (9.85 ft) in diameter and approximately 9 m (29.5 ft) in
over-all length. The two-level shielded compartment has an ID of

90-DAY EMERGENCY SUPPLIES

NAVIGATION STA.

PROPULSION CONTROL CENTER (4-MEN)

BUNKS (4) AND STORM CELLAR NEEDS

SHIELDING

POST-ESCAPE MANEUVER PROPELLANT

ESCAPE ROCKETS
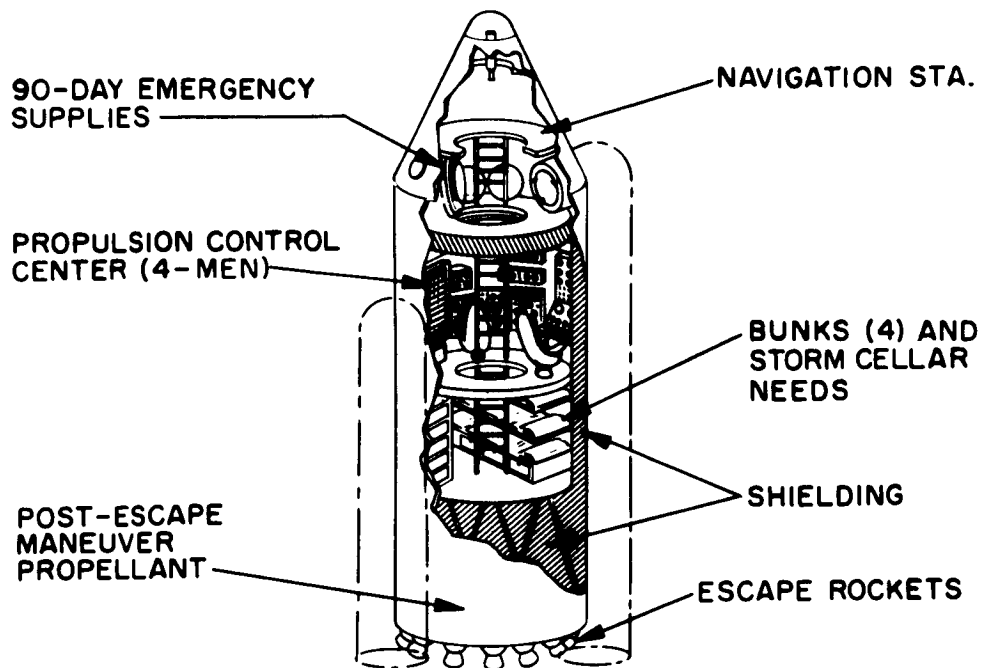
Fig. 3.4--Powered flight station—escape vehicle for 8-man
exploration missions with 10-m configurations

approximately 2.5 m (8.2 ft) and the room heights are 2.0 m (6.6 ft)
and 1.6 m (5.2 ft). The propulsion control center is in the higher com-
partment and the bunk room is in the lower. An upper unshielded room
serves as a navigation station, as an emergency-supply storage area,
and as the major accessway to and from the shielded compartment.
A secondary accessway is provided through the sidewall of the propul-
sion control center. There is no accessway through the thick bottom
shielding owing to the difficulty of controlling radiation leakage around
access ports and to the bulkiness of such thick ports.

A similar powered flight station—escape vehicle for the 20-man
exploration vehicles using the 20-m propulsion module is shown in
Fig. 3.5. The design principles are the same but the compartments
are commodious. The shielded compartment has a 4.0 m (13.1 ft) ID
and both rooms are 2.0 m (6.6 ft) high; two unshielded compartments
are provided above the shielded sections.

A similar, but smaller, 20-man powered flight station—escape
vehicle was designed for use with the 10-m exploration vehicle (since its
payload capability is considerably less than that of the 20-m vehicle).
This design has three levels in the shielded compartment, which has an
ID of 3.0 m (9.85 ft). There are two bunk rooms with low ceilings
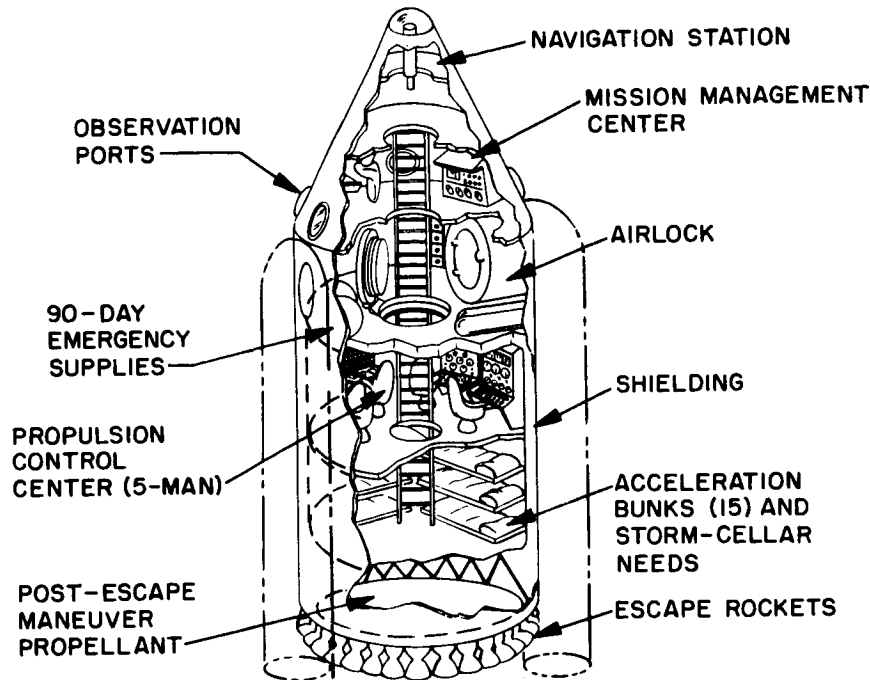
Fig. 3.5--Powered flight station—escape vehicle for 20-man
exploration missions with 20-m configurations

(1.6 m, or 5.2 ft) and a propulsion control center, which is the same as
in Fig. 3.4 except that the diameter is larger. The weight difference
between this design and the larger 20-man escape vehicle is primarily
due to shielding differences; 27,200 kg of shielding compared to 36,380 kg
for the larger one.

A weight summary for the 8-man and 20-man escape vehicles
pictured in Figs. 3.4 and 3.5 are given in Table 3.1. The total weights
represent the condition of the escape vehicles as they would depart from the
basic vehicle (the components listed here are itemized differently in the
weight summaries for the total vehicles).

3.1.3. Exploration-mission Living and Work Spaces

The exploration missions considered in this study have durations
varying from 150 days for a fast Mars round trip to 910 days for an
exploration of Jupiter. The more typical Mars and Venus missions varied
from 300 to 450 days. For these mission durations, it seemed important
to provide adequate spares and reasonably convenient arrangements for
work, relaxation, and sleep, and to provide suitable access from one
compartment to another. Pressure partitions with airlocks were

considered necessary to provide the main living spaces with at least
two pressure compartments that could be isolated from each other.
A serious puncture in one compartment could thus be repaired (by using
pressure suits) while a "shirt-sleeve" environment is maintained in
the other. Also, in the event of a catastrophic blowout, only the per-
sonnel in the affected compartment would be lost.

Table 3.1

WEIGHT SUMMARY FOR TYPICAL EXPLORATION
MISSION ESCAPE VEHICLES
(In kg)

	8-man Vehicle	20-man Vehicle
Vehicle and propellant:		
Shielding.	18,170	38,380
Structure	1,730	5,520
Motors, tankage and cases	600	1,600
Propellant (escape and maneuvers)	4,500	12,000
Totals	25,000	55,500
Contents at separation:		
Crew	725	1,810
Emergency life-support system	880	1,790
90-day life-support-system expendables	680	1,400
90-day food supply	1,080	2,700
Guidance, autopilot, navigation system	360	400
Control system	230	500
Emergency power supply	910	1,700
Communications system	135	200
Totals	5,000	10,500
Total escape-vehicle weights	30,000	66,000

Previous technical reports and papers on personnel require-
ments and accommodations and the concurrent efforts at General Dynamics/
Astronautics[*] were investigated. These inputs, combined with the
requirements and general arrangement of the nuclear-pulse-vehicle pro-
pulsion module, resulted in the personnel accommodations shown in
Figs. 3.6 and 3.7. The interior furnishings in both arrangements are
"upside down" since they are designed to experience artificial gravity
by rotation of the complete vehicle. There are two levels in the main
pressure vessel which are separated by a pressure floor.

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[*]A Study of Manned Interplanetary Missions, General Dynamics/
Astronautics, Report AOK 64-006, 1964.

PASSAGEWAYS

ECOLOGY STORAGE

GALLEY

STATEROOMS (4)

AIRLOCK

PRESSURE
FLOOR

LABORATORIES
AND
WORKSHOPS

REPAIR BAY/SPARES
STORAGE BELOW
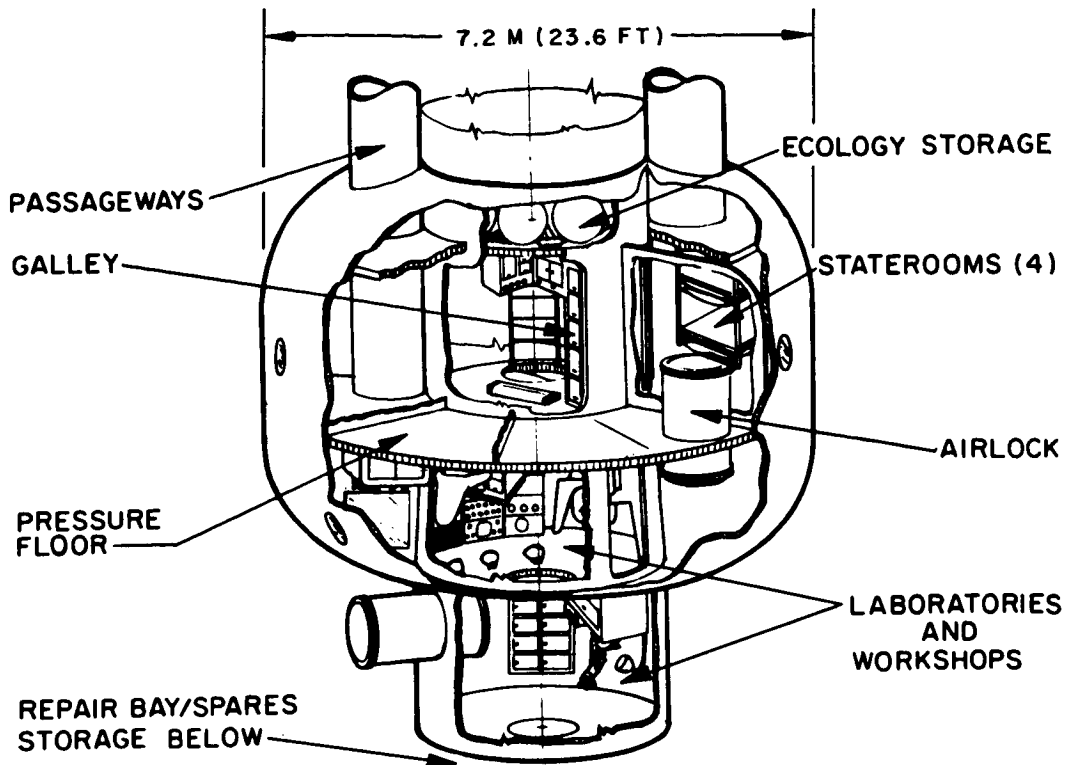
Fig. 3.6--Exploration-mission personnel accommodations
for an 8-man complement

ECOLOGY
SUPPLY

PASSAGE WAYS

STATEROOMS (10)

HEADS

ENTERTAIN
EXERCISE, ETC

PRESSURE FLOOR

LABORATORIES

SHOPS

EATING AREA

AIRLOCK

PRESSURE FLOOR

REPAIR BAY,
SPARES STORAGE

ENGINE ACCESS
AIRLOCK
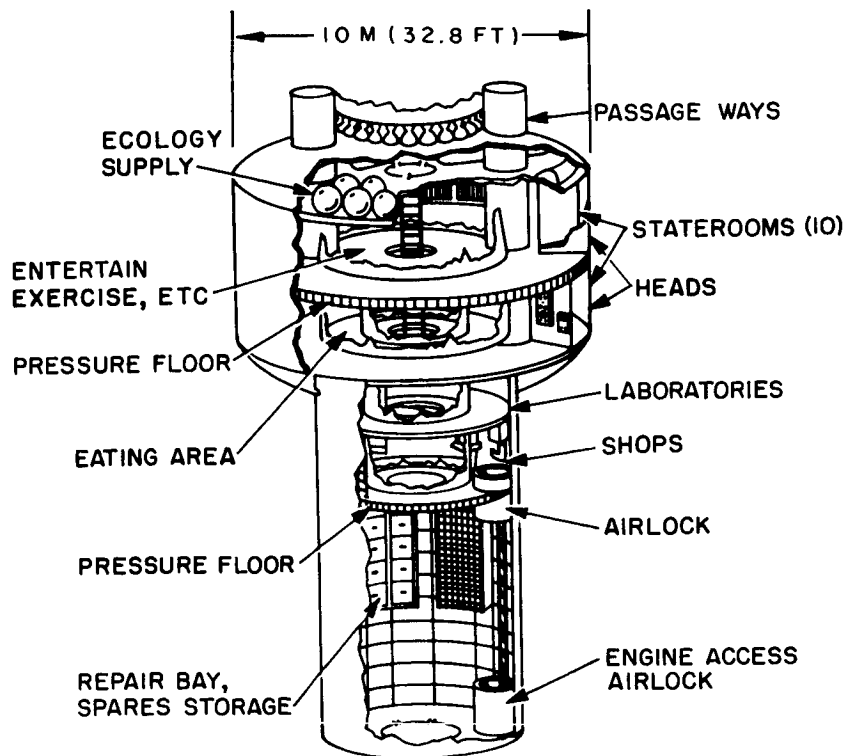
Fig. 3.7--Exploration-mission personnel accommodations
for a 20-man complement

39

The diameter of the 8-man accommodation of Fig. 3.6 is 7.2 m (23.6 ft). Both the upper and lower floors can be divided into eight 45° segments by nonstructural partitions between the outer wall and the inner structural continuation of the spine (which supports the accommodations). Two of these partitioned rooms on each floor serve as 2-man staterooms, one on each floor is a bathroom, three on each floor (but without partitions) serve as laboratories or workshops, and two more as a galley and dining-recreation room. The remaining two rooms are used for a command and communications office and for the main emergency-gear storage area. Another laboratory or workshop is located in the spine extension directly below the lower floor.

The additional volumes that are part of the continually pressurized personnel areas are the powered flight station and storage-room—navigation station in the escape vehicle. The areas in the escape vehicle are reached through one of the two cylindrical passageways shown. The right-hand passageway is the accessway to the upper main floor; it has an emergency airlock at top in the escape vehicle but only a pressure door at the bottom. Conversely, the left-hand passageway opens to the lower main floor through an emergency airlock and has only a pressure door at the top. All airlocks except those opening to the outside are for emergency use only. The total volume of the continually pressurized personnel accommodations, excluding passageways, is slightly more than 200 m^3 (7,100 ft^3) or approximately 25 m^3 (880 ft^3) per man. This volume does not include the relatively large repair-bay—spares storage area in the spine below.

The estimated weight of the structural components for the 8-man personnel accommodations of Fig. 3.6 is given in Table 3.2. Double-wall aluminum alloy construction with a 3-in.-thick 6 lb/ft^3 filler or core material for insulation and meteoroid barrier was assumed.

The arrangement of the 20-man personnel accommodations for the 20-m configuration shown in Fig. 3.7 is similar to the 8-man accommodations. Again, 2-man staterooms are provided and considerable space is allotted to workshops and laboratories. In this 10-m (32.8-ft) diam configuration the normally pressurized volumes, excluding passageways, total 590 m^3 (20,800 ft^3) or about 29.5 m^3 (1,040 ft^3) per man, not including the large repair-bay—spares storage area in the payload spine adjacent to the propulsion module.

The 20-man accommodations on the 10-m vehicle are similar to those for the 20-m vehicle, but do not have the shop or laboratory shown in the upper spine in Fig. 3.7. In this configuration, the normally pressurized accommodations are reduced to 24.5 m^3 (820 ft^3) per man.

Table 3.2

WEIGHT ESTIMATE OF STRUCTURE FOR PERSONNEL
ACCOMMODATIONS FOR 8-MAN EXPLORATION VEHICLE
(For payloads, A, B, or C)

	Weight (kg)
Floors	
3.1-m ID pressure floors (3)	414
Central large pressure floor (1)	706
Upper large nonpressure floor (1)	306
Total	1,426
Exterior surfaces	
7.2-m OD straight section	810
Domes (less center area) (2)	1,530
Spine stub ~2.2-m high	382
0.8-m ID passageways (2)	288
Passageway ends and expansion-separation joints (2)	60
Total	3,070
Internal vertical surfaces	
Internal spine extension	500
Floor-supporting walls	470
Nonstructure partition walls	120
Total	1,090
Miscellaneous internal structures	
7.2-m-diam structural rings (2)	200
Structural-wall openings and nonpressure doors (15)	180
Pressure hatches (6)	72
Airlocks (5)	200
Escape vehicle attachment-separation adaptor (1)	75
Tie-together and load-transfer structure	560
Total	1,287
Contingency (~6 percent)	727
Total structure weight	7,600

41

TYPICAL OPERATIONAL
PAYLOAD ENVELOPE AND CG

OPL CG

MAGAZINE
SUPPORT
COLUMNS
(TIED TO SPINE)

PAYLOAD SPINE
(3.2 M OD FOR IO-M DESIGNS,
6 M OD FOR 20-M DESIGNS)

PROPELLANT MAGAZINES
(OR EXTERNAL PAYLOAD CANISTERS)

I2-M
MIN.
SPINE
LENGTH

SPINE

MAGAZINE SUPPORT
COLUMNS (6)

PROPULSION MODULE
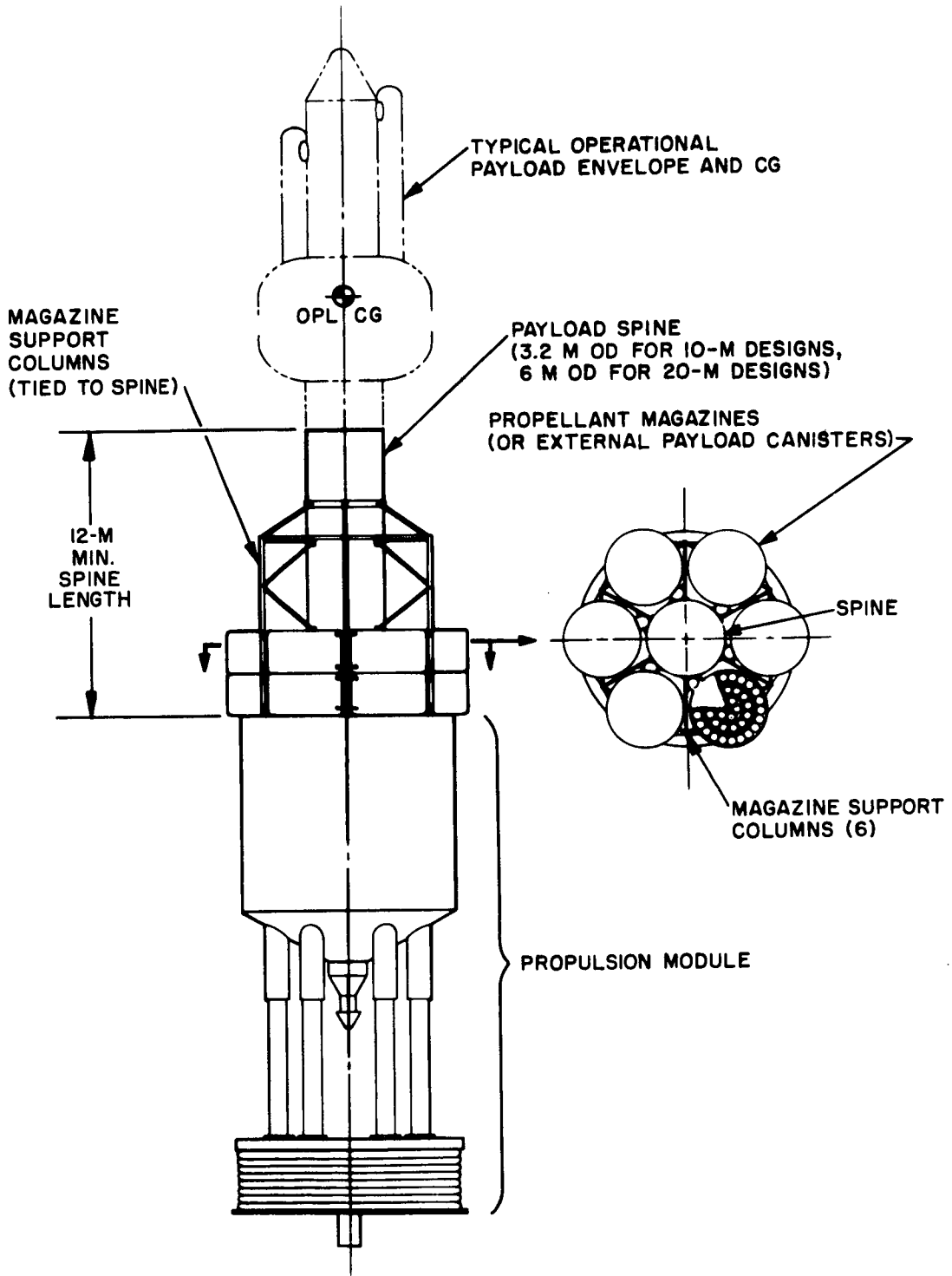
Fig. 3. 8--General arrangement of payload spine and magazine
payload support columns

42

One other personnel-accommodations configuration for explora-
tory missions was conceptually designed. It was an 8-man arrangement
of approximately the same total volume as the configuration of Fig. 3. 6,
but with some two-thirds of the volume in four expendable modules. The
design was patterned after one of the preferred configurations designed by
General Dynamics/Astronautics* for nuclear-rocket propulsion. Not only
was the configuration estimated to be somewhat lighter than that shown in
Fig. 3. 6, but the expendable modules could be discarded as the mission
progressed and their need diminished or they could be jettisoned if there
was a loss of propellant or other propulsion degradation occurred. How-
ever, because of the relatively high inert weight of the propulsion module
for the nuclear-pulse-propelled vehicles, expending the personnel modules
would have little effect on performance. Consequently, this configuration
was no longer considered for these vehicles after it was used to compute
the comparative performance reported in Volume II.

3.2. PLANETARY EXPLORATION VEHICLES

The over-all exploration vehicles are essentially the basic pro-
pulsion module and the personnel accommodations described for exploration
missions suitably tied together. This section treats of the payload spine
which accomplishes the structural tie and access connection between the
propulsion module and the upper vehicle, the propellant magazine and
external payload support structure, and the resulting over-all vehicle
configurations. A breakdown of the operational payload, which includes
the life-support structure as well as all equipment, expendables, per-
sonnel and items to maintain and operate the vehicle, is also given in this
section.

3.2.1. Payload Spine and Magazine Support Columns

The typical arrangement of the payload spine and the magazine
support columns is shown in Fig. 3.8. The spine serves as the only struc-
tural tie between the propulsion module and the operational payload.
Located around the spine are six support columns for the externally carried
propellant magazines or for payload carried externally in canisters (which
are the same diameter as the magazines). The support columns are struc-
turally tied to the payload spine through truss beams (the inner cap of the
beams is not shown).

The payload spine has a minimum length of 12 m (39. 4 ft), since
this length provides a minimum radius of 50 ft from the normally manned

*A Study of Early Manned Interplanetary Missions, General Dynamics/
Astronautics, Report AOK 63-0001, January, 1963.

personnel accommodations to the composite CG for artificial gravity purposes. The first designs of this concept during the study had a shorter spine whenever the space required for propellant magazines or external payload permitted and employed a coast-period spine extension to attain the 50-ft minimum radius. Provisions for the extension, however, were estimated to be about as heavy as a longer fixed spine and introduced another reliability problem (the longer spine also provides a radiation-shielding advantage because of the increased separation from the pulse source).

The height of the magazine support columns is a function of the number of propellant magazines and payload canisters carried. The estimated weight of the support columns (including the tie-in beams) was made a function of the mass of the magazines and external payload carried. After a point-design weight estimate was made, an allowance was established for the support-column weight: 0.5 percent of the net weight of propellant carried externally (in compact, high-density magazines) plus 1.0 percent of the weight of payload carried externally (estimated to average 400 g/m^3 (25 lb/ft^3) in density).

To estimate the weight of the payload spine and the operational payload structure adequately, it was necessary to approximate the critical design accelerations to which they would be exposed. The accelerations used are shown in Fig. 3.9 as a function of the thrust-to-weight ratio effective at any given time. The direction-of-flight accelerations under power are peak accelerations occurring during the typical pulsing acceleration profile (see Fig. 2.11) and may be either positive or negative.
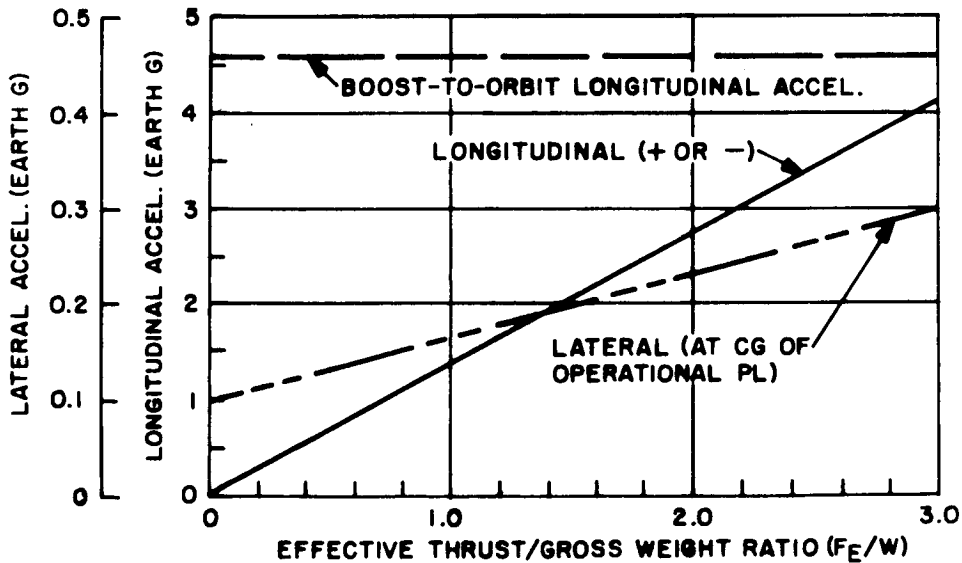
Fig. 3.9--Estimated critical design accelerations for upper vehicle

44

The boost-to-orbit longitudinal acceleration is shown as a one-time peak occurring at chemical booster first-stage burnout. During boost, a maximum lateral acceleration of 0.5 g was also expected (due to wind shear or booster dynamics), which was assumed to occur prior to the burnout maximum boost-to-orbit acceleration. The lateral acceleration shown is an allowance only that was assumed applicable during the powered-flight phase. The lateral accelerations were assumed to be simultaneous with the direction-of-flight pulses and are therefore additive.

Payload spine weights based on the above accelerations were estimated as a function of the operational payload mass supported by the spine and the spine length. The resulting weights of the spine for the 10-m exploration vehicle are shown parametrically in Fig. 3.10. For the solid curves of this figure it was assumed that the magazine support columns extend 6 m or more above the propulsion module. The support-column geometry forms deep beams which absorb most of the higher spine-bending loads near the spine base and thus relieve the spine's structural requirements. If there are no magazine support columns or if they are too low to be effective, the weight of the 12-m-long spine is shown by the dashed curve. (Such curves have no meaning for the longer spines since the longer lengths were only used to make room for longer columns carrying propellant magazines and payload.)
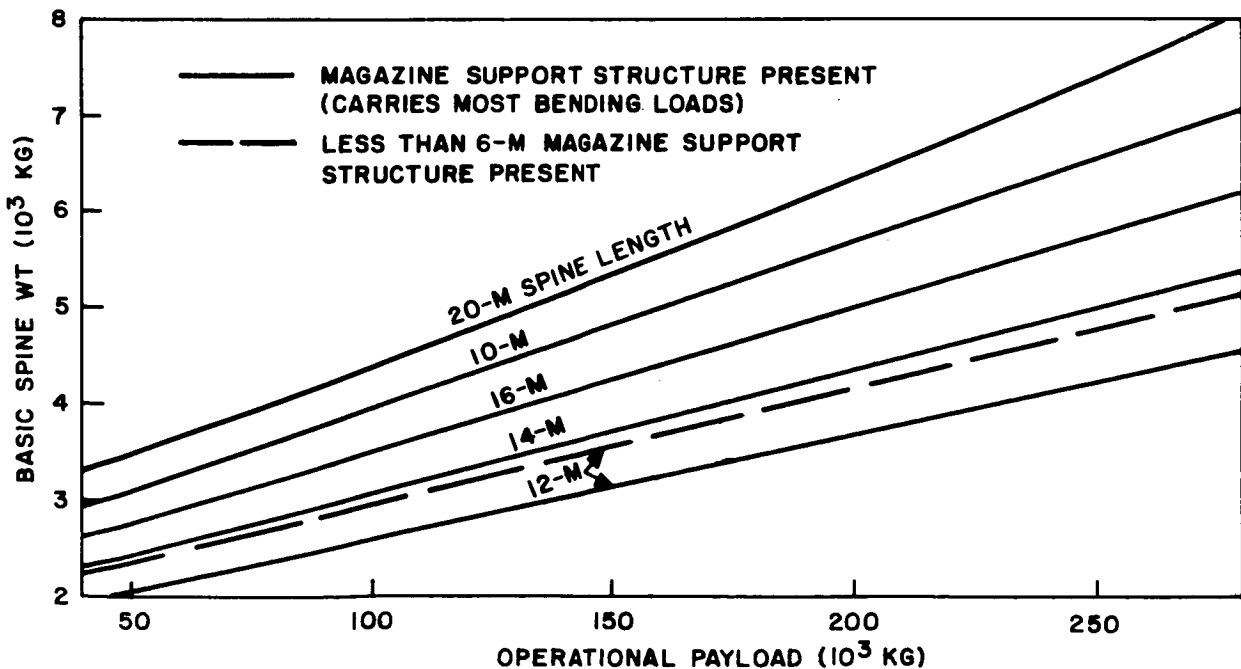
Fig. 3.10--Payload spine weights as functions of payload carried
and length of spine for 10-m configurations

3.2.2. 10-m Exploration Configurations

A typical over-all exploration vehicle configuration using the basic 10-m propulsion module is shown in Fig. 3.11. The vehicle is shown ready to depart earth orbit on a 450-day Mars orbital-capture mission, which requires a velocity increment of 72,850 fps for the complete round trip. This configuration carries only 750 kg of destination payload (mapping equipment plus a data-handling and storage system) and therefore will not support a Mars landing.

The two space taxis, although primarily intended for liaison between vehicles in a two-vehicle convoy, are carried on each vehicle. On most of the configurations they are located atop the personnel accommodations compartment where entrance airlocks protrude from the access-ways. Placing them at this higher location avoids using space required for propellant magazines or external payload on the more heavily loaded vehicles.

The reentry vehicle shown in the illustration is assumed to be capable of a successful earth reentry from an approach speed of 50,000 fps. For this capability it carries 3,000 kg more structure, ablative material, etc., than a similar reentry vehicle intended for a 36,300-fps approach (approximately Apollo reentry speed). Below the reentry vehicle is a stubby maneuver stage (capable of approximately 1,000-fps ΔV) which is attached to the vehicle and used to initiate the reentry.

The upper vehicle in Fig. 3.11, i.e., everything above the propulsion module (except the propellant magazines and the magazine support structure) comprises the mission's operational payload. A weight breakdown of this operational payload is given in Table 3.3. This weight summary applies to all of the 8-man, nominally 450-day-mission vehicles, to which must be added to the proper weight for a reentry vehicle, when required. A 50,000-fps earth-approach condition requiring a 7,000-kg reentry vehicle was assumed for the configuration in Fig. 3.11, resulting in a total operational payload of 80,000 kg.

The operational payload includes the basic 12-m-long payload spine. When a longer spine was required, an incremental spine weight was added to the weight of the propulsion module along with the incremental weight for the magazine support structure. This procedure prevented possible confusion due to slightly varying operational payloads, although the propulsion module weight is caused to vary somewhat more than required by propellant demands. (The basic 10-m propulsion module, below the spine and magazine support structure, was not varied at all for the variety of configurations designed.)
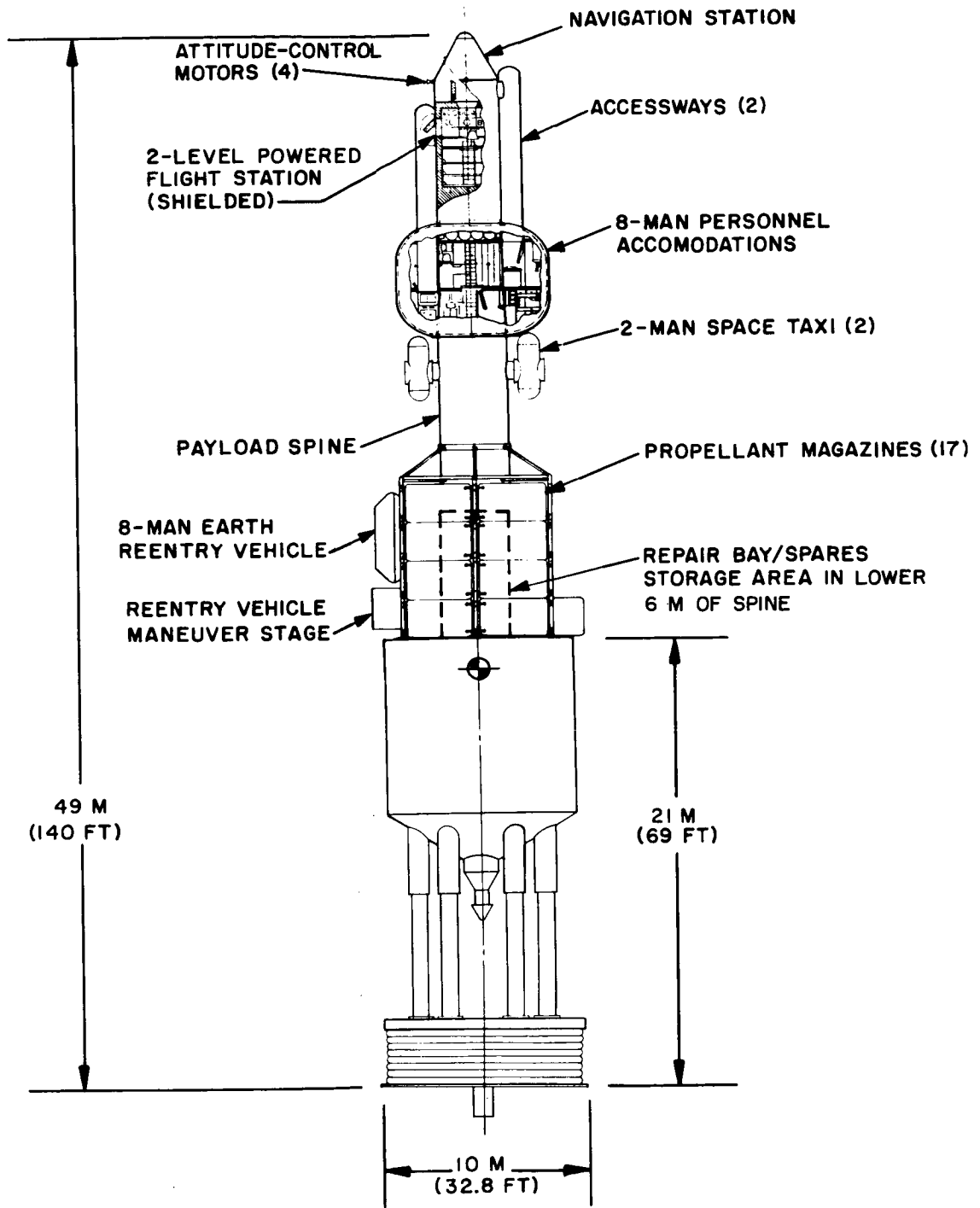
Fig. 3.11--Exploration vehicle configuration for Mars orbital capture mission using 10-m propulsion module

Table 3.3

WEIGHT SUMMARY OF OPERATIONAL PAYLOAD
(Payload A or B, 8-man, 450-day mission)
(In kilograms)

Life-support system net dry weight
  Structure
    Powered flight station—escape vehicle 1,730
    Basic 12-m-long spine 2,270
    Personnel accommodations section 7,600
      Total 11,600

  Equipment not listed elsewhere
    Furnishings 2,400
    Main power supply 3,470
    Emergency power supply 1,065
    Communications system 136
    Guidance, autopilot, and navigation system 363
    Vector control motors, tanks, and system 680
    Spin system tankage and nozzles 386
    Abort propulsion system 600
      Total 9,100
          Total life-support-system weight 20,700

Radiation shielding (nominal) 18,170

Ecological system and food
  Hardware
    Main system 1,054
    Emergency and reserve life-support system 1,923
      Total 2,977

  Expendables
    Main ecological system 3,515
    Emergency and reserve life-support system 3,810
    Food 5,398
      Total 12,723
          Total ecological-system weight 15,700

Reentry vehicle and maneuver stage (as required)
  For ~36,300 fps approach velocity . . . . (4,000)
  For ~50,000 fps approach velocity . . . . (7,000)

Abort propellant (nominal case) 4,500

Spin propellant (nominal) 4,540

48

Table 3.3 (continued)

Space-taxi capsule (2 men each)
Hardware 625
Propellant 825
1,450

Spares and repair equipment
Repair and maintenance equipment 1,130
Spares 2,270
3,400

Checkout instrumentation 500

Personnel (eight) 725
          Total, less contingency and with no reentry vehicle . . . 69,685

Contingency (~5 percent) 3,315
          Total (without reentry vehicle) 73,000

          Total with 36,300 fps reentry vehicle 77,000

          Total with 50,000 fps reentry vehicle 80,000

Complete vehicle configurations for a variety of exploration payloads are shown in Fig. 3.12. The configuration second from the left is the same one shown in Fig. 3.11. To its right is a vehicle with the same operational payload, but with a 75,000-kg destination payload to permit planetary landing operations. Three 2-man Mars excursion vehicles are assumed with a gross weight of 16,000 kg each. These weights and weights of the other destination payload items were taken from previous work by General Dynamics/Astronautics.[*] It was assumed for this study that the 2-man Mars excursion modules could be packaged in 3.2-m (10.5-ft) diam, 6.1-m (20 ft) long canisters by having landing legs, protruding arm, etc., folded for transport. The canisters then provide a measure of protection for the destination payload and are conveniently carried on the same support columns that carry the propellant magazines and the reentry vehicle if used.

It can be seen in Fig. 3.12 that the nominal 12-m spine length is only exceeded for the largest payload configuration shown. The over-all length increase for the smaller 20-man configuration is due to the larger personnel accommodations.

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[*]K. A. Ehricke, Study of Interplanetary Missions to Mercury through Saturn with Emphasis on Manned Missions to Venus and Mars 1973/82 Involving Capture, General Dynamics/Astronautics, Report GDA 63-0916, September, 1963.
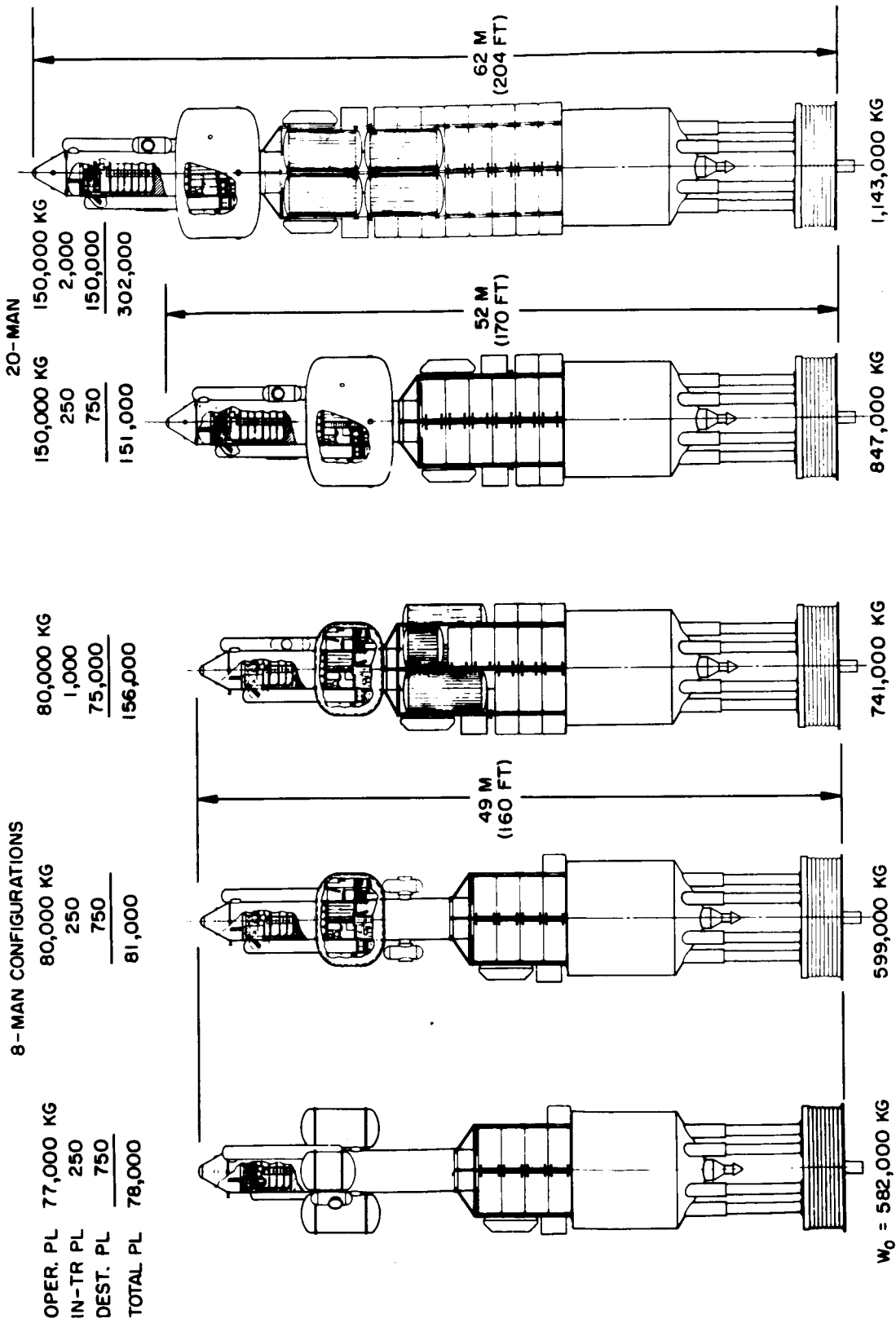
Fig. 3.12--Various payload configurations on basic 10-m propulsion module (with departure weights for 72,850 fps Mars mission)

Summaries of the payload breakdowns and the over-all weights
for the vehicles shown were given in Sections 2 and 3 of Vol. II. The
variation in propellant loadings for widely different mission velocity
requirements (from approximately 50,000 to nearly 100,000 fps) but with
a constant payload were also given in Vol. II. The significant point here
is the wide variety of both payloads and mission velocities than can be
accommodated while using the same 10-m propulsion module and per-
forming the entire round trip in one stage.

3.2.3. 20-m Configurations

The 20-m (diam) configurations, when designed for exploration
missions, are very similar in arrangement to the 10-m exploration vehicles
just discussed. This similarity is shown in Fig. 3.13, which compares a
20-m vehicle designed for a 20-man, 910-day round trip to Jupiter with a
10-m vehicle configured for an 8-man, 450-day round trip to Mars.

The capability of the 20-m vehicles is much greater than that to
be expected of a dimensionally scaled-up 10-m design, because of the
higher specific impulse of the larger propulsion module (3,150 sec compared
with 1,850 sec for the 10-m modules of this study). For this reason the
larger module is necessary to attain in one stage the >200,000-fps velocity
required for missions to Jupiter.

The structural design concepts for the 20-m vehicles are quite
similar to those of the 10-m designs. The payload spine weights were
estimated and plotted versus the operational payload for various lengths
as shown in Fig. 3.14. Structural weights and ecological systems were
estimated by the same methods used for the 10-m designs. Summaries
of payload weights were given in Section 2 of Vol. II.

The 20-m propulsion modules exhibit the same kind of payload
and mission velocity versatility as that demonstrated for the 10-m designs.
This is shown in Fig. 3.15. Exploration configurations are shown for
20-man and 50-man missions with destination payloads varying from
100,000 kg to 1,500,000 kg. An unmanned one-way logistic vehicle for
Mars orbit delivery is also shown carrying 3,000,000 kg of payload. This
vehicle is in many respects a large-size version of the logistic vehicles
for lunar deliveries, which are discussed in some detail in the following
section.

3.3. LUNAR FERRY VEHICLES

Two operational concepts were investigated for lunar ferry
vehicles. One operates from earth orbit to lunar orbit and uses separate

POWERED
FLIGHT
STATION
(SHIELDED)

4-MAN SPACE TAXI (2)

20-MAN PERSONNEL
ACCOMODATIONS

100,000 KG
DESTINATION
PAYLOAD

73 M
(240 FT)

PROPELLANT
MAGAZINES (46)

REPAIR
BAY/SPARES
STORAGE AREA
IN LOWER 10-M
OF SPINE

10-M MARS
LANDING
EXPLORATION
CONFIGURATION
(8-MAN)
FOR SIZE REF

32 M
(105 FT)

BASIC 20-M
PROPULSION
MODULE

20 M
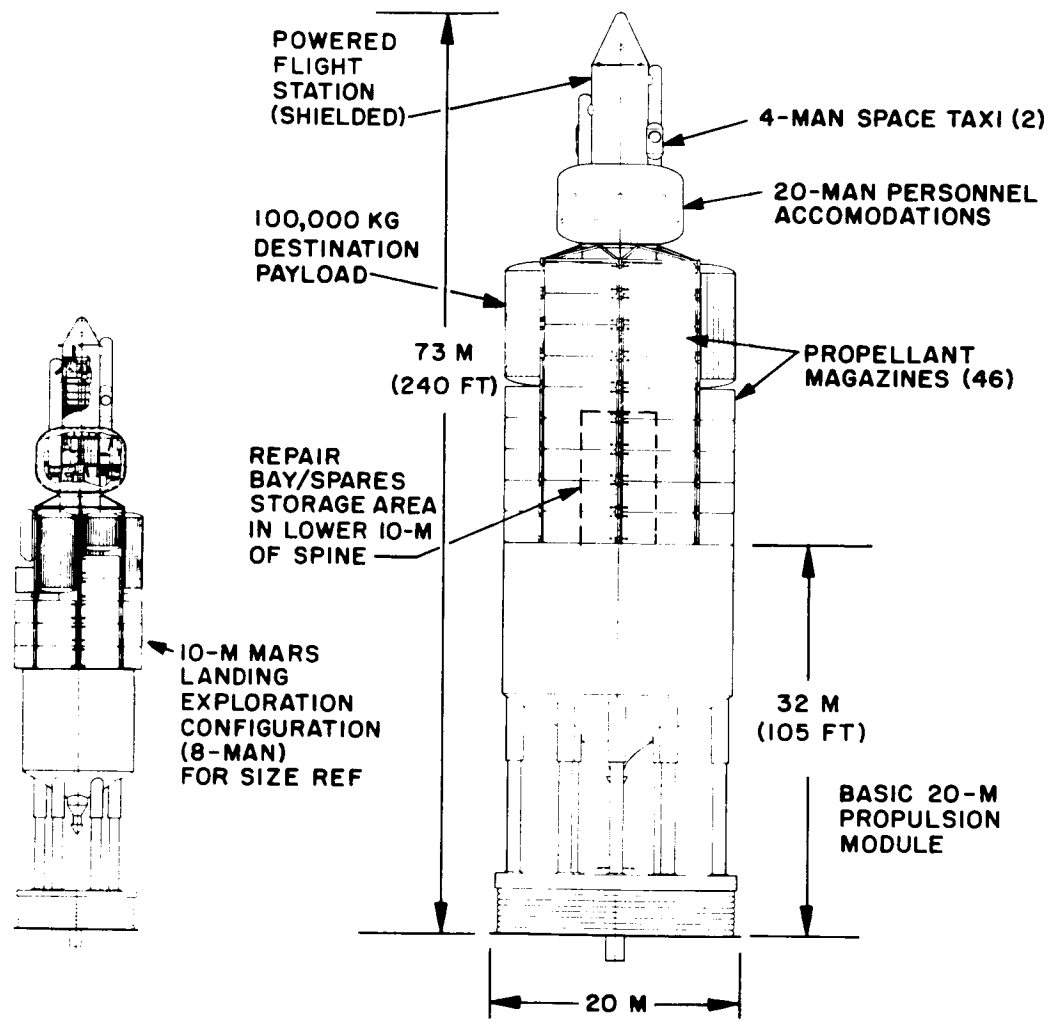
Fig. 3. 13--Exploration vehicle configuration for Jupiter moon
landing mission, 20-m propulsion module
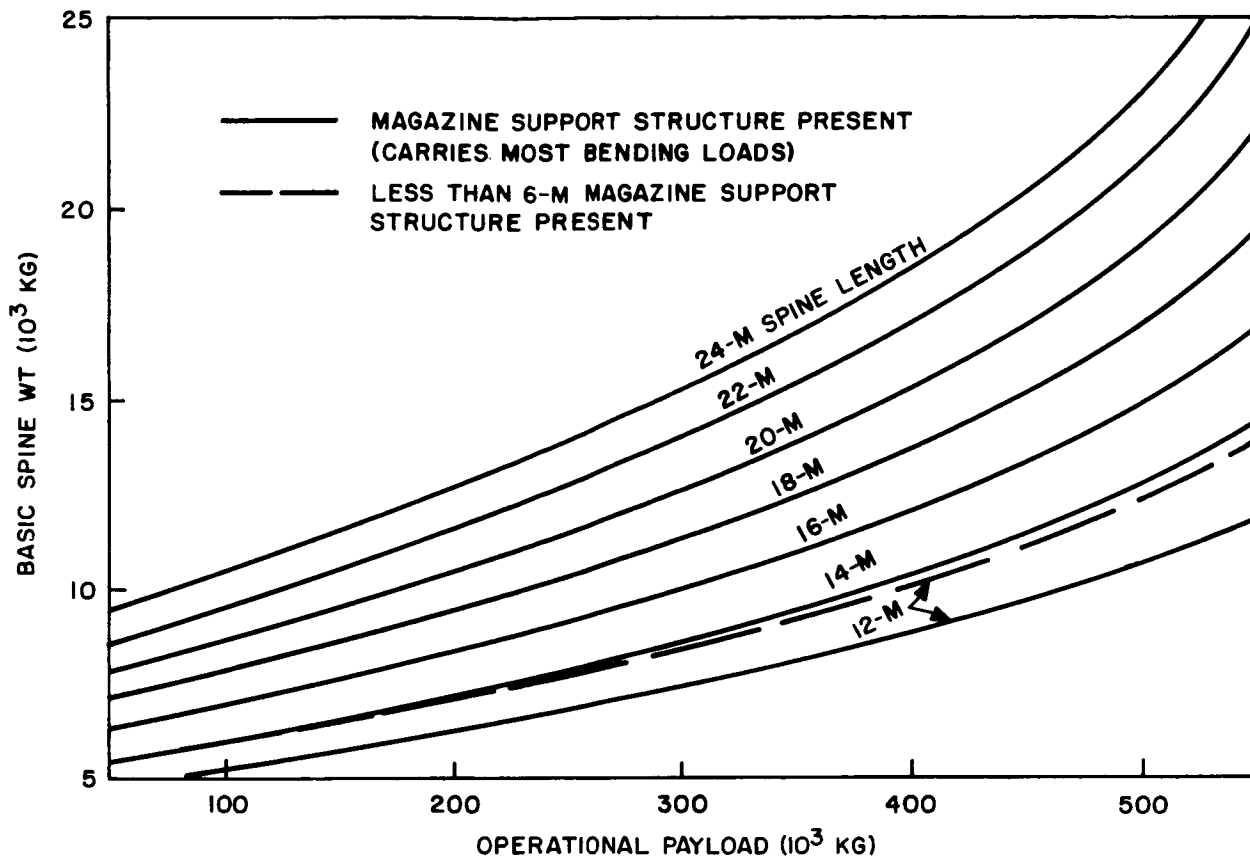
Fig. 3.14--Payload spine weight as function of payload carried and length of spine for 20-m configurations

chemically propelled lunar shuttle vehicles for lunar orbit-to-surface support. The other operates from earth orbit to the lunar surface and uses a built-in chemical rocket system for lunar landing and ascent from the lunar surface.

3.3.1. Earth-orbit-to-lunar-orbit Ferry Vehicle

The vehicle employed in the earth-orbit-to-lunar-orbit ferry operation is illustrated in Fig. 3.16. The system above the basic 10-m propulsion module includes a magazine section, the cargo modules, the command-module adapter section, the command module proper, and the passenger modules.

The magazine section is built up with a central cylindrical spine, circumferentially located support columns with appropriate truss member to the spine, and a payload support structure to distribute the loads from the spine and columns to the payload modules. The estimated weight of the support structure as a function of payload is given in Fig. 3.17.

53
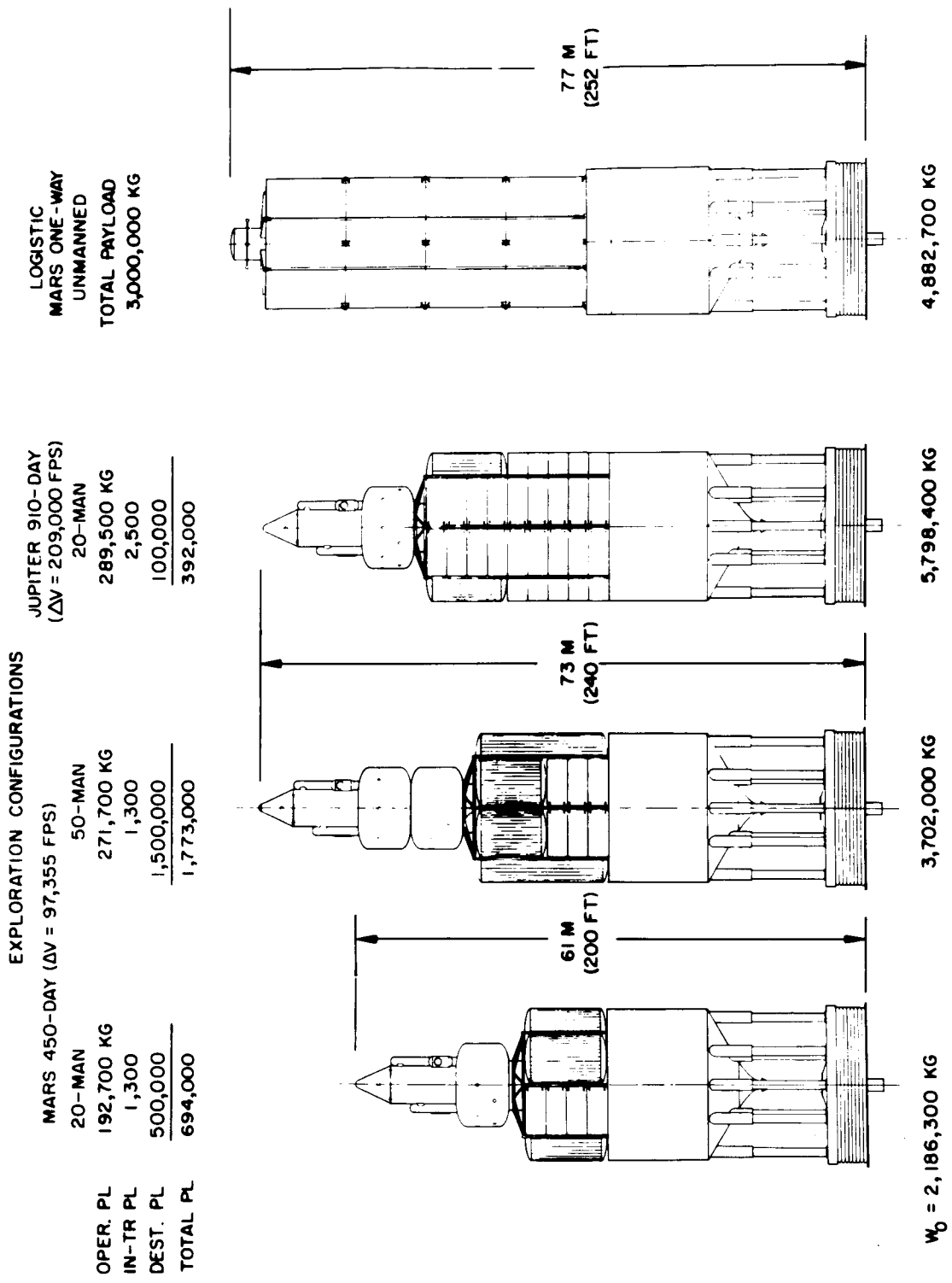
Fig. 3.15--Various payload configurations on basic 20-m propulsion module (with departure weights for given missions)
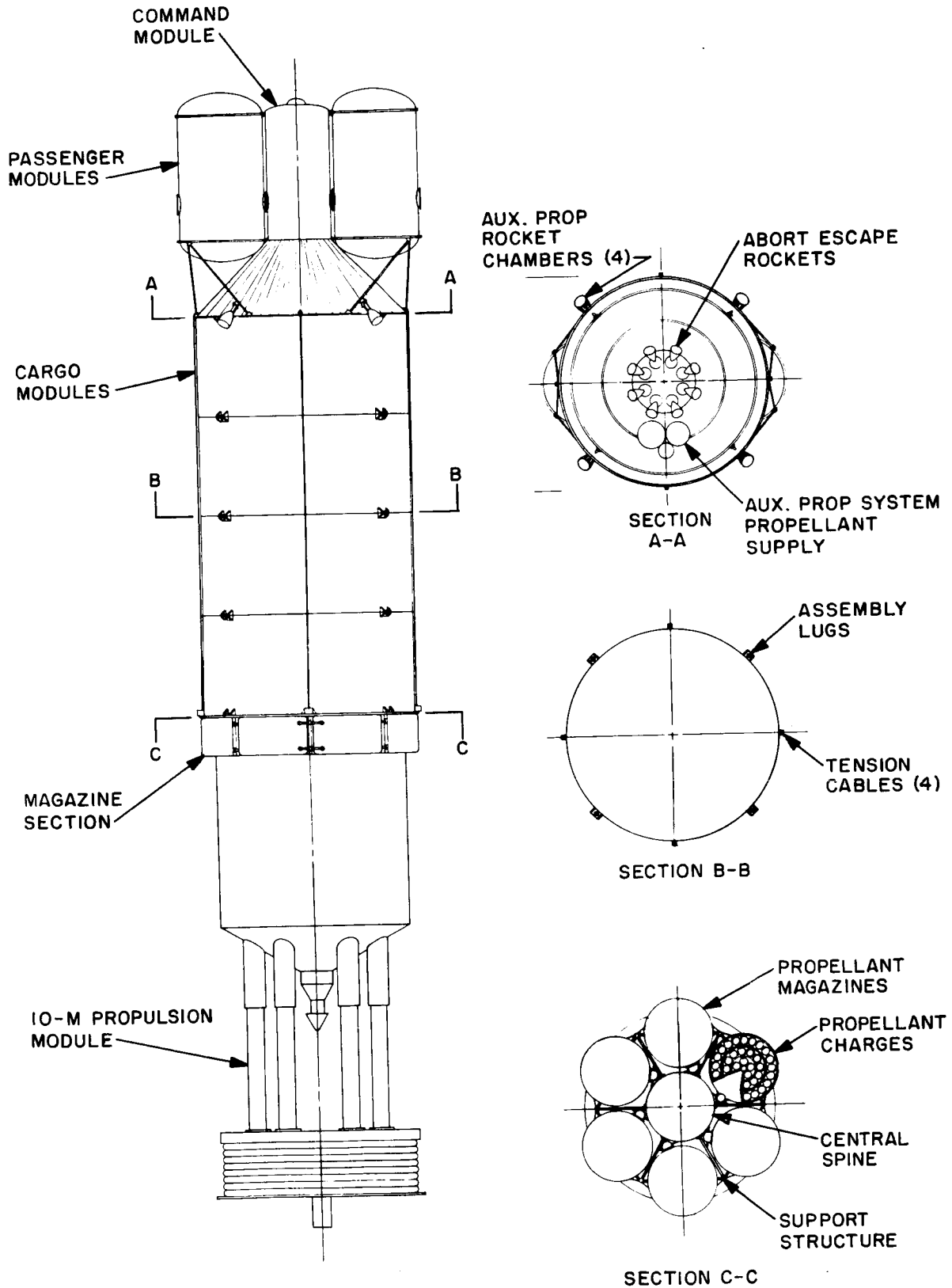
Fig. 3.16--Earth-orbit-to-lunar-orbit ferry vehicle
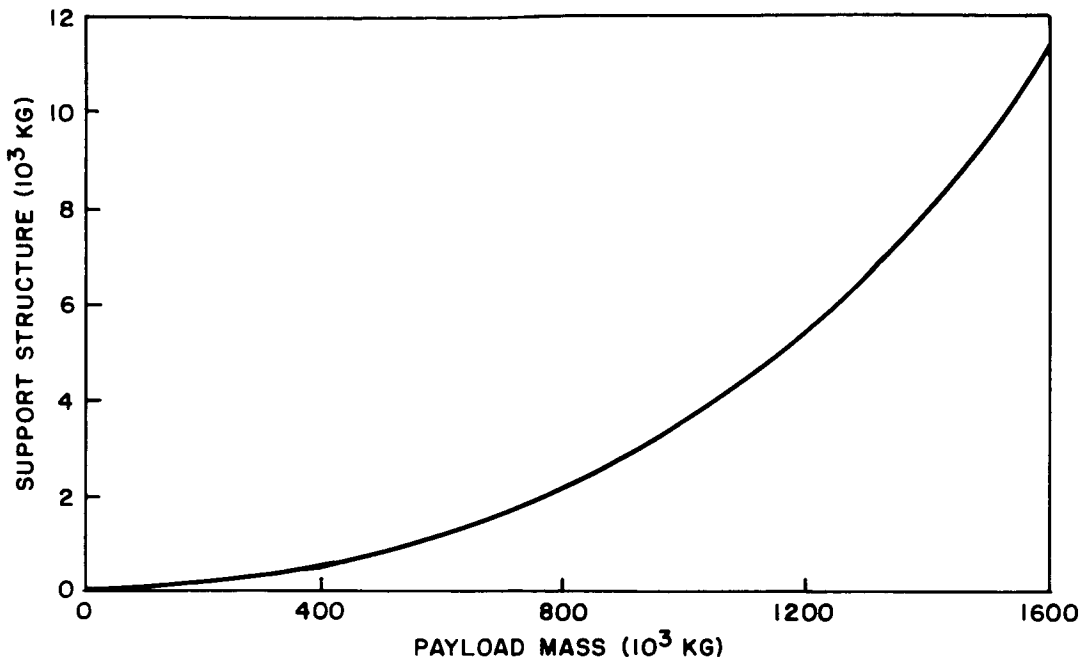
Fig. 3.17--Estimated support-structure mass

The payload modules are designed to be compatible with a Saturn V earth-orbit delivery system and to fulfill the requirement for an easily transferred and assembled system. Each module is 10 m in diam, the same as the Saturn V, and approximately 4.7 m long. The length was based on a cargo mass of 100,000 kg (220,500 lb; Saturn V orbit capability) and an average payload density of 272 kg/m (20 lb/ft^3; composite density of O_2/H_2 propellant at a mixture ratio of 5:1). The cargo modules are built-up aluminum containers designed to carry the full acceleration and bending loads of the vehicle. Each is fitted with four assembly-lug half-sections used for mating the cargo modules and system elements. In addition, four tension-wire cables are provided to carry bending tensile loads. These relieve the assembly lugs, thereby avoiding point loads and permitting construction of lightweight modules.

The command-module adapter supports the command module and accommodates the auxiliary-propulsion system, tension-cable take-up system, and other miscellaneous systems. It is a conical aluminum sheet and stringer structure with internal ring stiffeners. The auxiliary pro-pulsion system used for thrust vector correction and vernier velocity requirements is a pressure-fed nitrogen tetroxide and 50% hydrazine/50% unsymmetrical dimethylhydrazine (UDMH) system with four high-expansion-ratio rocket motors. Each motor has a thrust of approximately 5,000 newtons (1,120 lb).

The command module (Fig. 3. 18) is designed to accommodate a crew of three in the upper-deck flight control station and to provide living quarters for the crew in the lower deck. The upper compartment was sized on the basis of 5 m^2 (50 ft^2) for the crew of three. The lower compartment is sized on the basis of 5 m^3 (180 ft^3) per man, assuming that no more than two of the crew are off duty at one time.
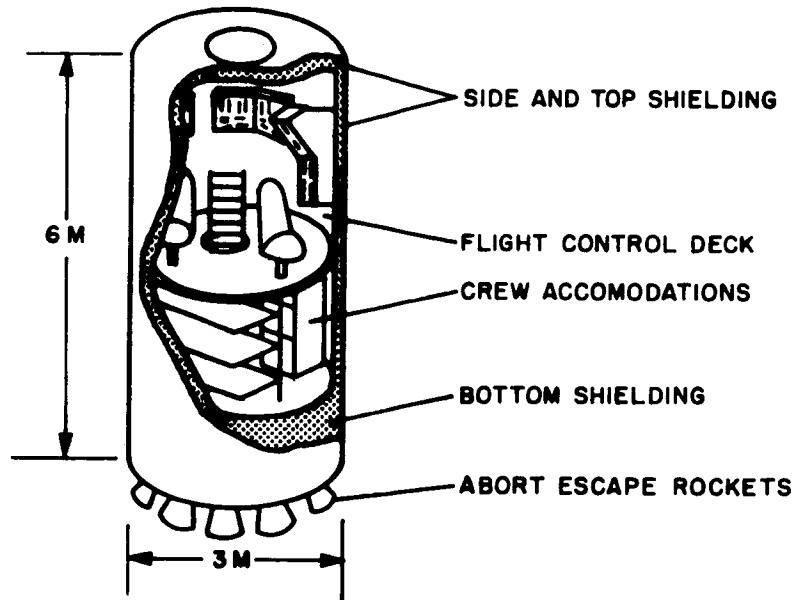
Fig. 3. 18--Lunar-ferry-vehicle command module

The command module is shielded to provide protection for the crew and passengers during earth radiation-belt penetration, operation of the nuclear-pulse system, and solar-flare activity. In the given configuration, the lower deck can accommodate 20 passengers during periods when shielded protection is required. For larger passenger requirements, the lower compartment is scaled up on a linear basis.

The command module is a completely integrated unit capable of supporting the crew on a continuous basis and the passengers on an emergency basis. The component weights for the command module are summarized in Table 3. 4.

The shielding requirements to limit an integrated dose during the nuclear-pulse firing to 50 rem were estimated to be

Bottom: $\begin{cases} 110 \text{ g/cm}^2 \text{ hydrogenous material (neutron),} \\ 55 \text{ g/cm}^2 \text{ lead or equivalent (gamma).} \end{cases}$

Side and top: 25 g/cm^2 hydrogenous material (neutron).

Table 3.4

WEIGHT SUMMARY OF LUNAR FERRY COMMAND MODULE
(In kilograms)

Shielding	22,380
Structure	170
Guidance and control	400
Power supply	350
Instrumentation and communication	450
Crew	300
Crew support	550
Life support	300
Abort propulsion	800
Miscellaneous systems	500
	26,200
Contingency (5 percent)	1,310
Total mass	27,510

For design purposes, polyethylene was selected for neutron shielding and depleted uranium for gamma. The corresponding shield thicknesses are

Bottom: $\begin{cases} \text{Polyethylene} & 110 \text{ cm} \\ \text{Lead} & 29 \text{ cm} \end{cases}$

Side and top: Polyethylene 25 cm

The weight of the abort propulsion system was based on a nominal requirement of 3 g's for 3 sec and a solid-propellant specific impulse of 270 sec. For the power supply, a fuel-cell system similar to that being developed for Apollo was assumed. The crew support allowance includes space suits, tools, and personal gear utilized by the crew.

The incremental command-module mass, including allowances for shielding, structure, and abort system for extra passengers (above 20), was estimated to be 114 kg/man.

A standard reference-design passenger module (Fig. 3.19) was adopted to determine the mass of the system for accommodating passengers. This module is an integrated unit which carries 10 passengers. It is not shielded for nuclear pulse and other high flux radiations since the passengers can be accommodated in the command module during such encounters. The reference module is a two-deck structure, one for a sleeping compartment and one for work, exercise, eating, and recreation activities. A volume of 5 m^3 (180 ft^3) is provided for each passenger.
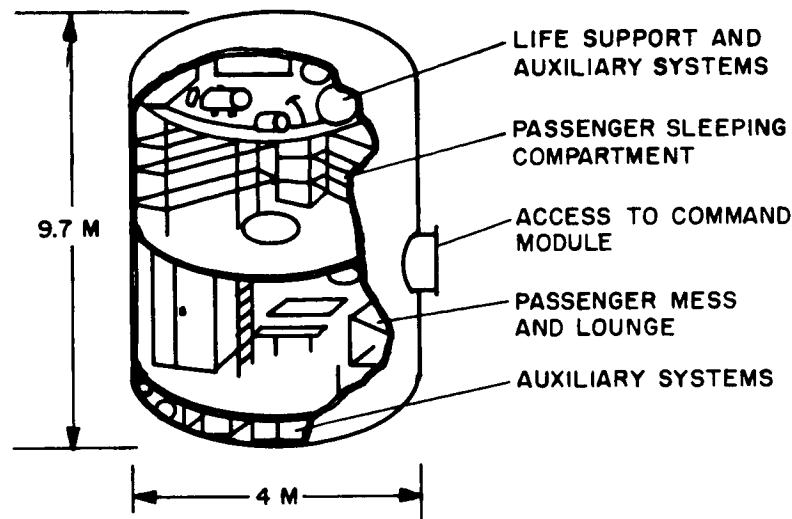
Fig. 3.19--Reference-design passenger module

The weight breakdown for the reference passenger module is presented in Table 3.5. The life-support system selected is a regenerative system. The basic wall is a double aluminum wall, foam-filled honeycomb structure design to accept an internal pressure of 7 psi, with either wall carrying the entire load.

For performance calculations, masses of even multiples of the reference module were used for the number of passengers of interest.

Table 3.5

WEIGHT SUMMARY OF LUNAR FERRY PASSENGER
MODULE (10-MAN CAPACITY)
(In kilograms)

Structure	1,480
Life support	1,200
Personnel	1,000
Personnel support	500
	4,180
Contingency (5 percent)	210
Total mass	4,390

3.3.2. Earth-orbit-to-lunar-surface Ferry Vehicle

The vehicle concept utilized in the earth-orbit-to-lunar-surface ferry vehicle is shown in Fig. 3.20. This vehicle is similar to the

COMMAND MODULE

CARGO MODULES

ROCKET CHAMBER DOORS (4)

LANDING MODULE

10-M PROPULSION MODULE

LANDING GEAR

AUX. PROP. ROCKET CHAMBERS (4)

ABORT ESCAPE ROCKETS

SECTION A-A

AUX. PROP SYSTEM PROPELLANT SUPPLY

ASSEMBLY LUGS

TENSION CABLES (4)

SECTION B-B

PRESSURIZATION SUPPLY

LH2 TANK

RETRACTABLE LANDING ROCKET CHAMBERS (4)

LO2 TANK

SECTION THROUGH LANDING MODULE
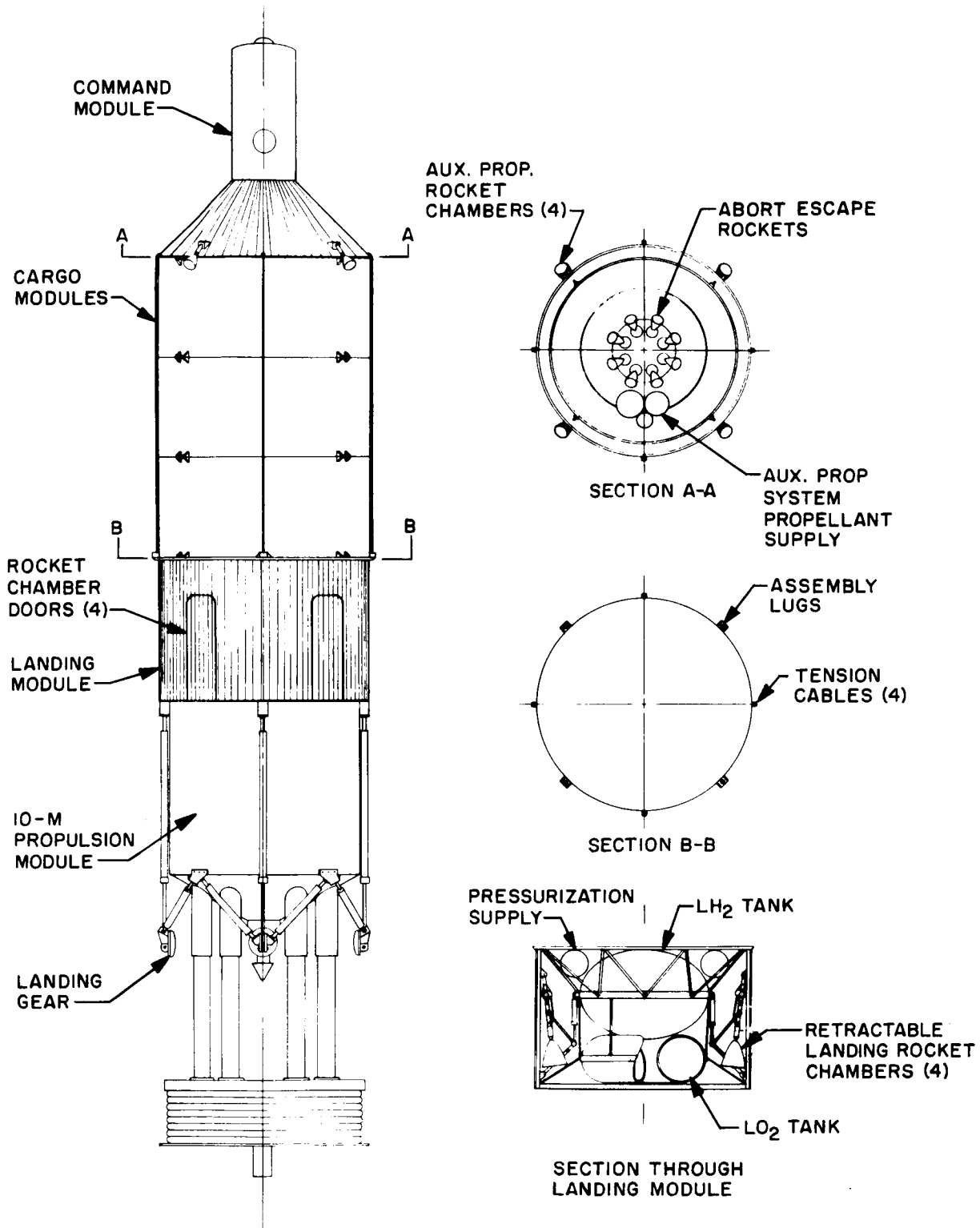
Fig. 3.20--Earth-orbit-to-lunar-surface ferry vehicle

orbit-to-orbit ferry except that it is provided with a chemical-propulsion landing module and landing gear for landing the entire ferry vehicle. No shuttle system is required since the payload is landed by the ferry vehicle.

The landing module incorporates an oxygen-hydrogen propulsion system. The fuel is accommodated in an ellipsoidal tank and the oxidizer in a toroidal tank. Four pump-fed rocket chambers are located at the periphery of the module. These engines are stowed in a retracted position and protected by doors during firing of the nuclear-pulse system. Upon activation they are extended to a position outside of the propulsion module cylinder with a cant of 30 degrees by means of a hydraulically actuated linkage system. The engines are assumed fully throttleable so that hovering and controlled descent can be accomplished. A specific impulse of 430 sec was utilized.

The parametric weight of the landing propulsion module was estimated to be

$$M_{LM} = 0.05 \, (M_{\lambda G} + M_{NPM} + M_{CM}) + 0.14 \, M_P \, ,$$

where M_{LM} = landing-module burnout mass,

$M_{\lambda G}$ = gross payload mass,

M_{NPM} = nuclear-pulse-module mass,

M_{CM} = command-module mass,

M_P = propellant mass.

Basically, the coefficient for $(M_{\lambda G} + M_{NPM} + M_{CM})$ is an allowance for landing gear, landing-gear support structure, and other payload-dependent requirements such as landing-engine mass. The coefficient for M_P provides for tankage, propellant feed system, and the basic module structure.

3.3.3. Lunar Shuttle Vehicles

Conceptual designs of the lunar shuttles associated with the orbit-to-orbit ferry system were generated in order to establish bases for estimating the performance of these components. To be consistent with the passenger-accommodations principle applied to the ferry system, the passenger shuttle was designed to accommodate 20 passengers, and it was assumed that each shuttle could make as many as two trips for every trip made by the lunar ferry. The cargo shuttle was scaled to transport the entire cargo load of the ferry in one trip. (Use of a standardized cargo shuttle would not significantly affect the results of this study.)

3.3.3.1. <u>Lunar Passenger Shuttle</u>. The passenger shuttle
concept is illustrated in Fig. 3.21. It consists of three basic units: the
passenger module, command cockpit, and propulsion module.

DOCKING HATCH

CREW COCKPIT

PASSENGER
COMPARTMENT

PROPULSION
MODULE

LANDING GEAR
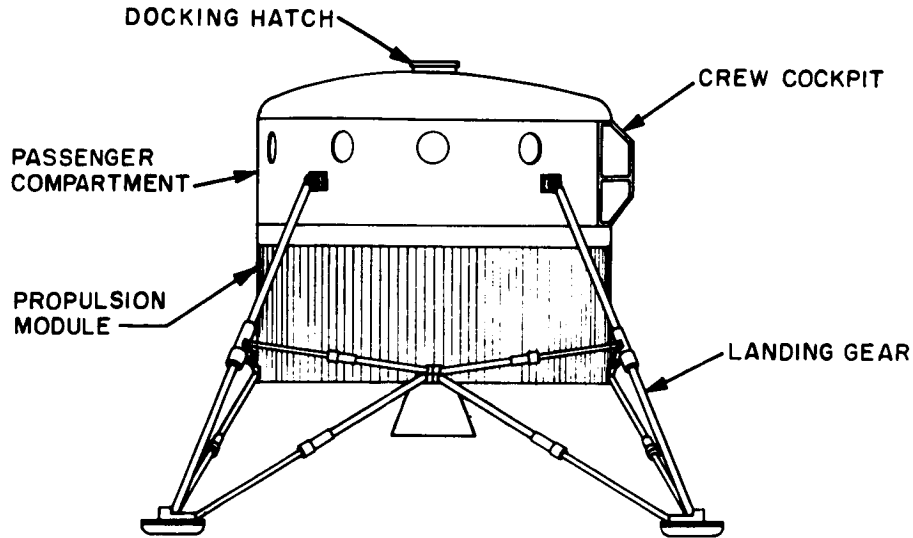
Fig. 3.21--Lunar passenger ferry

The weight summary for the passenger shuttle is given in
Table 3.6. The passenger-cabin size was based on a requirement of
2.5 m^3 (88 ft^3) per passenger. Assuming a single passenger deck, this
results in a cabin diameter of approximately 6 m (20 ft). An open-loop
life-support system was selected since the total time and frequency of use
cannot justify a more complex system. The corresponding system mass
requirements are 11.5 lb/man-day, including fixed container weights.
The passenger support allowance is based on an estimate of 50 kg/man
for space suits and personal gear.

A breakdown of the command cockpit weight is given in Table 3.7.
A crew of two was assumed. The command cockpit is a completely inte-
grated system which can be operated independently of the passenger cabin.

The propulsion module employs oxygen and hydrogen propellants
with a specific impulse of 430 sec. It is provided with a single pump-fed
throttleable main propulsion engine. In addition, auxiliary chambers are
provided for attitude control. The propulsion-module mass also includes
an allowance for landing gear. An ellipsoidal tank is used for the hydrogen
and a toroidal tank for the oxygen. The propellant tanks were sized only
for landing requirements on the assumption that propellant for ascent would
be tanked at the lunar base.

62

Table 3.6

WEIGHT SUMMARY OF LUNAR PASSENGER SHUTTLE
(In kilograms)

Passenger cabin	1,500
Life-support system	80
Passengers (20)	2,000
Passenger support	1,000
Command cockpit	3,350
Total	7,930
Contingency (5 percent)	400
Total passenger module	8,330
Propulsion module	1,610
Total weight	9,940

Table 3.7

WEIGHT SUMMARY OF COMMAND COCKPIT
(In kilograms)

Structure	1,000
Power supply	300
Guidance and control	400
Communications	300
Instrumentation	150
Life support	200
Crew	200
Crew support	400
Miscellaneous systems	400
Total weight	3,350

3.3.3.2. Lunar Cargo Shuttle. The lunar cargo shuttle is designed as a manned system used to transfer cargo from the lunar ferry in lunar orbit to the lunar base. The basic design concept for this vehicle is illustrated in Fig. 3.22.

In this vehicle a two-man command cockpit is provided in the propulsion section. The mass for this system is the same as for the passenger shuttle presented in Table 3.7, except that with a separate contingency of 5 percent, the total mass estimated is 3,520 kg (7,760 lb).

The basic vehicle size is a function of cargo mass and propellant capacity:

$$M_{BO} = 3,520 + 0.05 \, M_\lambda + 0.14 \, M_P,$$

CARGO MODULES

CREW COCKPIT

PROPULSION MODULE (O_2/H_2)

LANDING GEAR
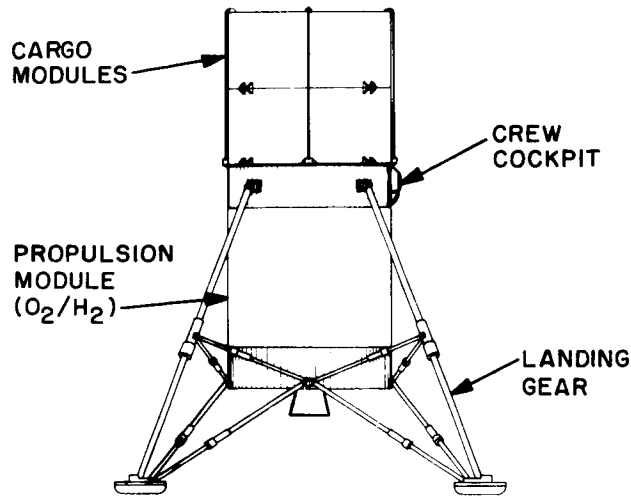
Fig. 3.22--Lunar cargo shuttle

where M_{BO} = burnout mass, kg,

M_λ = payload mass, kg,

M_P = propellant mass, kg.

As in the case of the passenger shuttle, oxygen and hydrogen propellants were utilized, and it was assumed that the cargo shuttle would be tanked prior to descent and prior to ascent. An engine and a tank arrangement similar to those on the passenger shuttle are also utilized with the cargo shuttle.

3.4. LUNAR LOGISTIC VEHICLES

Two operational concepts were investigated for expendable lunar logistic systems: an earth-launched system and an earth-orbit-launched system.

3.4.1. Earth-launched Lunar Logistic Vehicles

Solid-propellant-motor-boosted and liquid-propellant-rocket-boosted systems were investigated in this study of earth-launched lunar logistic vehicles. The solid-propellant systems are used to provide a low-velocity increment, whereas the liquid-propellant system is used for a high velocity boost.

Two solid-boosted configurations are shown in Fig. 3.23. A cluster of six solid-propellant motors with a total thrust of 1,205,000 kg (2,660,000 lb) provides a launch thrust-to-weight ratio of 1.5. These
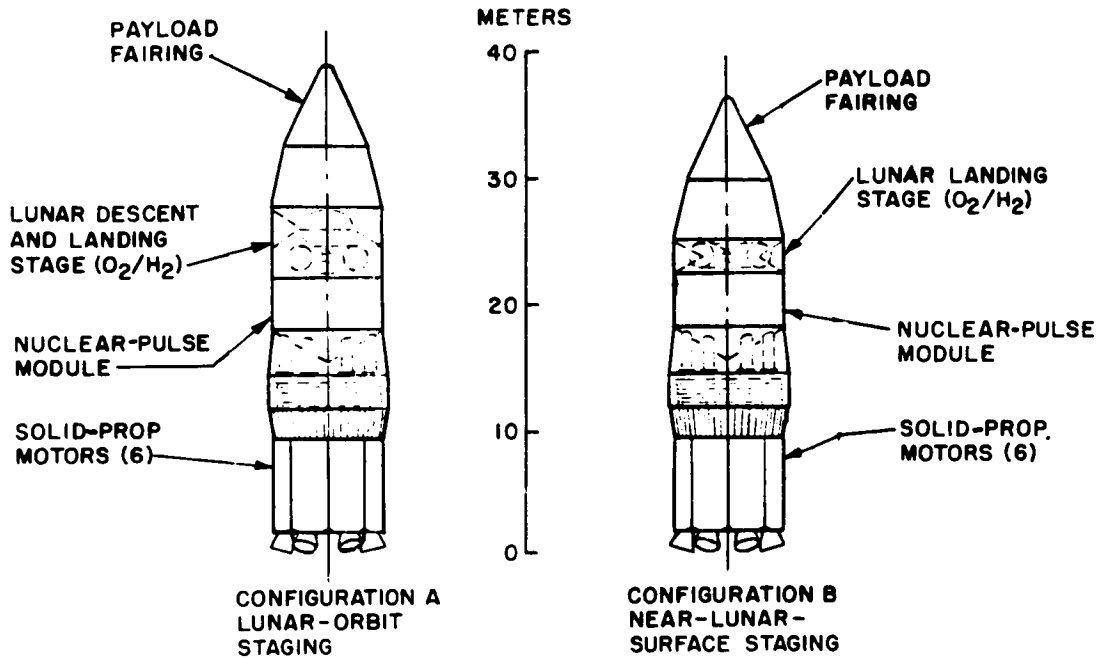
Fig. 3.23--Solid-propellant-boosted, earth-launched
lunar logistic vehicles

motors, which have an average specific impulse of 247 sec each, have a
usable propellant mass of 72,500 kg (160,000 lb). The burnout mass of
the motors, cluster structure, interstage adapter, and retrorockets was
estimated to be 370,000 kg (816,000 lb).

With configuration A, operation of the nuclear pulse propulsion
module is terminated in lunar orbit; with configuration B, operation is
terminated at near-lunar surface. Configuration B requires only a
relatively small number (145) of pulse units in excess of the 900-unit
capacity of the reference-design propulsion module, so it was assumed
that the propulsion module could be enlarged slightly to accommodate the
extra units; therefore, an incremental weight allowance of 747 kg (1,648 lb)
was made.

Both configurations utilize oxygen-hydrogen stages for lunar
landing: from lunar orbit for configuration A; from near-lunar surface
for configuration B. These stages were sized for velocities of 8,119 ft/sec
and 3,320 ft/sec, respectively, including a 5 percent performance reserve
(see Section 2, Vol. II). A specific impulse of 430 sec was used for the
oxygen-hydrogen propellants. The jettison weights of these stages were
derived as follows:

$$M_J = 700 + 0.14\,M_P + 0.5\,M_\lambda \quad \text{(Configuration A)},$$

$$M_J = 700 + 0.12\,M_P + 0.5\,M_\lambda \quad \text{(Configuration B)},$$

where M_J = jettison (burnout) mass of stage, kg,

M_P = propellant mass, kg,

M_λ = payload mass, kg.

In addition, it was assumed that an interstage adapter which might be constructed as an integral part of the stage basic structure would be severed and jettisoned with the nuclear-pulse module. The mass of this additional structure was estimated to be 300 kg (662 lb).

The S-IC, the first stage of the Saturn V launch vehicle, was selected as the basis for evaluating the use of a liquid-propellant booster with the 10-m nuclear pulse propulsion module. The over-all launch vehicles using the S-IC are shown in Fig. 3.24. The jettison weight of the S-IC, including retrorockets, residual propellants, and interstage adapter, as well as the dry S-IC, was estimated to be 185,000 kg (409,000 lb). The average specific impulse of the S-IC was assumed to be 280 sec.

As in Configuration B of Fig. 3.23, a few extra pulse units above the nominal 900 were required for both Configurations A and B in the S-IC-boosted arrangements. Configuration A requires an extra 102 units and Configuration B requires 24. The corresponding weight increment allowances were 398 kg (880 lb) and 1,690 kg (3,730 lb), respectively.

The lunar landing stages were based on the same scaling equations as those presented above for the solid-boosted configurations. The mass of the jettison interstage structure and specific impulse were also the same.

3.4.2. Orbit-launched Lunar Logistic Vehicle

The earth-orbit-launched lunar logistic vehicle using the 10-m nuclear pulse propulsion module is illustrated in Fig. 3.25. In this concept, additional magazines and the associated payload—landing-stage support structure are required for all but the smallest payload design requirements. An oxygen-hydrogen landing stage is located above the magazine section. Cargo modules identical to those described for the lunar ferry vehicles are used to accommodate the cargo. The auxiliary propulsion system similar to that described for the lunar ferry (Section 3.3.1) is mounted in a forward module. The guidance and control system is also located in this section.

66
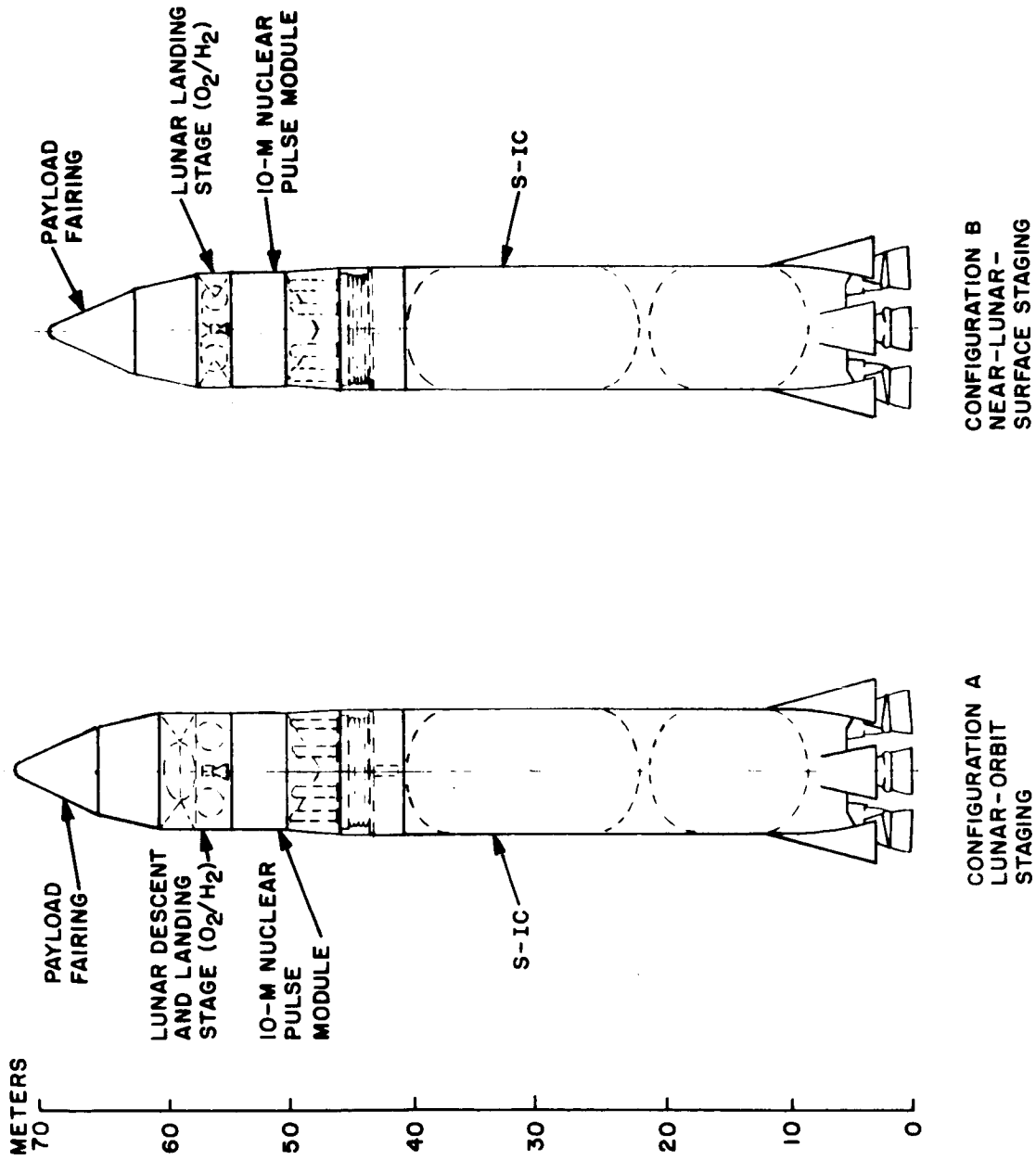
Fig. 3.24--S-1C boosted earth launched lunar logistic vehicle
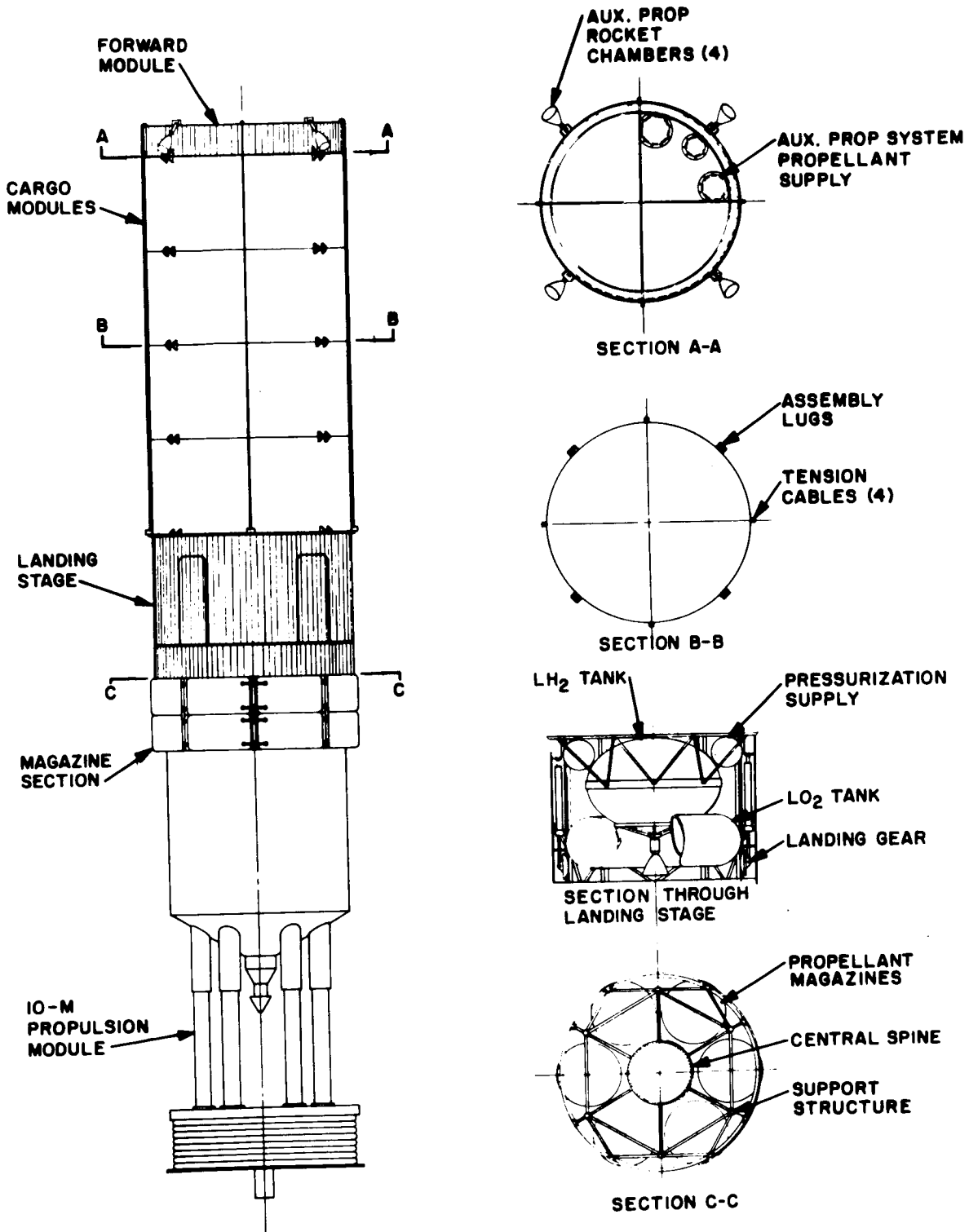
Fig. 3. 25--Orbit launched lunar logistic vehicle

The oxygen-hydrogen landing stage is somewhat similar to the landing system for the earth-orbit-to-near-lunar-surface ferry (Section 3.3.2). For this requirement a single main thrust chamber is provided at a central location since the nuclear propulsion module is jettisoned prior to landing. Also, the landing gear is located in the landing module rather than on the nuclear pulse module and is a much lighter system than for the orbit-to-surface ferry. The burnout mass of the landing stage and forward propulsion module used was as follows:

$$M_J = 700 + 0.12\ M_P + 0.5\ M_\lambda\ ,$$

where M_J = jettison (burnout) mass of stage, kg,

M_P = propellant mass, kg,

M_λ = payload mass, kg.

As in the landing stages described previously, the specific impulse was estimated to be 430 sec.

4. EARTH-LAUNCH VEHICLES AND LOFTERS

The earth-launch vehicle (ELV) used to accomplish the earth-to-orbit portion of a space mission is of great importance in the over-all mission system and to the accomplishment of the mission. Not only is the propulsion requirement for boost to orbit one of the most demanding in the entire mission, but the orbital delivery operation is usually the predominant cost in the total mission costs. Furthermore, if an advanced type of ELV should be required, its development and qualification would be very costly and time consuming. The 10-m propulsion module of this study was therefore tailored to be compatible with the Saturn V. The 20-m module was conceived for compatibility with projected post-Saturn ELVs. In this study the Saturn S-1C was considered as the "lofter" for the 20-m module in the Mode II operation (i.e., lofted to above the atmosphere and self-boosted to orbit followed by rendezvous to take on propellant and payload).

A lofter, as defined for this study, is a chemical-rocket-propelled booster whose major requirement is to loft the nuclear-pulse vehicle above the atmosphere and to a velocity of a few thousand feet per second prior to nuclear-pulse initiation. This terminology is used to differentiate this relatively low-requirement rocket system from typical space booster rockets having more stringent requirements. Lofting can, however, also be well done by the first stage of a ELV system if a suitably sized one is available. The Saturn S-1C stage, a recoverable or expendable first stage of a possible post-Saturn vehicle, and clustered solid-propellant motor configurations were considered for the lofting task.

4.1. SATURN V SYSTEM ACCOMMODATIONS

The 10-m propulsion module, its operational payloads, and its propellant magazines were all conceptually designed to fit the desired payload envelope for Saturn V. Neglecting the more detailed considerations of bending moments and dynamic control requirements, all three appear to be fully compatible with the two-stage Saturn V. Figure 4.1 illustrates the three classes of boost-to-orbit payloads in launch position on a Saturn V ELV for Mode III operation.

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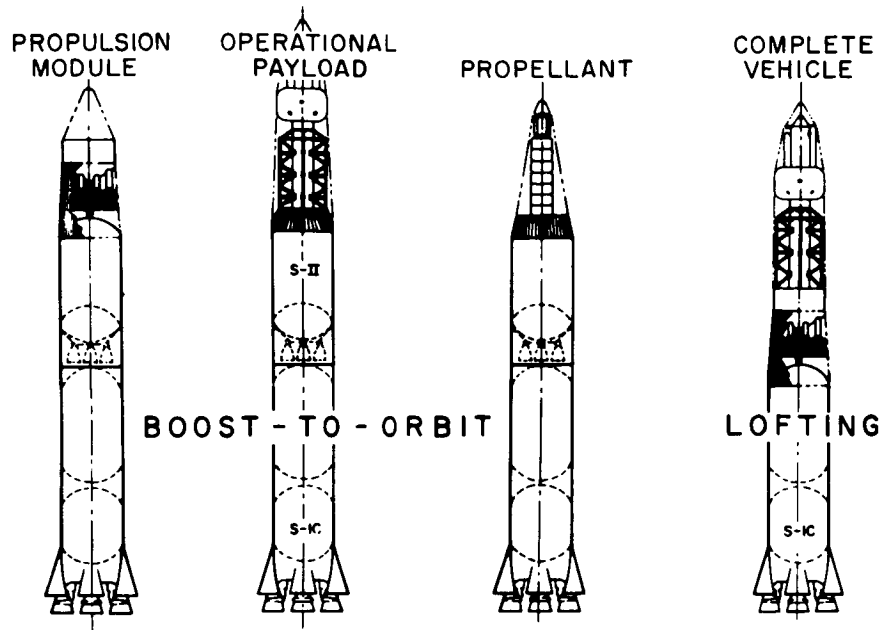
Fig. 4.1--Saturn V with three exploration mission payloads for orbital rendezvous and the S-1C stage as a complete-vehicle lofter

Also shown in Fig. 4.1, at the right, is a complete 10-m exploration vehicle atop the S-1C stage for an operational Mode II launch. After staging of the S-1C and a short coast period ($\sim$ 30 sec minimum) to prepare for start-up, nuclear-pulse operation is initiated to complete the boost to orbit. Prior to departing earth orbit on an exploration mission, the vehicle takes on additional propellant and payload, including personnel.

To minimize assembly operations during orbital rendezvous, it was decided that the propulsion module should be delivered to orbit in one fully assembled and checked-out package. The upper-vehicle portion of the operational payload was also conceived to be delivered in one fully assembled piece and to have simple and dependable structural ties for mating it to the propulsion module. The propellant magazines, as discussed in previous sections, were also designed and sized for convenience in delivery and loading onto the spine structure of the mission vehicle.

The propulsion module is a heavier item than either the operational payload structure or, of course, the individual propellant magazines. The total mass of the module, including the required transition structure, aerodynamic nosecone, fairings, etc., must therefore not

71

exceed the orbital delivery capability of the two-stage Saturn V. The mass of the transition structure, fairings, etc. , for the operational payload section or for a load of propellant magazines is of less importance, since additional miscellaneous payload may be included with either to make up a near-optimum payload mass. (It is important, however, for probability of successful delivery reasons, that the ELV payloads be confined to the minimum number of kinds, or three in this case as shown in Fig. 4.1. Thus, mixes of propellant magazines and miscellaneous payloads are made up such that they are interchangeable if more than one such delivery is required. See Section 4 of Vol. II.)

4.1.1. 10-m Interstage Adapter and Nosecone

The currently conceived interstage adapter and nosecone fairing for the 10-m propulsion module that is to be carried to orbit is shown in Fig. 4.2. Because of the flexible nature of the first-stage shock absorbers, the adapter structure must extend beyond the pusher and up to the cylindrical body portion of the propulsion module, as shown. The heavy
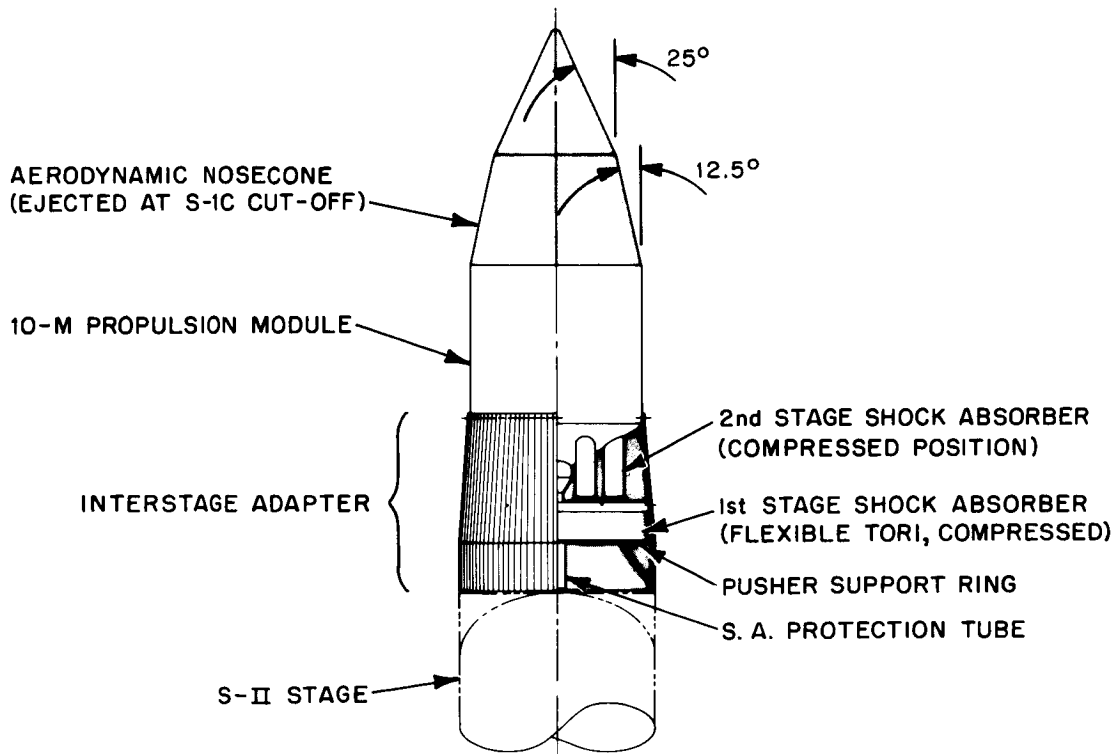
Fig. 4.2--Interstage adapter and nosecone concepts for 10-m propulsion module boost-to-orbit

pusher (~ 60 percent of total module mass) and the movable portions of the shock-absorber system, however, are clamped to and supported by the pusher support ring. The pusher support ring reacts most of its load inboard of the pusher edge, at about two-thirds the pusher radius, since the tapered pusher is not thick enough at its edge to withstand booster accelerations. The pusher support ring is located about 3 m (9 ft) above the S-II stage structure to provide clearance for the shock-absorber protection tube that extends through the pusher.

The module's upper cylindrical section and its contents (about one-third the total module mass) is supported by the upper adapter structure that extends above the pusher. This structure is currently shown as a continuous stiffened sheet design, which also functions as a fairing over the shock absorber. A multistrut design (perhaps six, with one adjacent to each cylindrical shock absorber) might be an acceptable alternative.

A preliminary weight allowance for the interstage adapter shown is as follows:

Upper structure, from pusher to the
   cylindrical module body . . . . 4, 500 lb
Lower structure, including pusher
   support ring . . . . 3, 500 lb
Contingency (for disconnects,
   fittings, etc.). . . . . 1, 500 lb
     Total weight allowance . . . . 9, 500 lb

The weight allowance is expressed in pounds rather than in metric units to be consistent with the nosecone and Saturn payload data provided by the Marshall Space Flight Center.

The nosecone shown in Fig. 4.2 is a low-drag, yet easy to manufacture, frustrum shape which evolved from a NASA study. [*]

Since the purpose of the nosecones in the present instance is to provide fairing for a blunt cylinder, with little or no nosecone volume requirement, a still simpler conical shape might also be satisfactory. The weight allowance for the nosecone was based on the curve of Fig. 4.3.

Since the nosecone is intended to be ejected at first-stage cut-off, an allowance of 1,500 lb was provided for ejection structure and

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[*] L. K. Linsley, "Nose Shape on Saturn Vehicles," George C. Marshall Space Flight Center, Memorandum No. M-AERO-A-20-63, March 6, 1963.
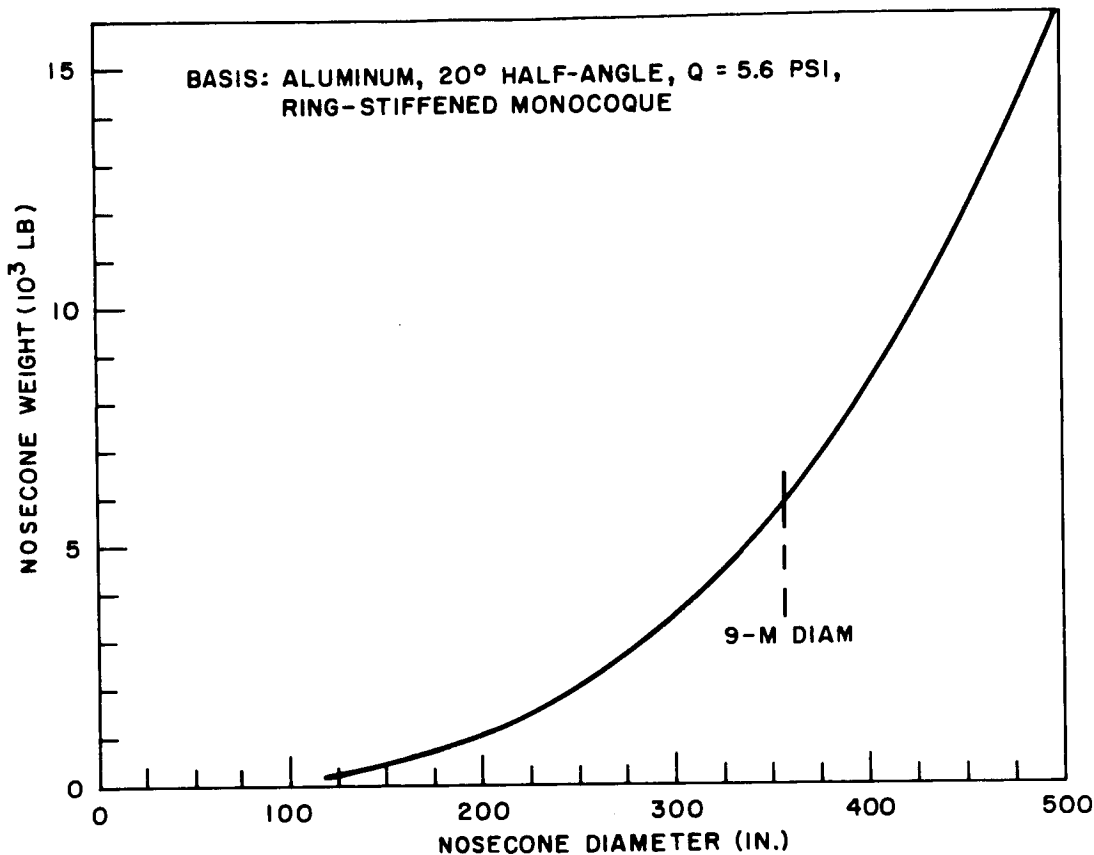
Fig. 4.3--Typical nosecone weight versus diameter data
(taken from Nevins and Helton)

mechanisms. Thus the total nosecone weight allowance for a 9-m
(355-in.) diam. cone with ejection provisions was 7,500 lb. This weight,
which is carried only through first-stage burnout, was considered the
equivalent of 2,000 lb carried to orbit, since the relative effect is of
the order of 4 to 1.

4.1.2. Saturn V Gross Payload Data

The two-stage Saturn V payload versus orbit altitude data[*] used
for this study are presented in Fig. 4.4. Referenced on the curve is the
212,000-lb payload point that represents the orbital weight of the 10-m
propulsion module plus its adapter structure and (effective)nosecone.

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[*]Nevins and Helton "An Investigation of Various Parameters Affecting
the Structural Weight of Rocket Vehicle Nosecones," Marshall Space
Flight Center Memorandum MPT-P&VE-5-63-4.

This total weight is comprised of the following:

Dry 10-m propulsion module	200,500 lb
Interstage adapter allowance	9,500 lb
Nosecone (effective orbital mass since the 7,500 lb is ejected at first-stage cutoff)	2,000 lb
Effective Saturn V payload	212,000 lb

From the data of Fig. 4.4, Saturn V will deliver this payload to a circular orbit at an altitude of 360 km. The orbital lifetimes (probably
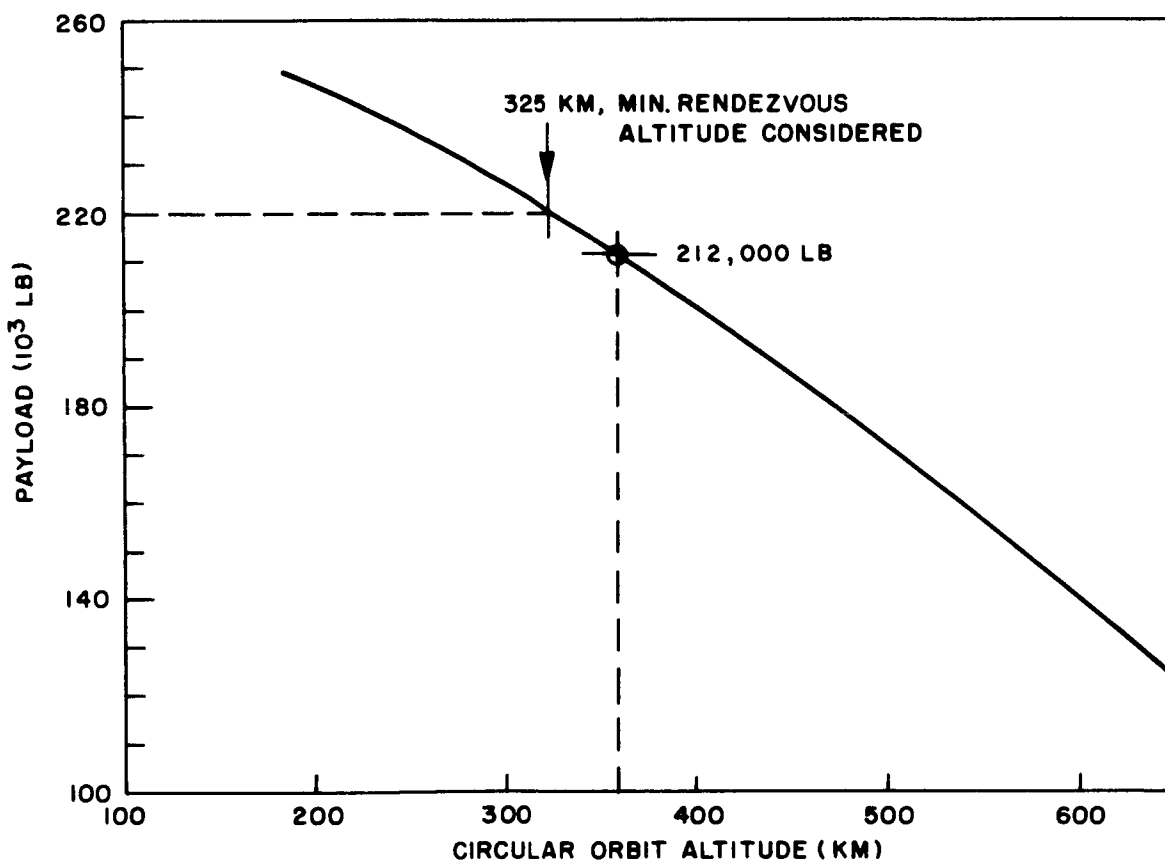
Fig. 4.4--Payload versus orbital altitude for a two-stage
Saturn V launch vehicle (11-20-63 data)

at least in months for the relatively high $W/C_D A$ of the nuclear-pulse vehicle components of this study) are such that reasonably involved orbital operations and a number of rendezvous are feasible. Higher rendezvous altitudes of 400 to 500 km are considered to be preferable by

advanced-operations planners if many rendezvous are involved, but these higher altitudes would significantly reduce the Saturn V gross payload capability to the range of 170,000 to 200,000 lb. For no operational mission considered in this study was a rendezvous altitude lower than 325 km assumed; at the minimum 325-km altitude, the Saturn V capability is 220,000 lb, or 100,000 kg.

4.1.3. 20-m System Compatibility

The 20-m vehicles of this study were also designed to permit the use of the Saturn V ELV for orbital delivery support. The 20-m propulsion module itself, however, is much too heavy to be carried to orbit by a Saturn V; therefore a Mode II operation (orbital build-up after the nuclear-pulse vehicle propels itself to orbit), rather than Mode III (orbital start-up), was assumed. Other than the propulsion module with a minimum upper vehicle for guidance and control, the remainder of the operational payload for exploration missions and the propellant magazines were sized to make up convenient Saturn V payloads. The 20-man and 50-man personnel accommodations were included in this consideration, and were designed as 10-m (32.8-ft) diam pressure vessels, two of them being used for the 50-man configuration. The Saturn S-1C stage was used as the lofter for the minimum 20-m vehicle for its self-boost-to-orbit launch. The resulting launch configuration would decidedly be a "hammer head," since the 20-m vehicle would be twice the S-1C diameter, and might require considerable adapter structure and a special or highly modified launch tower. The S-1C, however, has thrust and total impulse sufficient to adequately loft the 20-m minimum module to start-up conditions.

4.2. POST-SATURN ELVs

The post-Saturn ELVs considered in this study were primarily confined to systems using nuclear-pulse vehicles larger than the 10-m designs. In the early, parametric phase of the study, expendable post-Saturn vehicle first stages, recoverable post-Saturn vehicles, and clustered solid-motor vehicles were used as lofters for operational Modes I and II. In the later phase of the study, a hypothetical post-Saturn ELV was used (as was the Saturn V mentioned above) to support the 20-m vehicle operations.

A recoverable post-Saturn vehicle, such as the Nexus studied by General Dynamics/Astronautics,* appears to have very desirable

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* A Study of Post-Nova Launch Vehicles, Phase III, General Dynamics/Astronautics, Report AOK 63-035, December 1963.

operational and cost characteristics to support the larger nuclear-pulse
vehicle systems considered. For this study, however, the post-Saturn
ELV was a hypothetical one defined only by its orbital delivery capability
and its assumed operating costs. Its delivery capability was assumed to
be approximately four times that of Saturn V--delivery of a net payload
of 400,000 kg (880,000 lb) per flight. It is therefore nominally a
1,000,000-lb-to-orbit vehicle, the difference being an allowance for
the interstage adapter, nosecone fairing, etc.

For system costing purposes (as treated in Section 4 of Vol. II),
the direct operating cost per flight of the hypothetical post-Saturn vehicle
was 60×10^6. Its DOC per pound of gross payload is therefore $60/lb
(neglecting reliability), a cost figure that is reasonably consistent with
several recent industry studies of large advanced ELVs. The probability
of successful delivery assumed for this study was 0.95.

4.3. LOFTERS FOR OPERATIONAL MODES I AND II

The lofters of this study, as previously defined, are relatively
low-requirement first-stage boosters that loft the nuclear-pulse vehicle
above the atmosphere and to a few thousand feet-per-second velocity
prior to nuclear-pulse initiation. The nuclear-pulse vehicle then propels
itself into orbit or beyond. The seemingly most appropriate lofter depends
on the size of the nuclear-pulse vehicle and the availability of ELV
boosters. If a suitably sized ELV first stage is expected to be available,
such as the Saturn S-1C in the case of the 10-m vehicles, it would be
appropriate to use it even though it might be more refined and have a
higher capability than necessary for the task. If, on the other hand, no
suitable booster is expected to be available, a cluster of solid-propellant
rocket motors can be used for the lofting job. The development time
and the effort to design and qualify a cluster of existing solid motors
are considered to be much less than to develop a new size liquid-
propellant booster.

4.3.1. The Saturn S-1C Stage

The S-1C first stage of the Saturn V used as a lofter for the
10-m nuclear-pulse vehicle was shown in Fig. 4.1. Its performance as
a part of the earth-launched lunar logistic mission was discussed in
Section 3 of Vol. II. For this mode of operation the nuclear-pulse vehicle
is mounted atop the S-1C in much the same manner as the 10-m nuclear-
pulse propulsion module when it is being carried completely to orbit
except that there are some significant differences in the interstage
adapter requirements.

The interstage adapter for the lofted 10-m vehicle is similar in appearance to that shown in Fig. 4.2; the differences lie in its structural and release requirements. The dry 10-m propulsion module carried to orbit by the Saturn V weighs about 90,950 kg (200,500 lb), whereas the 10-m vehicle lofted by the S-1C stage typically weighs some four times as much as the dry module alone. For example, the initial stage weight of the lunar logistic vehicles lofted by S-1C is 357,000 kg (787,000 lb).

When the vehicle is lofted, the nuclear-pulse vehicle must be prepared to operate within a minute or two after first-stage cutoff. This requires that the propulsion module's shock absorbers be charged with gases at approximately the correct operating pressure prior to launch. The resulting shock-absorber forces tend to extend the pusher from the vehicle. In the compressed launch position, the total compressive force of the shock absorbers is approximately 320,000 kg (700,000 lb). Under static conditions, these compression forces relieve or slightly reverse the normally compressive loads in the interstage adapter between the pusher and module body; they also provide a "jack-in-the-box" energy source for stage separation after first-stage cutoff.

The stored energy in the compressed shock absorbers will provide a separation velocity of some 20 fps between the lofter and the nuclear-pulse vehicle, depending on the burnout mass of the lofter, the loading of the nuclear-pulse vehicle, and the shock-absorber pressures. Such a separation velocity will provide adequate separation by the time nuclear-pulse operation is initiated, typically 1 min or more. To use this jack-in-the-box separation energy, however, requires that the upper interstage adapter (1) be capable of a quick and clean disconnect and (2) be moved out of the way before the pusher begins to follow the vehicle at the end of the shock-absorber stroke (> 1 sec after release). The upper adapter could, for example, be built in sections that are hinged at the bottom and nearly simultaneous disconnect could be achieved with linear shaped charges or the like.

For the lunar logistic vehicle lofted by the S-1C stage, 15,000 kg (33,000 lb) was allowed for the interstage adapter, its associated release and removal mechanisms, and aerodynamic fairing about the nuclear-pulse-vehicle payload. About one-half of this weight allowance was for the interstage adapter itself, in comparison with the 9,500 lb estimated for the S-II stage adapter when only the dry propulsion module is boosted.

4.3.2. Clustered Solid-motor Lofters

The clustered solid-motor lofters were conceived as rather simple and structurally nonoptimized thrust producers. In most cases

they were considered to have a single, fixed, canted nozzle per motor (unless an off-the-shelf motor was otherwise suitably configured). No guidance function nor thrust vector control was assumed for the lofter itself since these functions exist in the nuclear-pulse vehicle for its own propulsion mode. In other words, the solid-motor lofters were conceived as comparable in function to aircraft JATO (jet-assist take-off) units—they are expendable thrust producers intended to provide a rapid initial acceleration to the basic vehicle.

Directional control of the vehicles during nuclear-pulse propulsion is achieved by liquid-propellant rocket motors that provide lateral thrust near the nose of the vehicle. Preliminary study has indicated that the directional-control requirement for such motors is similar to that required for the over-all configuration while being lofted.

Hence, the directional-control system of the basic vehicle would provide directional control during the brief lofting period as well as during nuclear-pulse operation.

A preliminary investigation of possible lofters and their performance was performed during the early, parametric phase of this study. At that time a wide range of vehicle sizes was being considered for gross lofted weights from 400,000 lb to 12,000,000 lb. Lofters considered for these vehicles varied in their requirements from three 120-in.-diam, 2- or 3-segment motors to ten 260-in.-diam motors. The weight allowance for the clustering structure varied from 1.8 percent of the initial weight of the motors being clustered for the larger lofters to 2.2 percent for the smaller lofters. In addition, a conservative allowance of 3 percent of the lofted vehicle weight was made for the interstage adapter and minor fairings.

The wide range of vehicle and lofter sizes was investigated primarily to establish the over-all trends of system performance and cost effectiveness with variations in vehicle mass. These results were reported in Vol. II. The cost criteria and lofter cost data are presented in Section 8 of this volume.

5. UNIQUE OPERATIONAL CONSIDERATIONS

To a large extent the operational considerations discussed in this section are unique to nuclear-pulse propulsion. Considerations such as nuclear environment, internal noise, ground and flight hazards, maintenance concepts, and the availability of fissionable material are included in this assessment.

5.1. NUCLEAR ENVIRONMENT

5.1.1. Radiation Levels

The radiation levels within the nuclear-pulse propulsion module or the upper vehicle are primarily functions of the separation distance between the explosion source and the vehicle and the attenuation by the heavy steel pusher. The combination of these factors provides several orders of magnitude of radiation attenuation.

Radiation levels in the vicinity of the vehicle body resulting from operation of the nuclear-pulse system decrease noticeably with increasing thrust or propulsion module size principally because of two factors:

1. Although both the pulse-unit yield and the propellant* thickness increase with vehicle size, the radiation attenuation provided by the propellant increases exponentially.

2. As the size of the vehicle increases, the standoff distance from the explosion source to the pusher face and to other vehicle components increases, providing a R^2 advantage. Low neutron-flux levels are anticipated as well as correspondingly low activation levels of the vehicle. Additional analysis is required in this area in order to take full advantage of all the possibilities which would influence low activation levels. However, preliminary studies indicate that almost unlimited personnel access would be permitted throughout the vehicle and engine shortly after propulsion-system shutdown.

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* "Propellant" is used here in the more restricted sense of the nuclear-pulse-unit design and refers to the layer of dense inert material in the pulse unit which is vaporized and directed toward the vehicle pusher. See Section 2 for a description of the pulse unit.

5. 1. 2. Fission Products and Trappage

Some of the products of a nuclear reaction involving fission are radioactive and may be the source of undesirable atmospheric contamination or the source of hazardous radiation to the vehicle and its contents if these products pass above or adhere to the vehicle.

To assess these possibilities, it is necessary to know the amount of fission products produced during, and their disposition after, each pulse-unit detonation.

For the 10-m propulsion module selected for this study, a yield of approximately 1 KT per pulse is generated. This results in total yields for the earth-departure maneuver in the Mars mission of 0. 5 to 1. 2 MT, depending on the payload and mission-velocity requirements.

A characteristic of the current pulse-unit design is that most of the fission products and the unburned fuel tend to leave the point of explosion within a 2π solid angle and hemispherically away from the vehicle. Assuming that all of the fission products are ejected in this manner provides an upper limit to the trapped fission yield per explosion for any particular vehicle.

The average velocity of the fission products produced by the pulse unit is approximately 2×10^3 km/sec. Since this velocity is well in excess of earth escape velocity, it is believed that the trappage by the earth's magnetic fields of the fission products that do not intercept the earth's atmosphere can be reduced to the point of being almost negligible. This is particularly true if the operating altitude and latitude are carefully selected. Preliminary calculations indicate that fission products from the pulse unit traveling in a more or less vertical direction (upward) and released at an altitude of more than 150 km would not be trapped except possibly at low geomagnetic latitudes. Fission products which, after leaving the explosion source, intercept the earth's sensible atmosphere will be trapped, however, and current estimates indicate that they would probably return to earth, creating fallout in times less than the strontium 90 half-life (27. 7 years).

A number of possible countermeasures are proposed to reduce substantially the fission-product trappage in the atmosphere. One of the most obvious is to utilize orbital start-up at a few hundred kilometers altitude and thereby reduce fission-product trappage by factors of 2 to 4. Improvements in the design of advanced pulse units might be achievable

whereby fission products could be reduced by factors of 10^2 to 10^3. Such improvements would be obtained by reducing the fraction of total yield due to fission by two to three orders of magnitude, the remaining yield being contributed by fusion. A further advantage from this approach is the improvement in fuel economics.

It is also possible to consider the focusing of fission products upward from the point of explosion, at some sacrifice in I_{sp}. This could reduce those fission products likely to be trapped in the atmosphere to approximately 1 percent or less of that estimated in the 2π expansion condition.

It is believed that some combination of these suggested techniques could conceivably result in a reduction of the trappage to only 10^{-6} of current estimates. Considering the longer-term possibilities, if and when pure fusion devices become a reality, fission products, by definition, will not be present.

5.1.3. Electron Injection

An important aspect of nuclear-pulse propulsion in near-earth operation is the possibility of electron injection creating artificial radiation belts. The STARFISH high-altitude nuclear test produced artificial radiation belts of high-energy electrons, the source of these being the β^- decay of fission products. The STARFISH tests were detonated reasonably close to the magnetic equator and injected β particles on magnetic lines which lie well within $L = 2.0^*$ earth radii (geocentric). Since the launch of a large nuclear-pulse vehicle may involve a megaton or more of fission-energy release, it is important to assess the corresponding production of artificial radiation belts. Considerable information is being produced to further understand the implication of the electron-trappage problem. Recent data obtained with artificial satellites indicate that the lifetime of trapped electrons along magnetic lines which are $L \geq 2.0$ at the equator is on the order of a few weeks. The mechanism which restricts lifetime is unknown, but it is most likely caused by plasma instability. Within $L = 2.0$ the lifetime of trapped electrons is determined by ionic collisions and is on the order of years. Thus, nuclear-pulse-vehicle operation resulting in electron ejection on magnetic field lines of $L > 2.0$ should not create a long-lived artificial radiation belt. This may be achieved by restricting the low-altitude operation of the nuclear pulse propulsion system to regions of magnetic lattitude $40°$ north or south. ($L = 2.0$ intersects the magnetic equator and at an altitude of approximately 6,700 km.) Figure 5.2 shows the general conditions of the lifetime of trapped electrons associated with various injection latitudes.

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*L is the distance from center of earth to intersection of earth magnetic field line with earth magnetic equator.

The calculated nuclear radiation levels at various points during operation of a 10-m propulsion module are given in Fig. 5.1. No allowance was made for atmospheric and structural scattering (the start-up normally being at 90 km or higher) and no allowance was made for additional attenuation provided by the stored propellant, structure, payload, or other equipment except the pusher. Further reduction in radiation levels would undoubtedly result from normal development improvements in the design of the pulse unit and from designing the nuclear device of the pulse unit specifically for propulsion purposes rather than utilizing state-of-the-art nuclear devices.

350 REM/SHOT

2×10^3 REM/SHOT

10^5 REM/SHOT

L

PULSE UNIT

L 2×10^6 REM/SHOT
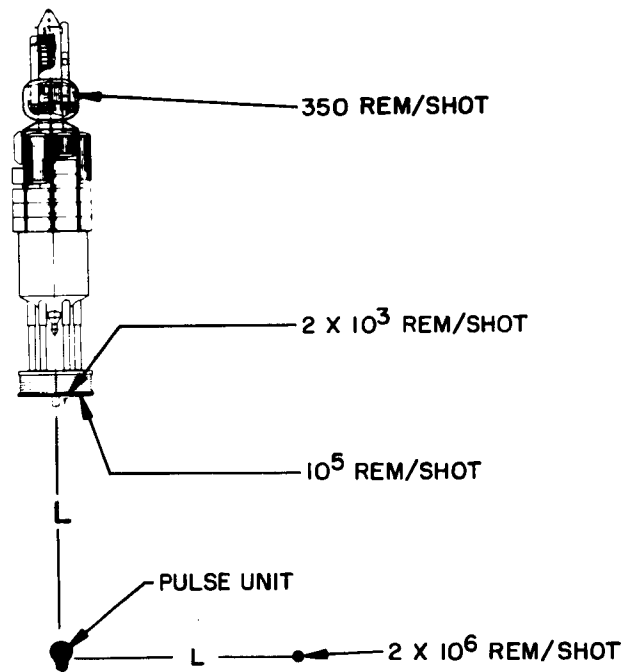
Fig. 5.1--Operational nuclear environment about the
10-m propulsion module

It should be noted that the radiation levels in Fig. 5.1 are substantially lower, and believed to be more accurate, than the scaled parametric values which were derived earlier in the study. The earlier data indicated that radiation levels at the top of the pusher varied from approximately 10^4 rem/shot at a thrust of 3.5×10^6 newtons (10-m module) to 10^3 rem/shot at a thrust of 100×10^6 newtons. The newer data in Fig. 5.1 reflect that radiation at the top of the pusher of the 10-m module is a factor of five lower than that estimated earlier, resulting in an over-all personnel shielding weight reduction of some 7 to 10 percent. The earlier and higher levels were used as the basis for all powered-flight-station shielding analyses in this study, providing some degree of conservatism.
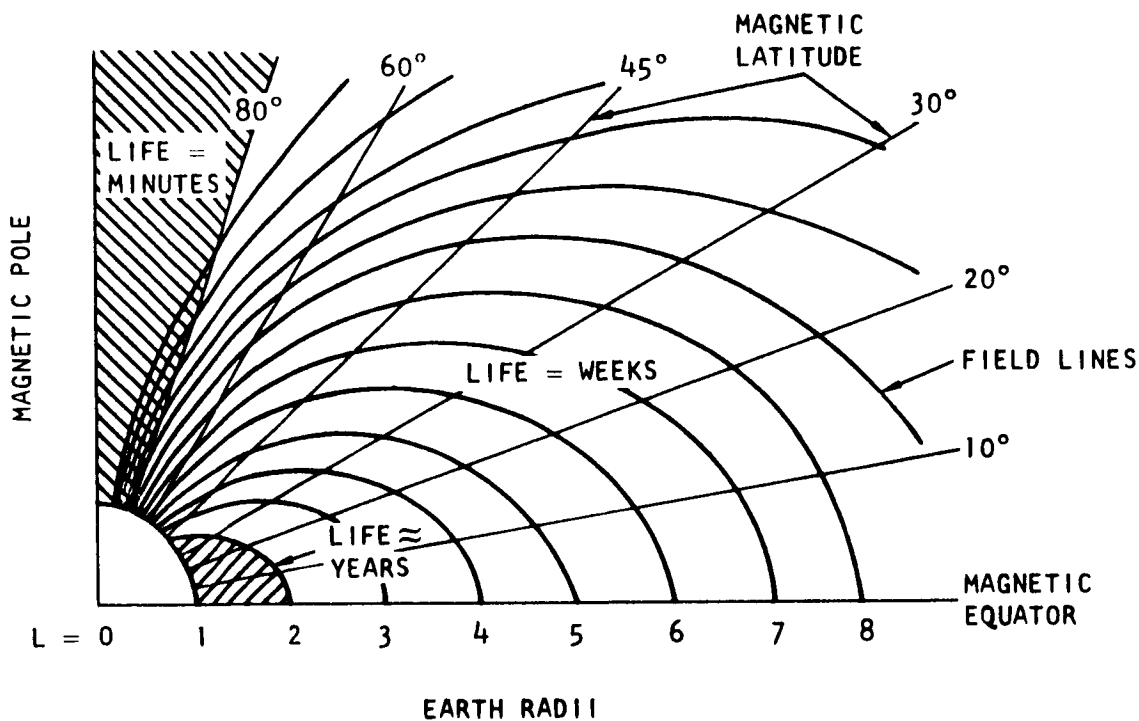
Fig. 5.2--Trapped electron lifetime summary

5.1.4. Eyeburn possibility in Vicinity of the Earth

An attendant effect of nuclear-pulse explosion in the vicinity of the earth, even at high altitude, is that it would be visible from a very large area of the earth's surface. There is a reasonable probability that individuals would be looking at the detonation points at the time of the explosion. It is therefore necessary to assess retinal damage for nuclear-pulse-vehicle operations and to determine altitude thresholds.

Eyeburn is caused by the reradiation of air heated by the bulk of the explosive energy. When the detonation is between 60- to 80-km altitude, the bulk of the energy is reradiated in a very short time compared to the human eye blink time of 0.15 sec.

The thermal flux which propagates from the explosion to a distance R (kilometers) is

$$\phi = \frac{6.4\eta Y}{R^2} \text{ cal/cm}^2,$$

where Y is the total yield in kilotons and η is the fraction of the yield reradiated below 3,000 Å in the first 0.15 sec. To assess retinal damage,

it is necessary to account for focusing of the eye lens. This is effected by multiplying the flux given above by the factor $(P/f)^2 (R/D)^2$, where P is the pupillary opening ($\leq$ 0.8 cm), f is the eye focal length (1.7 cm), and D is the effective fireball radius in kilometers. The result yields the energy incident on the retina:

$$\phi' = \frac{1.1\eta Y}{D^2} \text{ cal/cm}^2.$$

Figure 5.3 shows an assessment of eyeburn protection require-ments as a function of altitude. It is estimated that a ϕ' greater than 0.6 cal/cm^2 can cause retinal burn. By evaluating the radiative flux for several different yields by use of the PEEP code,[*] it was determined con-servatively that at altitudes above about 90 km, the flux would not be sufficient to cause retinal burn to the unprotected eye. This is certainly conservative, and other estimates[†] of the same problem, assuming different damage criteria and fireball image size on the retina, indicate that critical altitudes might decrease to approximately 30 km before protection is required. For the operational modes considered in this study all nuclear-pulse operations are started above 50 km altitude and most are started above 90 km.

5.2. INTERNAL NOISE ANALYSIS

An accurate estimate of environmental noise and vibrations is rather intimately dependent on the details of structural elements, joining methods, and local surface treatment. For this reason the noise analysis described herein must be considered limited to the indication of the exist-ence (or absence) of a problem area, the order of magnitude of the problem, and the design approaches which might be employed to alleviate the problem.

It was concluded from the analysis performed that most of the acoustic energy in the shielded crew compartment (the powered flight station or command module of the vehicle) would be at low frequencies. Since these noise levels are directly related to the structural vibration levels, reduction of structural vibration would provide a commensurate reduction of noise. Such reductions can be accomplished by increasing the stiffness of the structure and by the addition of damping.

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[*]Theoretical Study of the Effects of Very Large Yields Above the Atmosphere, Air Force Special Weapons Center, Report AFSWC-TR-61-53, August, 1961.

[†]ORION Retinal Burn Study, Glenn L. Martin Co., Report MND-2895 Add., 11/20/62.
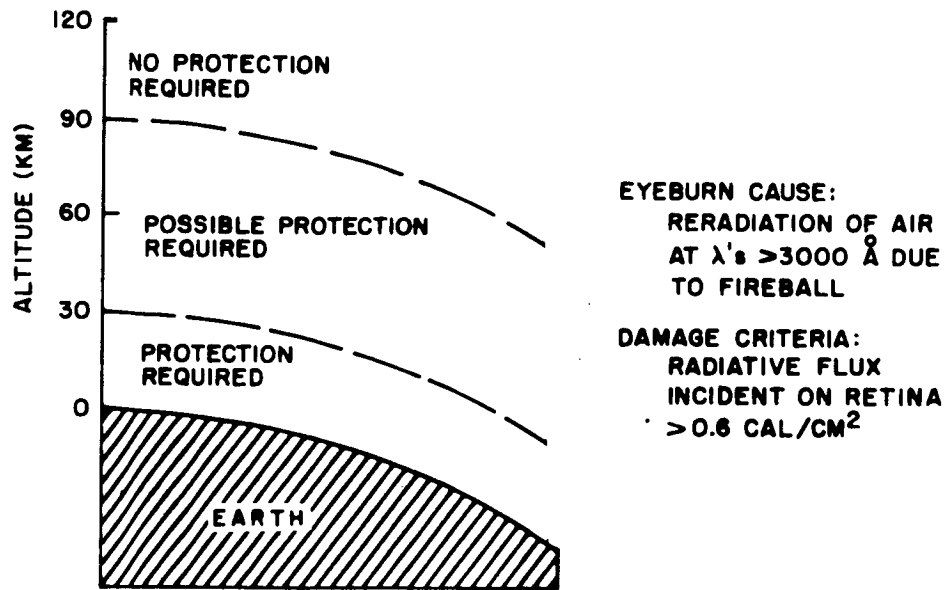
Fig. 5.3--Preliminary assessment of eyeburn conditions

After the noise analysis discussed below was completed, the conceptual exploration configurations designed in the later phase of the study were reviewed for their acoustic qualities. The proposed configurations for the crew compartments, utilizing significantly thick layers of polyethylene or other hydrogenous shielding material, were considered to provide excellent damping properties. Furthermore, the possible use of a foamed polyurethane core material for the payload spine walls would provide a means of high vibrational energy absorption. It was therefore concluded that the configurations proposed in the specific conceptual designs provide sufficient stiffness and damping for the control of low-frequency noise and vibration, as required. At the higher frequencies, the provision of conventional Fiberglas sound-proofing treatment should adequately control any noise.

For the purpose of this analysis, the section of the crew compartment of the nuclear-pulse vehicle which is occupied during nuclear-pulse propulsion is considered to be a 14-ft-diam cylinder, 6.5 ft high, with a double 0.25-in. plate sandwich of built-up steel floor having the equivalent stiffness of a solid isotropic steel floor 3.5 in. thick.[*] For a first approximation, all surfaces of this cylinder were considered to be rigid except the floor. The interior noise was assumed to be generated by vibration of the floor.

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[*]This analysis was conducted early in the study period prior to serious consideration of the smaller (10-m) vehicles. The reference crew compartment is a single-level, 20-man design for a vehicle 20-m to 33-m in diameter.

During normal operation of the nuclear-pulse engine, the crew compartment responds impulsively at approximately 1-sec intervals. The impulses assumed for analytical purposes are shown in Fig. 5.4. It should be noted that although the two cases shown are not actually representative, they were selected to establish boundaries for the noise analysis.

The mathematical procedures which are presently used to delineate the propulsive response impulses of the vehicle do not include the high-frequency perturbations necessary for an evaluation of the acoustic noise problem. Therefore, it was agreed that for this first approximation the frequency content of the vehicle response function might be represented by its expansion as a Fourier series. Although it was recognized that this procedure provided "ersatz" frequency information, it was considered acceptable within the limited objectives of this analysis.

Figure 5.5 shows the envelope curves drawn through the harmonic amplitudes of the functions shown in Fig. 5.4. The important points to be noted are: (1) Below about 6 cps, the relative harmonic amplitudes between Case 1 and Case 2 are within one order of magnitude. (2) At the higher frequencies, Case 2 effectively has little harmonic content compared with Case 1. Since Case 1 provided more conservative numbers, it was used exclusively for the remainder of the analysis. As a matter of interest, Case 1 expanded essentially as odd harmonics of a sine series.

The curve for Case 1 shown in Fig. 5.5 may be considered as the nonresonant spectrum acceleration of the crew-compartment floor. In order to account for the amplification at resonant frequencies of the floor, the following procedure was used. The modal density of the floor was first computed, i.e., the number of resonant modes per unit frequency bandwidth. For a flat panel, the modal density is

$$\frac{\Delta N}{\Delta f} = \frac{\sqrt{3}}{C_L} \left(\frac{A}{H}\right),$$

where C_L is the longitudinal wave speed in steel, A is the floor area, and H is the equivalent thickness of the floor. For the floor dimensions provided, the modal density computes to be about 0.10 modes/cps.

Reference is now made to the eight ASA standard octave bands used in acoustic work, starting with the first octave at 37.5 to 75 cps through the eighth octave at 4,800 to 9,600 cps. To estimate the number of resonant (floor) modes in each octave band, the modal density was multiplied by the geometric mean frequency (GMF) for each band. For example, in the first octave where the GMF is 53 cps, N = 5.3 modes, and in the eighth octave where the GMF is 6,900 cps, N = 690 modes.
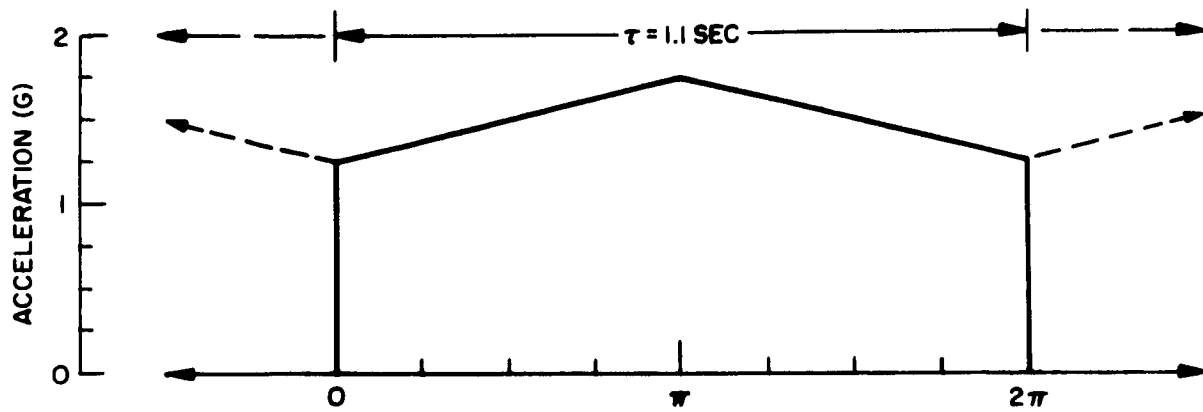
$$f(x) = 0 \qquad \text{FOR } x = 0 \text{ AND } 2\pi$$
$$= (1.25 + mx) \qquad \text{FOR } 0 < x < \pi$$
$$= 1.75 \qquad \text{FOR } x = \pi$$
$$= 1.25 + m(2\pi - x) \quad \text{FOR } \pi < x < 2\pi$$

CASE 1
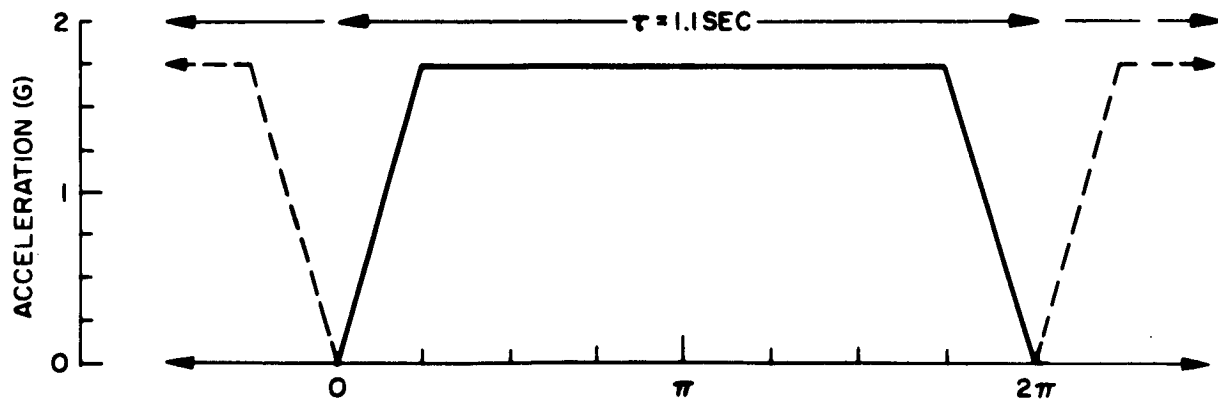
$$f(x) = 0 \qquad \text{FOR } x = 0 \text{ AND } 2\pi$$
$$= 7x/\pi \qquad \text{FOR } 0 < x \pi/4$$
$$= 1.75 \qquad \text{FOR } \pi/4 < x < 7\pi/4$$
$$= 14 - 7x/\pi \quad \text{FOR } 7\pi/4 < x < 2\pi$$

CASE 2

Fig. 5.4--Assumed impulses for crew-compartment noise analysis

88
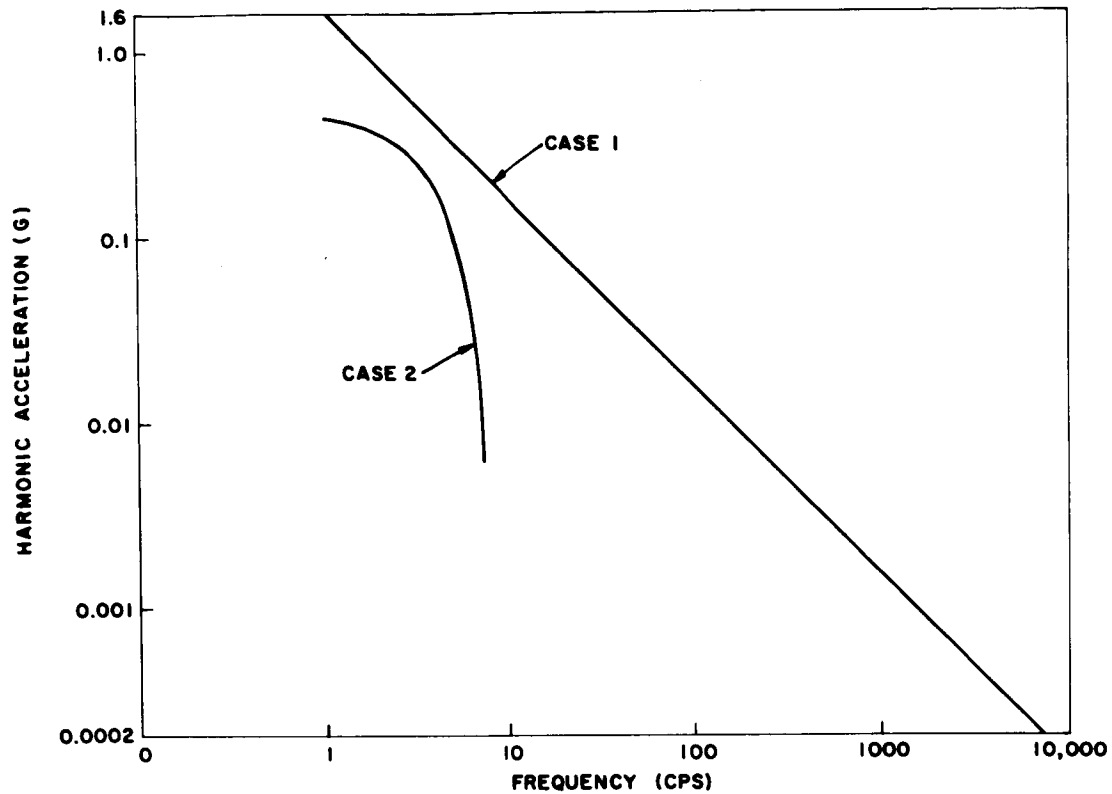
Fig. 5.5--Spectrum curves (smoothed) of harmonic amplitudes

The next assumption was that the structural damping of the floor (from all causes), ζ, is 2 percent of critical. On this basis, the response magnification at each resonance will be 25 times. In order to provide the root-mean-square (rms) acceleration of the floor for each octave band, the nonresonant spectrum acceleration at the GMF for each octave band (Fig. 5.5) was multiplied by 25, squared, and then multiplied by the number of resonant modes in the octave. This provided the mean-square acceleration and the square root of this term provided the rms acceleration. For example, in the first octave,

$$\text{Octave rms } G = \sqrt{(0.03 \times 25)^2 \, (5.3)} = 1.73 \text{ (see Fig. 5.6)}.$$

From the rms acceleration, the mechanical power in each octave was computed by determining the velocity at each GMF and multiplying by the weight of the floor. These values in foot-pound-second units were converted to watts for manipulative purposes. For the first octave, for example, the mechanical power was computed to be 5,370 watts.
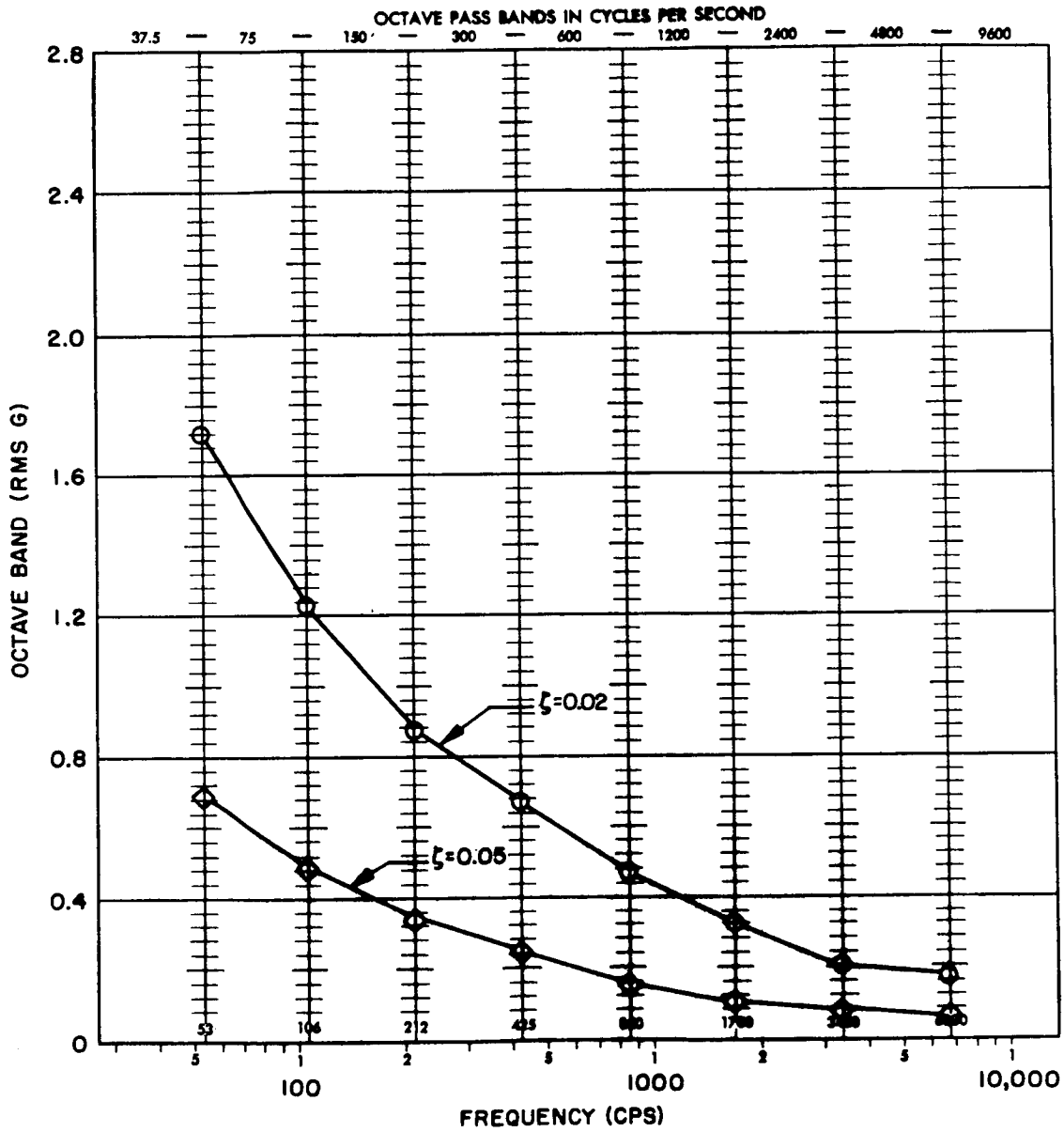
Fig. 5.6--Crew-compartment floor vibration (Case 1)

In estimating the acoustic output provided by the vibrating floor, it was considered that for this very preliminary exercise the most straight-forward procedure would be to assume an accoustical-mechanical conversion efficiency, $\eta = 0.05$, for radiation into half-space. This provided 269 acoustic watts in the first octave. The expression for the acoustic power level[*] is

$$PWL = 10 \log_{10} Pwr \text{ (watts)} + 130 \text{ db re } 10^{-13} \text{ watt.}[\dagger]$$

In the first octave, $PWL = 1.54$ db re 10^{-13} watt.

Under standard atmospheric temperature and pressure conditions, the sound pressure level (SPL) in the crew compartment is

$$SPL = PWL + \log_{10}\left(\frac{1}{4\pi r^2} + \frac{4}{R}\right) + 0.5 \text{ db re } 0.0002 \mu bar,[**]$$

where r is the distance from the radiator (in this case, $r = 6$ ft above the floor) and R is the room constant, $R = S\,\overline{\alpha}/(1 - \overline{\alpha})$, in which S is the area of bounding surface of the compartment and $\overline{\alpha}$ is the average sound absorption coefficient for the chamber. With

$$S = 594 \text{ ft}^2,$$

$$\overline{\alpha} = 0.05 \text{ (assumed for hard walls)},$$

then $R = 31.3$ and $SPL = PWL - 9$ db.

In the first octave, $SPL = 145$ db re $0.0002 \mu bar$. If the cabin pressure is assumed to be 10 psi, then $SPL = 143$ db re $0.0002 \mu bar$.

The octave sound pressure levels for a 10-psi cabin atmosphere are plotted in Fig. 5.7. Levels are shown for both 2 percent and 5 percent critical structural damping and for both 1 percent and 5 percent acoustical-mechanical efficiency. For comparison, maximum permissible noise levels for military airborne vehicles are also shown.

From the foregoing analysis of the noise problem, the following conclusions have been reached:

1. Most of the acoustic energy in the cabin is at low frequencies.

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[*] L. L Beranek, Acoustics, McGraw-Hill.

[†] This formula is not precisely applicable to the situation being studied, but it was used as a first approximation.

[**] This formula is also a first approximation only.
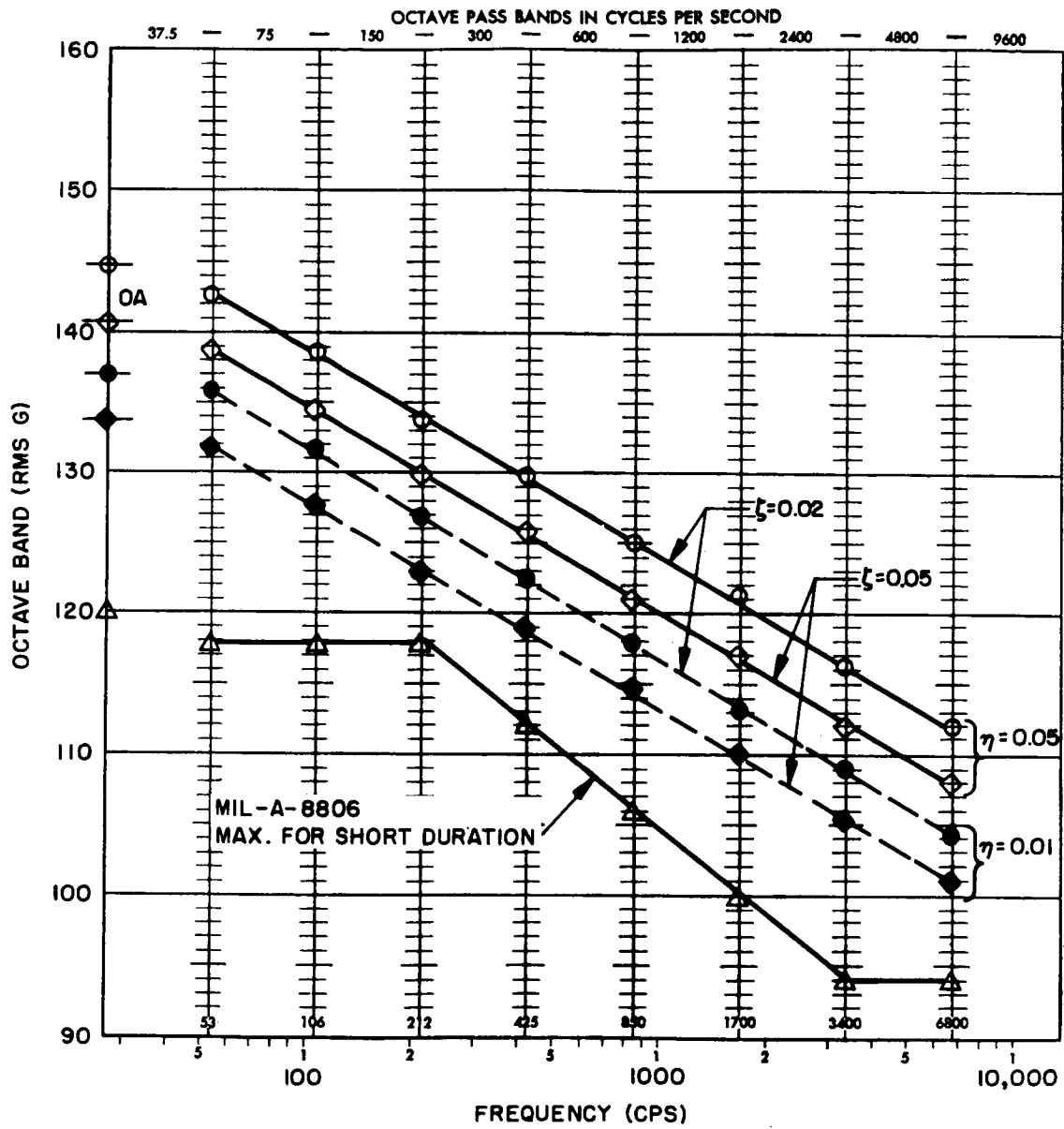
Fig. 5.7--Crew-compartment noise (case 1) at 10-psi pressurization

2. Low-frequency noise is not absorbed by sound-proofing treatments, such as the installation of Fiberglas on the interior walls, and must be controlled by reducing the vibration level of the structure (see Fig. 5.6).

3. The vibration level of the structure can be reduced by the addition of damping (e. g., by laminating low modulus materials or by joining structural elements by means of low modulus fittings) and by increasing the stiffness. The vibration and noise levels described herein are related to assumed harmonic responses of the structure and thus the results are quite conservative. A randomly vibrating structure would provide levels roughly only about one-fourth of those stated. The required additional damping and stiffness therefore appear attainable within the state of the art.

4. At the higher frequencies (above 500 cps), the provision of conventional Fiberglas sound-proofing treatment should provide adequate noise reduction.

5. Although cabin vibration in the infrasonic frequency range, below about 20 cps (see Fig. 5.5) was not specially treated in this analysis, some study should be made of possible deleterious psychological effects on crew members. (Since the duration of typical maneuvers is in the range of 5 to 15 min, however, there may not be any problems from such vibrations.)

5.3. PRELIMINARY GROUND-HAZARDS ASSESSMENT

Nuclear-pulse vehicles utilize a large number of plutonium and/or uranium-bearing nuclear explosive devices. These explosive devices are similar to those used in conventional stockpiled weapons in that they have multiple fail-safe mechanisms plus inherent safety characteristics so as to preclude their accidental (nuclear) detonation. Each device also contains a quantity of high explosive which may or may not detonate under certain accidental conditions. Accidental detonation of the high explosive contained in a device will rupture the case, fracture, disperse, and perhaps vaporize the nuclear material but will not create a nuclear event.

Nevertheless, because of the biological implications of plutonium dispersal and also the very serious political consequences of any accident involving special materials, a preliminary assessment has been made of maximum credible accidents which might occur during operational use of nuclear-pulse vehicles and of the possible resulting hazards to the ground environments.

Three classes of failures that might result in ground hazards are indicated in Fig. 5. 8. The Class I failure represents a catastrophic pad abort of the booster or lofter stage with ensuing explosion and fire; Class II is a propulsion failure short of orbital velocity wherein a fueled vehicle (or additional propellant units for orbital refueling) may fall back toward the earth; and Class III is an operational misfire of a pulse unit with the possibility of its subsequently returning to the earth.

CLASS I—PAD ABORT   CLASS III—MISFIRE

CLASS II—FAILURE TO ORBIT
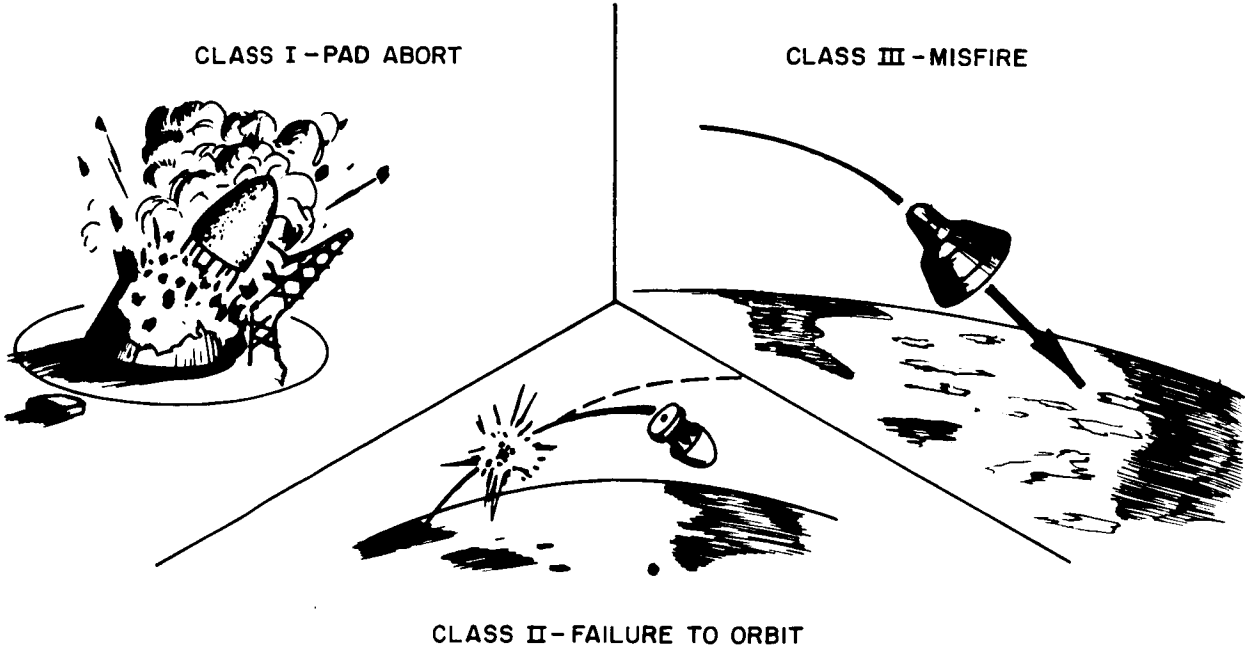
Fig. 5. 8--Operational failure mode classification for ground hazards

5. 3. 1. Class I—Launch-pad Abort Failure

The Class I failure considered would be a booster failure that resulted in an explosion and fire with the possibility that approximately 1, 000 pulse units would fall in the fire and all of the HE (20, 000 kg) would detonate. Assuming current nuclear-device design practice, there would be no nuclear explosion or criticality event resulting from such an accident. However, if all the high explosive detonated, overpressures of up to 1 psi might be expected at distances of approximately 300 m (1000 ft) and a possible shrapnel hazard might exist to 2, 000 m (6, 500 ft) from the explosion point. For most chemical-propellant booster stages considered, the potential blast and shrapnel hazard from the chemical stages themselves would be in excess of the high-explosive-produced effects, and hence it is

concluded that the additional explosion hazard incurred by employing
nuclear-pulse vehicles would be negligible.

A more serious problem would be the possible burning or vapori-
zation of plutonium, which could produce a downwind inhalation hazard and
ground-contamination hazard. Tolerances on maximum permissible con-
centrations have been established by the Atomic Energy Commission. *

If all of the plutonium available were vaporized in the form of an
aerosol, a substantial down-range hazard might exist, requiring a combi-
nation of remote basing and a considerable over-water, downwind exclusion
area. Some recent field experiments by the Defense Atomic Support Agency
(Operation Roller Coaster) might have bearing on this problem, both to
assess the fraction of plutonium available as an aerosol and the resulting
downwind concentration levels. Results of this assessment should become
available relatively soon.

A very interesting possible countermeasure to a Class I accident
would be to mount the launch gantry and booster support over open water
of a few tens of meters deep. Any resulting failure of the launch vehicle
or the nuclear-pulse stage would result in collapse and submersion, which
would quench any fire without compromising individual pulse-unit mechanical
integrity. The salvage of the dense and sturdy nuclear-pulse units might
then be possible for their refurbishment.

5.3.2. Class II—Failure to Orbit

If the nuclear-pulse vehicle or an ELV carrying a load of pulse
units should fail short of orbital velocity, the loaded vehicle or the load of
pulse units could reenter and impact the earth. Again, all of the HE might
detonate and produce overpressures up to 1 psi at 300 m (1000 ft) and with
shrapnel to 2,000 m (6,500 ft). This event differs from Class I in two
important ways: the impact might well be out of a controlled area and high-
temperature chemical fires would not necessarily occur. Even if the impact
should occur on land, nuclear explosions are not credible, but resulting
plutonium contamination, possibly in an uncontrolled area at ballistic
ranges, would make this class of accident potentially more serious than
Class I.

In the most interesting operational cases considered so far in this
study, orbital rendezvous of two or more Saturn or post-Saturn ELVs is
considered prior to the nuclear-pulse vehicle's embarking on a lunar or

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*For the uncontrolled population under conditions where dust
presents a problem, a ground concentration of 10 to 25 $\mu g/m^2$ is permis-
sible. The maximum permissible plutonium concentration in air should be
less than 10^{-12} $\mu c/ml$ (1.6×10^{-17} g/ml).

planetary mission. And whether the nuclear-pulse vehicle boosts itself to orbit or is boosted to orbit, it will be unmanned until after reaching orbit.

Therefore, a possible countermeasure for a Class II accident, wherein there is a hazard that nuclear-pulse vehicle and/or plutonium-bearing pulse units might return to the earth's surface, would be to intentionally single-point detonate each pulse unit or perhaps one or more pulse units of such yield as to guarantee vaporization of the fuel at the maximum altitude possible. Dispersal of nuclear materials by this technique would then accrue over a broad-enough area to attain dilution well below established tolerance levels.

5.3.3. Class III—Pulse-unit Misfire

The type of accident considered in the event a pulse unit should fail to detonate after ejection, and both the automatic and command destruct systems should also fail, would be one in which the pulse unit reenters the earth's atmosphere undamaged and falls to earth. The HE in the pulse unit might detonate on impact, producing overpressures up to 1 psi at 35 m (115 ft) and spreading shrapnel to 300 m (1000 ft). No nuclear explosion would occur but local plutonium contamination would extend to 300 m (1000 ft). Actually, this might be considered a special case of a Class II accident with two exceptions: (1) only one pulse unit would be involved, and (2) for most of the missions considered, orbital departure is contemplated which would leave the unfired pulse unit in an orbit of some duration.

The assumption that both the automatic and command destruct systems fail is extremely pessimistic in that multiple and redundant systems operating on a variety of environments (time after arming, barometric pressure, reentry temperature, deceleration, etc.) could be employed to guarantee single-point high-explosive detonation and dispersal of pulse-unit debris. It is especially important to guarantee that no pulse unit reenters the earth's atmosphere intact. Although no nuclear event would occur upon impact, the political implications, at least for the foreseeable future, of such an event could be sizable. With proper redundancy design of the destruct system, the probability of such an event's occurring can be made arbitrarily close to zero.

5.4. PRELIMINARY FLIGHT-HAZARDS ANALYSIS

From a preliminary investigation of the flight hazards to mission personnel, the following five types of hazards were considered in the conceptual design of both the propulsion module and the personnel accommodations:

Nuclear radiation,
On-board fire or explosion,

Boost-to-orbit abort,
Meteoroid damage, and
Propulsion failure.

The design philosophy and countermeasures currently proposed for these flight hazards are discussed below.

5.4.1. Nuclear Radiation Hazards

The three major sources of nuclear radiation expected to be encountered during a typical mission are the nuclear-pulse explosions, solar flares, and the earth's radiation belts. The greatest flux density, although very short lived, is due to the nuclear-pulse explosions. As previously discussed in Section 3.1, the radiation from this source stipulates the type and amount of personnel shielding, which in turn is then adequate to attenuate any probable solar-flare intensity and is more than adequate to protect the crew while traversing the radiation belts.

The radiation intensity from the nuclear-pulse units (approximately 350 rem per pulse in the personnel accommodations area, without considering attenuation of any structure or storage in the line of radiation) is such that unprotected exposure to a very few pulses would be lethal. Hence, it is mandatory that the personnel be within the powered flight station during pulse operation. On the other hand, such radiation intensities do not create severe material problems; most space-compatible materials are relatively unaffected by the radiation accumulated during a typical trip (1,000 to perhaps 3,000 pulses). Radiation-sensitive electronic or other equipment may be either self-shielded or located within the shielded powered flight station.

Care must be exercised that the unexpended nuclear-pulse units themselves are not damaged by excessive radiation. At neutron densities approximately a factor of 10 greater than presently calculated in the pulse-unit storage compartment, pit heating of the nuclear device could become a problem. Although unexpected, should such radiation heating become a problem, the storage compartment could be redesigned to carry the pulse units inverted, which would provide a factor of 10 to 50 additional neutron attenuation as a result of the self-shielding of the propellant slab.

5.4.2. On-board Fire or Explosion

Each nuclear-pulse unit contains a significant amount of high explosive. It is conceivable that some major accident—penetration by a large meteorite or a drastic malfunction of the pulse-unit loading mechanism, for example—could detonate the HE in a unit. The resulting explosion would severely and probably irreparably damage the propulsion module; it would

not, however, cause a nuclear event, as the nuclear devices require multipoint detonation.

An uncontrollable fire in the personnel areas or elsewhere if an oxidizer is available is also remotely possible.

Either fire or explosion under the appropriate conditions could necessitate prompt escape from the basic vehicle. An explosion in the propulsion module would most likely occur during periods of propulsion. In this event the personnel would already be aboard the escape vehicle. They would separate from the mission vehicle and rendezvous with a sister ship, await rescue, or (if fortunately situated) return to earth using the ~2000 ft/sec ΔV available to the escape vehicle.

It appears probable that if a complete propulsion module or vehicle failure should occur, however, it would be due to some less dramatic malfunction than an explosion, uncontrolled fire, or similar catastrophy. In this event the personnel would doubtless elect to stay with the more completely equipped hulk until such time as leaving via the escape vehicle would be most advantageous.

5.4.3. Boost-to-orbit Abort

In the majority of missions considered in this study, an orbital start-up mode of operation is used and the mission personnel are carried to orbit by a chemical booster which is man-rated for the task. For the operational modes in which the nuclear-pulse module boosts itself to orbit, that phase of the operation is unmanned. Any hazards are therefore ground hazards (as discussed in the previous section) or "financial hazards," (as considered in the probability-of-success effects on system costs discussed in Vol. II) and not in-flight hazards to personnel.

One of the methods considered for transporting orbital assembly personnel to orbit was to carry them on the same ELV that delivers the operational payload structure. The personnel would be within an 8-man reentry vehicle, abort-tower equipped, which would be located atop a well-equipped orbital assembly module. Other than this isolated situation, boost-to-orbit safety was considered to be a problem of the earth-launch-vehicle program.

5.4.4. Meteoroid Damage

The nuclear-pulse vehicle is subject to meteoid showers and interactions as are other space vehicles. For a number of important reasons, however, the nuclear-pulse-propulsion module is less vulnerable to meteoroids than chemical or chemo-nuclear systems and thus provides

a greater margin of operational safety for the crew and vehicle system. A foremost advantage is that there is but one relatively dense and compact stage, resulting in a relatively small target. Further, the current weight estimates of the module shell have been based on design requirements which take into account the dynamic pressures experienced during launch and the influence of impulsive loading during nuclear-pulse operation. These requirements, coupled with design considerations, resulted in relatively heavy module shell structures, normally double skinned. Thus the entire vehicle would be considerably less vulnerable to meteoroid penetration or damage than the thinner-skinned pressure structures of many space systems.

An additional durability feature of the nuclear-pulse module is that its invulnerability is further enhanced during coast periods, which, of course, represent the major portion of any particular mission. Upon completion of a nuclear-pulse maneuver, the first-stage shock absorber is compressed after most of its gas is pumped into the second-stage cylinders. A fixed meteoroid bumper is provided for the compressed first-stage shock absorber and, if necessary, extended bumpers can be provided for the second-stage shock absorbers.

Furthermore, because of its inherent design requirements, the pusher is capable of withstanding the impact of any reasonable size meteorite without damage; therefore, the pusher could be positioned in the direction of an oncoming meteoritic shower for shielding much of the vehicle. Using the pusher as a shield, however, would obviously interfere with rotating the vehicle for artificial g; the two concepts are only compatible if there is warning of an impending shower and appropriate allowances are provided in the spin-up propellant.

The magnitude of the meteoroid problem for space travel is as yet unresolved, but, regardless of the outcome, it would be far less significant for the nuclear-pulse vehicles than for any other currently conceived system.

5.4.5. Propulsion Failure

Failure of the nuclear-pulse propulsion module, owing to its essentially mechanical nature, would probably result from failure of one of its mechanical components or of the control system that synchronizes their operation. It is entirely possible that most failures of this nature could be repaired in flight by the crew with on-board equipment and spares. A convenient access way is therefore provided to the propulsion module's internal mechanisms and a relatively large repair bay and spares storage area is provided in the over-all vehicle designs. The current concepts for maintenance and repair of various module components are discussed in the following section.

If an irreparable failure should occur on an exploration mission and the vehicle becomes inoperative, it can only proceed on the trajectory established at the time of failure. In most such instances the mission personnel would elect to leave the vehicle weeks or months after the failure, to take advantage of the more commodious and redundant facilities of the basic vehicle, prior to using the mission-velocity capability of the escape vehicle. The ΔV capability of the post-escape maneuver system provided in this study, using storable liquid propellant, is approximately 2,000 fps. Two to three times this maneuver capability could, however, be provided at a modest weight penalty, since that much additional propellant could also be used for personnel shielding.

For single-vehicle missions it might be highly desirable to have more than 2,000-fps velocity capability in the escape vehicle to provide a higher post-escape survival probability. For two-vehicle convoy missions, however, less than 2,000 fps would be adequate to rendezvous with and join the other vehicle.

The escape-vehicle concept, at the very least, provides mission personnel with a "second chance" in the event of complete propulsion failure of the main vehicle. This second-chance philosophy has been highly effective in maintaining crew morale and sense-of-well-being in various military aircraft.

5.5. SENSITIVITY TO COUNTERMEASURES

Certain of the operational problems discussed above require (or may require) countermeasures to eliminate or reduce potential hazards. Some consideration was given to the cost of such countermeasures, both in system performance and in dollars. An elementary trade-off of performance and cost versus the effectiveness of certain countermeasures was accomplished. The requirement for and effectiveness of most of the countermeasures, however, was too tentative to merit any elaborate cost-effectiveness analysis.

5.5.1. Radiation Shielding

Radiation protection for the mission personnel is one counter-measure that is unquestionably required, but the amount of radiation protection may depend on the penalties incurred. The amount of shielding provided in a nominal design of this study was enough to limit the radiation dose due to propulsion to about 50 rem per mission; this amount of shielding would typically permit a similar dose from solar flares. It was found that an additional 20 g/cm^2 of hydrogenous shielding material over the entire powered flight station would reduce the solar-flare dose by a factor of

about 5 and would reduce the propulsion dose by a factor slightly greater than 5. Therefore, the typical mission dose would be reduced from approximately 100 rem to approximately 20 rem. For the 8-man exploration missions, the additional shielding would add 13,000 kg to the weight of the powered flight station—escape vehicle. This increase in inert weight, for the typical M-2 Mars mission in which 75,000 kg of destination payload is carried, would cause the earth-orbit departure weight to increase from 741,000 to about 790,000 kg. It likewise would cause the mission DOC estimate to increase by about $43 million (or about 4.3 percent, since the total DOC is estimated at nearly $1 billion).

The desirability of decreasing the probable mission dose to some 20 rem may not merit the 50,000-kg penalty in departure weight (i.e., there are probably other inert weight items that would be considered more valuable than dose reduction). However, the data do indicate that significantly greater radiation attenuation, should it be required, can be attained at tolerable penalties.

5.5.2. Fission-product Trappage

Fission-product trappage in the atmosphere, from current pulse-unit designs, would result in radioactive fallout. One way to minimize this problem is to operate in Mode III using chemically propelled earth-launch vehicles so that the nuclear-pulse vehicle is not operated suborbitally. Such an orbital start-up mode of operation, however, is not a significant penalty for exploration missions, since the vehicle mass needs to be built up in orbit anyway to attain a lower and more efficient over-all thrust-to-weight ratio.

For the lunar logistic systems studied, which deliver payload from the earth surface to the lunar surface, the use of chemical boost to orbit results in a performance penalty (since a relatively low performance system is then used for a large part of the mission ΔV). However, the lunar-mission performance data of this study indicate that this penalty is rather modest since the performance loss for chemical boost to orbit is partially offset by the gain due to the use of a lower over-all thrust-to-weight ratio for an orbital start-up system. The resulting cost comparison is illustrated by the mission data in Section 4 of Vol. II.

From the above discussion it is concluded that orbital start-up is an effective countermeasure to fallout for most missions. Each mission will have to be analyzed in more detail, however, in order to assign a quantitative figure to the penalty incurred.

Another and preferable way to overcome the fission-product trappage is to redesign the pulse unit so that fission-product release is largely eliminated. It is expected that such a redesign would result in less costly over-all propellant (due to cheaper nuclear fuel) and probably improved specific impulse as well.

5.5.3. Internal-noise Insulation

The study of internal noise (Section 5.2) indicated that the current vehicle designs provide sufficient stiffness and damping to control low-frequency noise but that some Fiberglas soundproofing treatment may be required for higher-frequency noise. It is estimated that a weight allowance of a few hundred to one thousand kilograms would be adequate to so treat the relatively small powered flight station.

The exploration-system sensitivity to inert propulsion-module weight (reported in Vol. II, Section 4) indicated for a typical Mars exploration mission that departure weight increased 134,000 kg as a result of a propulsion-module weight increase of 40 percent (or 36,400 kg). The increase was essentially linear; therefore, departure weight increases about 3.7 kg for each kilogram of additional inert weight. Similarly, direct operating costs were found to increase at a rate of about $3,300 per kilogram of additional inert weight on such a mission. Thus, an added 1,000 kg of soundproofing for the powered flight station would increase an exploration mission departure weight by some 3,700 kg and increase direct operating costs by some $3.3 million. System penalties of this magnitude, to eliminate the necessity of sound-absorbing helmets or other protective gear, appear quite tolerable.

5.6. IN-FLIGHT MAINTENANCE AND REPAIR CONCEPTS

The nuclear-pulse system offers certain inherent advantages over competitive propulsion systems with respect to in-flight maintenance and repair. One of these advantages is that the system is entirely noncryogenic. As such, all components are readily accessible and easily repairable. Second, the propellant is stored in solid units rather than as a bulk liquid. This obviates the attendant leak and other fluid-handling problems. A third point is that except for the ablative surface of the pusher plate the entire system operates at moderate temperatures ($\leq 200°$ F). This permits the use of steel, plastics, and other nonexotic materials which can be readily machined, welded, bonded, or otherwise reworked as necessary.

It is anticipated that a considerable amount of preoperation inspection of the propulsion system will be desirable, particularly after the long coast periods typical of exploration missions. Hence, access to most of the

internal mechanisms is planned for, since, as already discussed, radiation activation is sufficiently low that it does not constrain even prolonged work with the module's internal mechanisms. Further, a relatively large repair bay—spares storage volume was provided on exploration vehicle designs so that, if necessary, large parts could be taken to the repair bay where repair equipment would be available.

In the current propulsion-module design concepts, reliability has been given considerable thought, with the idea of minimizing the need for maintenance and repair as well as minimizing the probability of propulsion failure. This has been reflected by the use of redundancy in some mechanisms, control subsystems, the employment of fail-safe concepts, and the application of conservative design allowances where redundancy is impractical. It is expected that much more effort will be devoted to high reliability as the propulsion-module design continues to evolve; some noticeably different propulsion-module arrangements have been considered in the past and still other arrangements may yet be found preferable.

Some of the major mechanisms that were considered candidates for preventive maintenance (during preoperation coast periods) or for the carrying of spare parts are as follows: the pulse-unit conveyor and loading system, the lower section of the pulse-unit ejection tube, the ejector-tube breech sections, the ejector-tube protective plasma deflector cone, the ablation-oil spray system and nozzles, the ejector-gas plumbing, gas generation and control system, and the smaller items that are a part of the shock-absorber system. In addition to the pulse-unit conveyor and loading system being repairable, it also has redundancy in operation. A typical design (as shown in Fig. 2.3) has four pulse-unit storage floors, each carrying a conveyor system and a partial or complete breech loading system; in such designs any one delivery system can go out of action without interrupting the pulse-unit delivery cycle during a given maneuver. The out-of-action system can then be repaired during a coast period and redundancy restored prior to the next powered-flight phase.

Other components of the propulsion module which are either too large or not sufficiently critical to carry spares may, of course, still be repairable. Such items include the large shock-absorber elements, the pusher, and numerous tanks carrying liquids, gases, etc. These items can be repaired with hand equipment of the type being developed for space operation or, in some cases, by removing parts which are then rebuilt in the vehicle's repair bay. Tanks carrying liquids and gases would, in addition to being repairable, be in sufficient quantity that puncture or leakage losses would not be critical to the mission.

Maintenance and repair of the operational payload, which includes the spares for personnel, must, of course, also be provided for. No

analysis was performed during this study of what life-support-system maintenance might be required. A complete emergency and reserve life-support system and an emergency power supply system, however, were included in the operational payload (reported in Section 3). The allowances for spares and repair equipment were presumed to include items for the operational payload as well as for the propulsion module.

5.7. FISSIONABLE MATERIAL REQUIREMENTS AND AVAILABILITY

Nuclear-pulse-propulsion systems obtain their high performance from the explosion of nuclear-pulse units, each of which contains an appreciable amount of fissile material (current designs, about 2 kg). The attainable specific impulse and required propulsion-module and structural weights are such that a relatively large quantity of pulse units are required, from one thousand to a few thousand, to cover the range from relatively simple to more difficult missions. Since it is the more difficult type of mission to which nuclear-pulse propulsion seems most uniquely adapted, it is of interest to examine the larger mission requirements in the light of the over-all availability of raw materials and the present and projected production rates.

A more extensive report[*] has previously been issued in which the performance figures were based on larger vehicle sizes (approximately 44×10^6 newton thrust). The data presented here were derived from those in the earlier report but adapted to the smaller nuclear-pulse propulsion modules of this study.

5.7.1. Fissionable-material Availability

The following figures are based on the recent U.S. procurement of about 3×10^7 kg of U_3O_8 annually. This would seem to be a realistic basis since the available supplies are quite adequate for current needs and the weapons requirements are presently decreasing. It is anticipated, however, that the current procurement price of about \$22/kg will be expected to rise over the years because of the utilization of lower grade ores.

The amount of Pu^{239} obtainable from this 3×10^7 kg of U_3O_8, using present conversion techniques (in appreciable breeding and 55 percent extraction efficiency), is about 10^5 kg annually. Present production of Pu^{239}, however, may be estimated at 20 percent of this quantity, or 2×10^4 kg annually, since not all of the ore is used for plutonium production. If large-scale breeding is used (with 40 percent breeding efficiency), the amount of Pu^{239} available from the same 3×10^7 kg of U_3O_8 rises to 10^7 kg annually.

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[*]T. Teichmann, ORION Fuel Requirements (U), General Atomic, Informal Report GAMD-4068, February, 1963, Air Force Contract AF29(601)-2207. (S/RD)

The cost of plutonium is estimated to be between \$10,000 and \$30,000 per kilogram and is predicted to decrease to the range \$100 to \$1,000 per kilogram with large-scale breeding.

An estimate of the quantity of plutonium produced that would be available under different production assumptions and at corresponding plutonium costs per kilogram are shown in Table 5.1. These data are based on the recent U_3O_8 procurement rate of approximately 3×10^7 kg per year. It is estimated that this rate, if continued, would exhaust the readily available (and therefore low cost) supply in about 30 years. However, vastly larger quantities of U_3O_8 become available by paying more for it; by paying 10 times the present cost (approximately \$220 per kilogram), over 100 times the above supply would be available—enough to last thirty-five centuries at the present rate of use. By another factor of 5 in cost (approximately \$1,100 per kilogram), some 1 percent of the total uranium in the earth's crust would become available, which is a million times more than the readily available 30-yr supply assumed for Table 5.1.

Table 5.1

PLUTONIUM AVAILABILITY AND ESTIMATED COSTS
FOR DIFFERENT PRODUCTION METHODS
(Using the current U_3O_8 procurement rate of 3×10^7 kg/yr)

Available Production Method	Plutonium Production/Year	Estimated Cost/kg[*]
1. Conversion of 1/5 of U produced, with 55% extraction efficiency	2×10^4 kg	\$25,000
2. Conversion of all U produced, with 55% extraction efficiency	10^5 kg	\$10,000
3. Breeding of 1/5 of U produced, with 40% efficiency	2×10^6 kg	\$ 250
4. Breeding of all U produced with 40% efficiency	10^7 kg	\$ 100

[*]The factor of $2\frac{1}{2}$ reduction in cost per kilogram due to a factor of five increase in quantity presumes a major amortization increment in the lower quantity costs per kilogram for process development and facilities.

5.7.2. Plutonium Requirements for Selected Missions

Based on present nuclear-explosive-device technology, approximately 2 kg of plutonium is required per pulse unit. This fissile-fuel

requirement is essentially independent of pulse-unit size or yield until the propulsion-module effective thrust approaches 40×10^6 newtons, or well beyond the larger propulsion module (20-m diam; $F_E = 16 \times 10^6$ newtons) of this study. Future development of nuclear devices for propulsion are expected to reduce this requirement by a factor of 2 to 4, particularly for the smaller, lower-yield units, but for this analysis 2 kg of plutonium per pulse unit is conservatively assumed.

Using the requirement of 2 kg per pulse unit, typical mission plutonium requirements were computed in percentage of annual production. The four production methods of Table 5.1 were considered, again assuming the annual rate of U_3O_8 procurement remains at 3×10^7 kg. These data are presented in Table 5.2., along with plutonium costs per kilogram of mission payload based on the plutonium costs of Table 5.1. The fraction of annually produced plutonium used per mission is not disturbing even when present-day conversion methods of production are employed; it becomes very nominal should large-scale breeding be undertaken.

The over-all propellant costs used in this study are based on paying in excess of $60,000 per pulse unit for fissionable material alone, which accounts for about three-fourths of the over-all propellant cost (the remainder being fabrication of the nuclear device and pulse unit plus the other materials used, see Section 8). It is clear that if the raw fissionable material costs drop by a factor of 100 due to breeder-type production, as predicted in the above tables, it becomes a minor part of the over-all propellant cost (some 3 percent instead of approximately 75 percent). Accordingly, it would not be prohibitive to again pay more for plutonium to vastly increase the available supply should really large numbers of nuclear-pulse missions become desirable.

Table 5.2

PLUTONIUM REQUIREMENTS AND COST INDICATIONS FOR REPRESENTATIVE
MISSIONS USING VARIOUS PRODUCTION ASSUMPTIONS

Configuration and Mission Description	Plutonium Requirements in Percent Annual Production and Plutonium Cost per Kilogram of Payload			
	Production Method A (Conversion of 1/5 U at 55% efficiency)	Production Method B (Conversion of all U at 55% efficiency)	Production Method C (Breeding of 1/5 U at 40% efficiency)	Production Method D (Breeding of all U at 40% efficiency)
10-m Configs. (I_{sp}=1850) Lunar logistic (1,180,000-kg payload, 2,943 pulses required)	29% $125/kg	6% $50/kg	0.3% $1.3/kg	0.06% $0.5/kg
Mars exploration (156,000-kg initial PL, 2,782 pulses required)	28% $894/kg	6% $363/kg	0.3% $9/kg	0.06% $3.6/kg
20-m Configs. (I_{sp}=3150) Mars logistic (3,000,000-kg payload, 2,324 pulses required)	23% $37.5/kg	4% $15.4/kg	0.2% $0.37/kg	0.04% $0.15/kg
Jupiter exploration (392,000-kg initial PL, 8,291 pulses required)	83% $1,060/kg	17% $330/kg	0.8% $10.6/kg	0.17% $3.3/kg

107

6. GROUND COMPLEX AND OPERATIONS

The ground facilities, support equipment, and operations required to prepare nuclear pulse vehicles for space missions have been investigated in a preliminary way. Both the 10-m and 20-m configurations were considered for the three operational modes previously discussed. The use of both Saturn V and post-Saturn vehicles was considered, the latter being confined to a brief consideration in support of the 20-m vehicle operations.

The compatibility of the 10-m vehicle configurations with the Saturn V was established and discussed in some detail in Vol. II, Sec. 4, as were ELV requirements for the 20-m configurations. The compatibility of the 10-m vehicles with Saturn V facilities is considered here, and the necessary modifications and additions are shown to be relatively minor. Extensive revisions of Saturn V facilities would be needed to support the 20-m vehicle, but it appears that this vehicle is readily adaptable to the facilities proposed in the studies of several post-Saturn launch vehicles.

A brief consideration is given to remote or outlying launch sites located some distance from the present Cape Kennedy site. Finally, the selection of success probability factors and their effect on operations are discussed, in a more general way than in the mission cost section (Vol. II, Sec. 4).

6.1. 10-METER VEHICLES

The 10-m vehicles were designed to be compatible with the Saturn V ELV in all three operational modes defined for nuclear pulse vehicles. In operational Mode I (self-boost to orbit or beyond after being lofted above the atmosphere, with no orbital refueling), the S-1C stage is used as a lofter. In the present study, this operational mode is confined to lunar logistic missions when the 10-m vehicles are used. In Mode II (orbital loading-up with propellant and/or payload after self-boost as in Mode I), the S-1C is again used as a lofter while the two-stage Saturn V is used for orbital deliveries of the needed logistics. In Mode III (orbital start-up, after orbital mating of the propulsion module, the operational payload, and one or more loads of propellant), all three kinds of ELV payloads are delivered to a rendezvous orbit by the two-stage Saturn V.

108

The compatibility of the normal Saturn V facilities, support equipment, and operations are discussed below.

6.1.1. Facilities

The adaptability of a nuclear pulse propulsion module to facilities which were built to support the chemical Saturn stages was considered to determine facility compatibility. Such support areas as the stage preparation building, the vertical assembly building, the pay-load area, the launch pad, and the nuclear-pulse-unit loading area are the principal areas of concern in determining the extent of required facility modifications and/or additions.

6.1.1.1. Stage Preparation Building.

It is proposed that for operational Modes I or II, support operations performed on the nuclear pulse propulsion module prior to its assembly with the S-1C would be made in the low bay building normally housing S-II preparation activities. The envelope of the module, 10 m in diameter and approximately 21 m long, is within that of the S-II stage, and the 175-ton overhead crane in the area is more than sufficient to support the 100-ton module weight. Minor modifications to the access platforms and the track-mounted dolly for transferring the module within the building may be necessary. The extent of test, modification, and checkout equipment (though different) would not appear to exceed similar requirements of the S-II stage.

For Mode III operations, a separate propulsion module pre-paration building will be required, since the S-II stage is to be used, as are its facilities. This building area would provide for receiving and inspection procedures as well as checkout operations to be performed on the module after receipt and before delivery to the vertical assembly building. The size of the bay and the support equipment required could be patterned from the S-II stage low bay.

6.1.1.2. Vertical Assembly Building.

Assembly of the pulse propulsion module to the S-1C or S-II stage would be performed in the high-bay area of the vertical assembly building (VAB). A 250-ton bridge crane in this area would assist in the mating operation. Assembly of the operational payload to the spacecraft for Mode II operation as well as installation of the interstage adapter and the nose fairing would also be performed in this area. No major modifications in the area are foreseen.

6.1.1.3. Payload Area.

A building to house the spacecraft operational payload prior to its assembly with the launch vehicle would be required. This area should support receiving and inspection operations after delivery from the manufacturer as well as test and checkout

operations before its delivery to the VAB. The building could be located either at the industrial area on Merritt Island or in the vicinity of the vertical assembly building. In either case, the building should be sufficiently large to house two completely assembled payload sections as well as support equipment for test and checkout. The independent verification of its readiness requires a minimum of launch preparation time for the vehicle after complete assembly and results in minimum turn-around times.

6.1.1.4. Pulse-unit Storage Area. Although the probability of an inadvertent nuclear event due to the nuclear pulse units is too remote to consider in the facility design, the possibility of an accidental high-explosive occurrence from this source should be considered. While it appears unlikely that such an event would occur, for safety considerations it is proposed that a conventional ordinance storage facility at a separate area in the vicinity of the launch pad be used to store the required propellant supply. Access to this area would be required only by the propellant delivery carrier (barges) and by the propellant loader (arming tower).

6.1.1.5. Launch Pad. It appears that the Saturn V launch pad structure would require no additions or modifications to support the launching of the nuclear pulse vehicles. An over-water launch pad has been suggested (Sec. 5), however, as one possible countermeasure to surface contamination in the event of an ELV failure and a resulting fire on the pad (i. e. , submersion of the nuclear pulse vehicle and/or nuclear pulse units should quench and avoid HE explosion of the pulse units). Locating the launch pad off-shore in some 40 ft of water would obviously amount to a significant change. The operational and cost implications of such a move have not been evaluated.

6.1.2. Support Equipment

Support equipment such as checkout, assembly, and launch equipment used for Saturn V operations in most cases is applicable to pulse-propulsion spacecraft use. Items for transportation unique to the pulse-propulsion module must by necessity be acquired. Some of the equipment, such as the arming tower and the rail-mounted dollies in the low-bay VAB, would require some modifications.

6.1.2.1. Stage Transporter. A separate transporter to carry the nuclear pulse propulsion module is required. Ground transfer operations must be performed between the manufacturer and a dock site, the vehicle assembly area dock and the VAB low bay area, and within the VAB. Although the transporter must carry a large weight (100 tons), the module, with its low center of gravity and its ruggedness, does not

cause difficult transfer design problems. The nonpowered transporter (its function 99 percent of the time is only to support the propulsion module) would rely on two vehicles for transfer maneuvers. The module would be transferred in a vertical attitude. Such a position is desirable because stage rotation procedures would be unnecessary and the module can best be supported in this manner. The module would be fixed to the transporter by special hold-down structures. A possible transporter design is shown in Fig. 6.1.
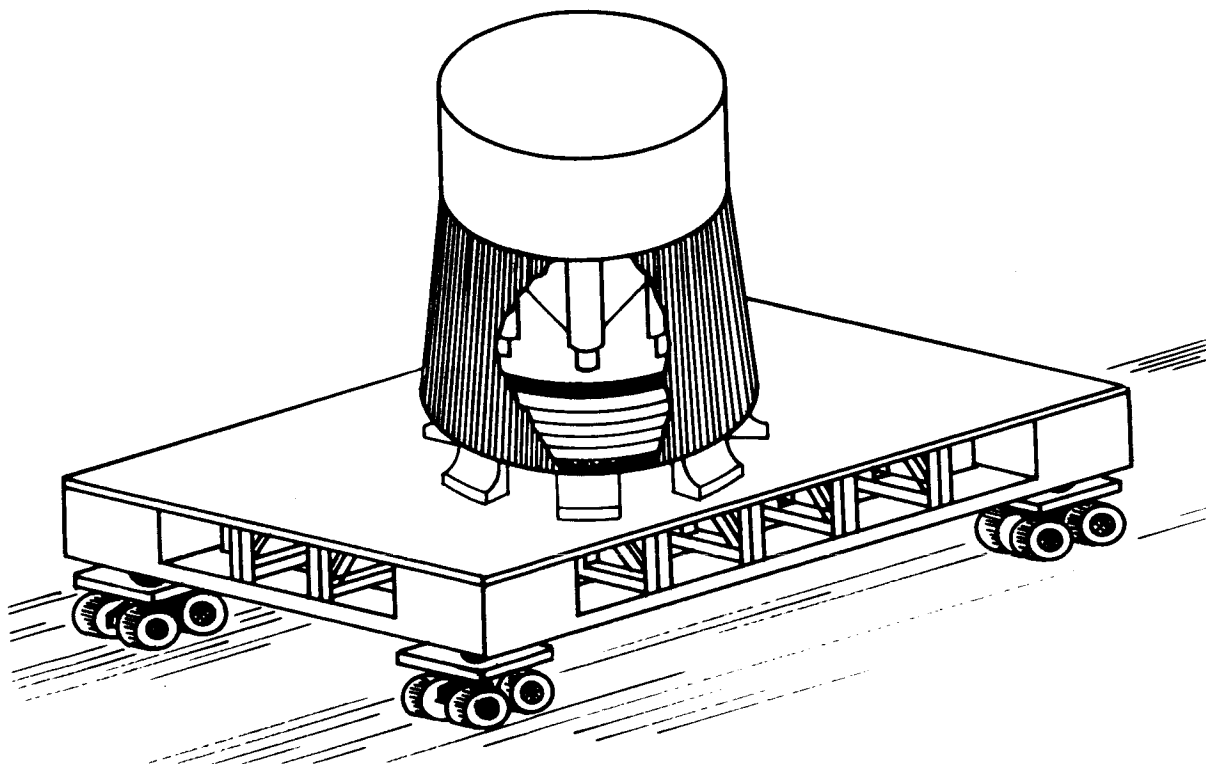
Fig. 6.1--Conceptual transporter for the nuclear pulse propulsion module

6.1.2.2. Payload Transporters. The operational payload's major structure for the 10-m exploration vehicles is a 7.2-m (23.6-ft) diam cylinder some 17.5 ft high which is mounted atop a 3.2-m (10.5-ft) spine about 46 ft high. Its weight, with equipment, is only some 21,000 lb. The 23.6-ft diameter is too large for air transportation unless larger "guppy" aircraft continue to be produced for space vehicle stages. Should such aircraft be available by the time period required, air transportation can be used in lieu of barges. Relatively simple rubber-tired transporters would carry the payload or payload sections in a horizontal attitude to and from the aircraft or the dock site.

6.1.2.3. <u>Crawler-Transporter</u>. The crawler-transporter used to transfer the Saturn V vehicle between the VAB and the launch pad could also be used for nuclear pulse vehicle payloads if reconfigured. The loads to be carried by the transporter would then have a smaller envelope than the Apollo configuration and therefore less wind loads would need be reacted. Although the dry weight of the pulse-vehicle configuration (approximately 820,000 lb maximum for Mode I or II loft operations) may exceed the dry weight of the Apollo vehicle, the increased static load on the crawlers is only 5 percent more than normal. The crawler-transporter "as is" therefore appears compatible with pulse-vehicle use.

6.1.2.4. <u>Arming Tower</u>. The mobile arming tower will require major modifications when used to support nuclear-pulse operations. It is proposed that this structure be used to transfer the nuclear-pulse units from their storage area to the spacecraft at the launch pad. Two methods of pulse-unit loading must be employed. The propulsion module includes an area which can contain 900 pulse units. These items must be loaded individually for Mode I or Mode II operation; this would be done largely by automatic handling mechanisms built into the propulsion module. The propellant-carrying ELV, on the other hand, transports the pulse units already assembled together in propellant magazines, each holding approximately 80 pulse units. A total of seven magazines is the maximum load that is assumed to be placed on a propellant-carrying ELV. Derrick cranes with a capacity of 30,000 lb would be needed to install the propellant magazines. The existing 40-ton crane atop the arming tower could not be used to install the magazines on the vehicle because of access limitations, but it would be used at the storage area for lifting the magazines to the proper tower level. A sketch of a revised arming tower is shown in Fig. 6.2. Additional investigation is needed to determine the extent of structural reinforcement which should be added to the tower.

6.1.2.5. <u>Transport Barges</u>. Transport of the propulsion module over any significant distance would be performed by barge because of the module size. For this mode of transportation, transfer would not be difficult nor would the barge design prove a problem. It is also assumed that barges would be used to transfer pulse units from their manufacturer to the storage area. The size of the barges used to transfer the module or the pulse units could be such as to permit the use of intercoastal waterways.

6.1.3. <u>Operations</u>

When operating in Mode III, as is assumed for most of the exploration missions of this study, all nuclear-pulse-vehicle components

USE DERRICK CRANE TO
LOAD MAGAZINES FROM
STORAGE AREA TO TOWER

ADD SPECIAL HANDLING
EQUIPMENT TO LIFT AND
LOAD MAGAZINES FROM
TOWER TO LAUNCH VEHICLE

ADD TOWER PLATFORMS TO HOLD
AND CARRY PULSE UNITS
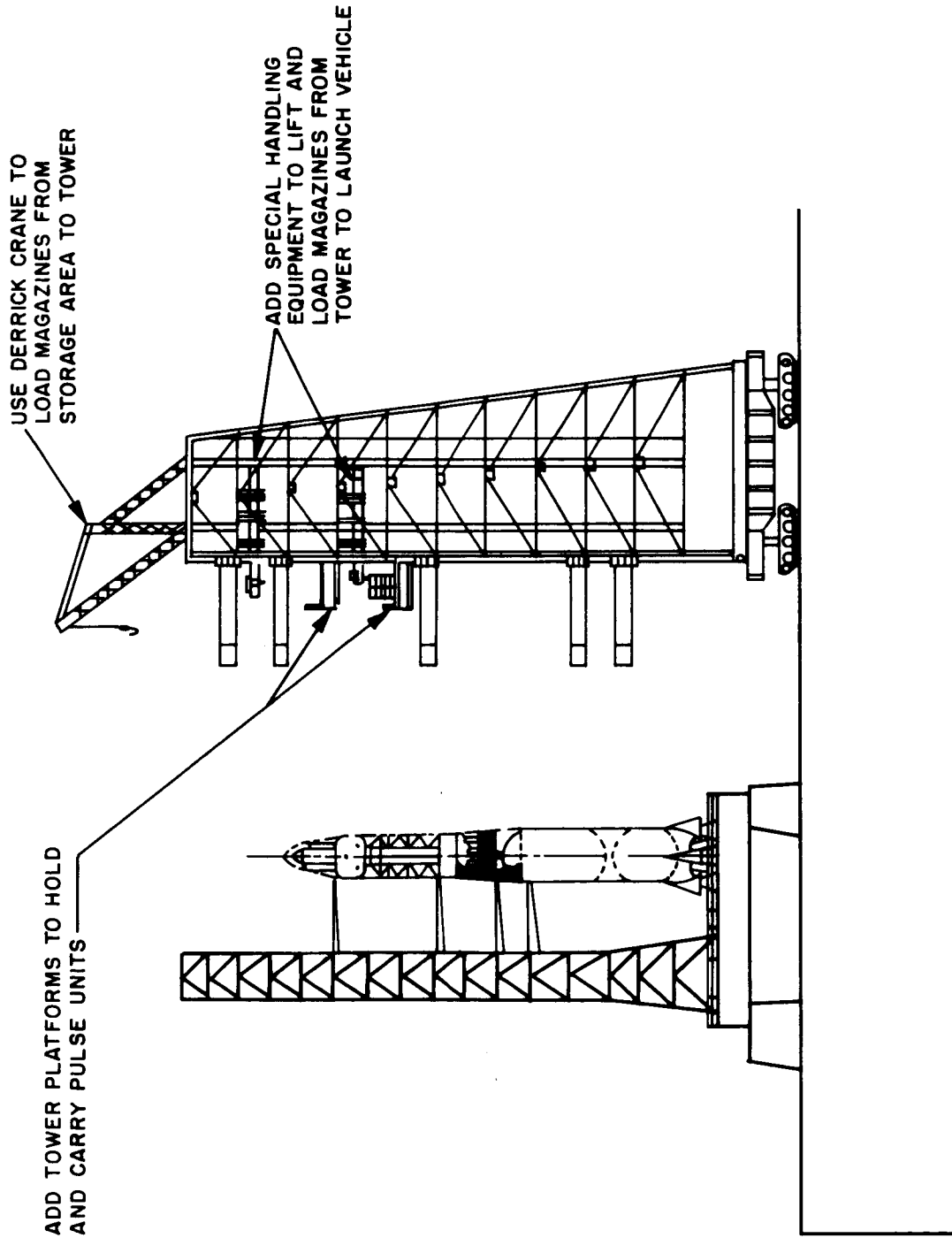
Fig. 6.2--Conceptual arming tower modification for nuclear pulse 10-m vehicle operations

are delivered to orbit by the two-stage Saturn V. During ground operations these components are treated as any Saturn V payload. No special treatment is required other than nuclear safety precautions when loaded propellant magazines are the payload.

The most significant difference from typical ground operations occurs when operating in Mode II. For this event, a possible sequential order of operations needed to ready the nuclear pulse vehicle for launch is outlined here. Assume that the vehicle plus its extra propellant must be launched in two sections: the assembled vehicle lofted by an S-1C and additional pulse units in magazines to be delivered to orbit by a Saturn V. Initially, the S-1C stages, the S-II stage, the pulse propulsion module, the stage adapters, and nose fairings arrive at Cape Kennedy by barge and are transferred to their respective areas. The two S-1C stages are delivered to the high bay area of the VAB, assembled to the launcher platform, and checked to prove launch readiness. The S-II stage and the pulse-propulsion module are towed to the low bay areas of the VAB and loaded aboard the rail-mounted dollies for checkout. (The pulse-propulsion module, already in the vertical position, could be checked while remaining fixed to its transporter after the transporter is moved into one of the S-II bays.) After its delivery to the Cape, the operational payload is transferred to the payload test building at Merritt Island. When ready, the pulse-propulsion module and the S-II stage are transferred to the high bay area of the VAB and mated to the S-1C stage adapter. Assembly of the operational payload section is made with the pulse-propulsion module after its delivery from the payload area. Assembly of the nose fairing and complete vehicle checkout precedes the launcher-transporter transfer to the launch pad. During this time, the arming tower will have loaded itself with the proper amount of pulse units from the storage area. It is then transferred to the launch pad and the pulse units are loaded aboard the vehicle by handling equipment installed on the tower. On completion of nuclear propellant transfer, the final launch fairing is assembled and the vehicle is ready for launch. Similarly, the S-1C/S-II vehicle, used as the propellant carrier, is checked while in the VAB with the launch fairing partially assembled. After being transferred to the launch pad, propellant magazines are loaded by the arming tower prior to the complete assembly of the launch fairing and the nose cone.

6.2. 20-METER VEHICLES

The facilities and support equipment needs for 20-m-diam nuclear-pulse vehicles were surveyed to determine their compatibility with existing and planned Cape Kennedy items. In the investigation, all three modes of operation were again considered. Mode I and II operations

could conceivably be carried out with either the Saturn S-1C or a post-Saturn stage as the lofter, with additional Mode II orbital deliveries using either Saturn V or a post-Saturn ELV. The compatibility determination was therefore based on either Saturn or post-Saturn facilities.

The propulsion module of the 20-m vehicles has a dry weight of 360,000 kg (793,000 lb). This propulsion module, equipped with a minimum upper vehicle to provide guidance and directional control and the necessary operating fluids and nuclear pulse units, can be lofted by an S-1C to deliver itself plus some 300,000 lb into earth orbit. The remaining sections of the operational payload and additional propellant or magazines can be delivered to orbit by the two-stage Saturn V. Earth orbit departure requirements for several 20-m vehicle missions were described in Sec. 3 of Vol. II; the number of successful Saturn V launches required was determined by allowing not more than 100,000 kg (220,000 lb) for any one Saturn V delivery.

6.2.1. Support Areas

The engine preparation building for the 20-m propulsion module would necessarily be an addition to Saturn facilities. The S-II low bay area would not be adaptable to pulse-vehicle use as would be the case when the 10-m module is used. Not only is this area sized to support vehicles half the size of the 20-m module, but support equipment, cranes, and work platforms designed for use with 33-ft-diam modules would prove unacceptable. A similar condition would exist in the high bay VAB. Although the assembled vehicle envelope would not exceed the volume limitations of each bay, the hammerhead configuration requires modifications to work areas supporting upper stage checkout. A major limitation in this area is due to the overhead bridge crane capacity. At 250 tons, this crane is not designed to lift such items as the 400-ton propulsion module. As these crane loads are reacted by the building, modifications to increase the lift capacity of the crane might also require reinforcement for the building structure. The operational payload preparation building and the pulse-unit storage area would be as for the 10-m vehicle although greater in size to support the larger requirements. No changes at the launch pad would be needed to support the 20-m configuration.

6.2.2. Support Equipment

The launcher-transporter which delivers the Saturn V vehicle from the VAB to the launch pad would also be required to support significantly greater loads when the 20-m propulsion module is to be carried by the S-1C stage. The frontal area of the vehicle, and therefore the wind loads to be reacted by the transporter, would be increased by a factor probably greater than two. In addition, the static load of

the dry lofter/pulse vehicle is 10 percent greater than the Saturn Apollo vehicle used for transporter design criteria. Although it is questionable that the launcher/transporter "as is" could support Mode II operations for the 20-m configuration, the extent of modifications needed to make it acceptable are not known.

Transporters to support the propulsion module during transit between manufacturer and the VAB would consist of a structural platform to hold the module in a vertical position and four crawler units at each corner to provide mobility. Transporters for the various portions of the operational payload would be made up of structural frames mounted on rubber-tired "bogeys." Modifications to the arming tower in addition to those required for the 10-m vehicle would consist of adapting the access platforms at the upper levels to 20-m diameters.

6.2.3. Operations

The events leading up to the launch of the 20-m configuration are similar to those for 10-m vehicle operations. For the 20-m vehicles, probably no sections of the spacecraft could be airlifted and all would be transported by barge. Preparation of the propulsion module would be performed in a separate building adjacent to the VAB, and assembly of the vehicle would be made in the modified high-bay area. Launch and support of the operational payload and propellant-carrying ELVs would be exactly as for 10-m vehicles.

Saturn V and its facilities appear to be marginal for use in support of Mode II operations of the 20-m vehicles. Saturn V feasibility is questionable not only because of the extent of modifications and additions required by the facilities and support equipment, but also because of the many ELVs required to place such spacecraft into earth orbit.

6.2.4. Post-Saturn ELVs

Two types of launch facilities have been considered for use with post-Saturn ELVs. The mobile assembly method, which would be patterned after Saturn V facilities, would have separate preparation areas for each part of the vehicle, assembly of these sections at a VAB, and transfer to a remote launch pad. The semi-fixed pad system requires individual preparation of each stage and assembly of stages and payload at an assembly building which also acts as the launch pad. Such facilities, which would be designed for the chemical two-stage vehicle and a 10^6-lb payload, would not need to be expanded for the 20-m pulse-propulsion module. Sections of the interplanetary vehicle would be treated just as payload for post-Saturn ELVs when Mode III operations are followed.

The feasibility of launch facilities for this type of vehicle (which would not be exceeded for pulse configurations) has been studied previously. *

6. 3. REMOTE LAUNCH SITE CONSIDERATIONS

A launch site not located at Cape Kennedy has been considered as an alternative for Mode I and Mode II operations. A possible launch site located along the coast of Georgia would be preferable to avoid nuclear-pulse propulsion while in the geomagnetic trapping regions. In this event, vehicle performance would not need be reduced because of hazard limitations which might require nonoptimum azimuths or "dog-leg" maneuvers, yet the site would be close enough to permit use of the Cape area as a base of operations. This site would be used only for those vehicles which required self-boost-to-orbit. ELVs to deliver propellant or operational payload sections into orbit would use existing facilities at the Cape and be launched from that area.

The assembly area of the Saturn facility would continue to be the operations base for remote site activity. The S-1C, pulse-propulsion module, and payload would be prepared and checked as already stated at their areas and assembled at the VAB. Completion of readiness operations would be followed by transfer of the vehicle to the launch pad. But in this case, instead of traveling a few miles, the vehicle would have to be transferred some 200 miles or more. Such a condition could be met by maneuvering the crawler-transporter at the assembly area to a specially constructed barge at the dock area. After loading the crawler-transporter with its launch vehicle aboard the barge and securing them to each other, the barge would be towed by tug up the intercoastal waterway to the Georgia site. There the launch vehicle would be rolled off the barge and transferred to the launch pad. Launch pad operations would be similar to those at the Cape.

Such an extension to the Cape facilities requires duplication of all of the launch pad requirements. The pad structure, launch control center, arming tower, and propellant loading and storage facilities would have to be copied at the remote site. But by utilizing the VAB and stage preparation areas of Complex 39, the cost of such a remote site could be minimized.

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*Nova Launch Facility Study, The Martin Company, Denver Division, Report NASA-CR-63-72, September, 1963 (Confidential).

A Study of Post-Nova Launch Vehicles, Phase III, General Dynamics/Astronautics, Report AOK 63-035, December, 1963 (Unclassified).

Phase III. Second Progress Briefing, Post-Nova Launch Vehicle Study, Douglas Aircraft Corporation, Report SM-45793, December, 1963 (Confidential).

6.4. SUCCESS PROBABILITY FACTORS

Mode II and Mode III operations require that a number of orbital operations be performed in order to obtain an orbiting, "mission-ready," interplanetary vehicle. For each orbital operation as well as each launch, there is some probability that the required event will not successfully be achieved. Therefore, the more operations necessary to ready the vehicle for orbital departure, the less likely is the chance that this condition will be attained. The unsuccessful performance of a required event would result in an incomplete interplanetary vehicle and additional or redundant events would then be required. Determination of the number of redundant events required in preparing the vehicle for orbital launch is a necessity because each redundancy causes an additional ELV requirement. Such information is needed to generate cost and schedule estimates as well as to compare vehicles and their feasibility.

After listing all of the operations necessary to prepare the interplanetary vehicle, the next step is to assign reliability factors to each event. As the required operations are a series of events, the probability of successfully preparing the vehicle can be ascertained by a cumulative multiplication of the reliability factors. The resultant product, the probability of success, must then be compared to some value which has been set as a goal. If unacceptable, redundant events must be added until a satisfactory level of success probability can be attained.

Such a procedure has been followed in determining the requirements for nuclear-pulse interplanetary vehicles. Depending on the configuration, the required events consist of the over-all vehicle delivery (the separately delivered propulsion module, operational payload, and propellant), vehicle mating, and propellant loading. The following reliability (success probability) factors were assigned to their respective events:

Orbital Mating of Operational Payload
  to Propulsion Module . . . . $P_M = 0.97$
Orbital Mating of Payload sub-units . . . . $P_{M-1} = 0.99$
Orbital Propellant Transfer (7 magazines) . . $P_F = 0.98$
Orbital Delivery and Rendezvous . . . . $P_D = 0.85$
(on occasion, 0.95)

The goal for the probability of successfully preparing the vehicle (probability of mission readiness) was set at $P_S \geq 0.75$. The total number of launches required to attain at least this probability of mission

118

readiness has been determined for typical interplanetary configurations using either Mode II or Mode III operations with a Saturn V or a post-Saturn ELV. These data were used in computing the direct operating costs of Sec. 4, Vol. II. Requirements were computed for both single-vehicle trips and two-vehicle convoys. The number of required Saturn V ELVs for selected earth-orbit departure weights and a comparison of different values of the most significant parameter, P_D, are shown in Fig. 6.3 for Mode III operations and in Fig. 6.4 for Mode II operations. Also shown is the minimum number of ELVs required to prepare the vehicle, assuming that every delivery is successful ($P_D = 1.0$). The estimates and assumptions concerning the ELV payload capability stated in Sec. 4 are applicable. Although no justification of the individual success probability factors is given, it is felt that the values used are at least representative. A point which should be made is that the number of events required to prepare a specific mission vehicle (or vehicles) for orbital departure is a most significant factor in planning for mission readiness.
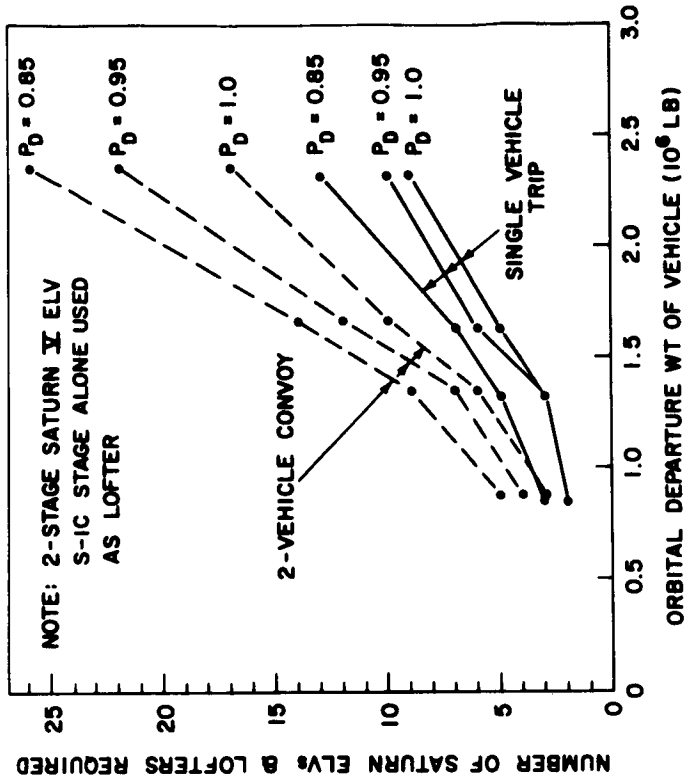
Fig. 6.4--Mode II operation launch-vehicle
requirements for selected earth-orbit depar-
ture weights (exploration missions)
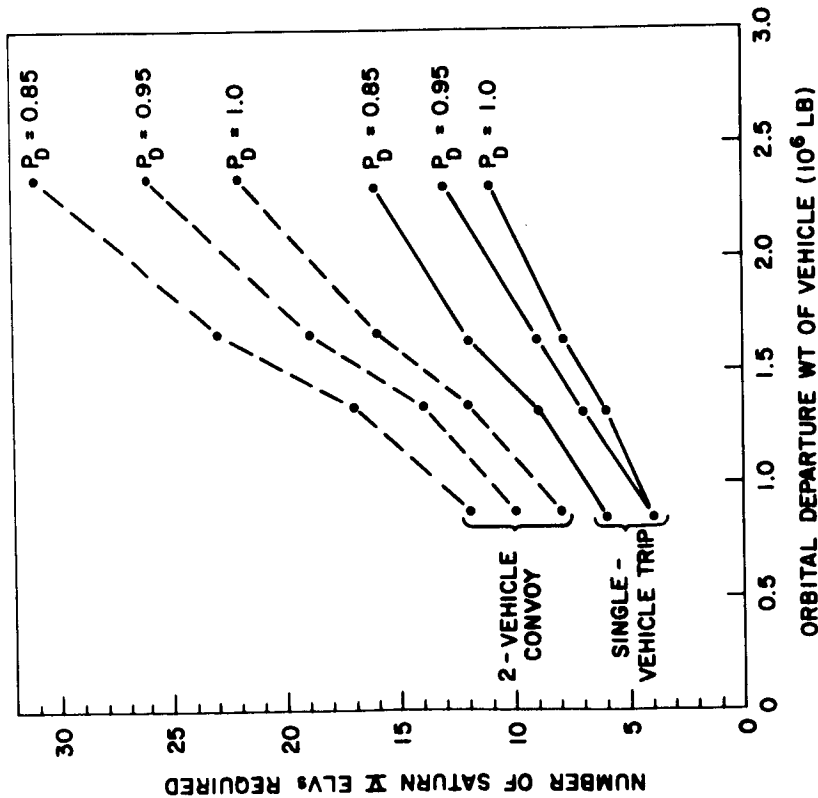
Fig. 6.3--Mode III operation launch-vehicle
requirements for selected earth-orbit depar-
ture weights (exploration missions)

7. DEVELOPMENT PROGRAM PLANNING

A plan for the complete development of a nuclear-pulse module has been prepared and an over-all cost estimate made. The requirements for the development of this system, as with any system, are closely related to the conditions of the system operation. Some of these more important characteristics are: (1) the propulsive energy source consists of a repetitive identical special nuclear explosion that is remote from the main propulsion system; (2) nuclear and nonnuclear environments and effects appear to be completely separable; (3) neither high nor low-temperature extremes exist internal to the propulsion system, and high-temperature ablation problems are constrained to a thin surface region of the propulsion module pusher plate; (4) nuclear radiation effects after powered flight are quite low. The implications of these operating characteristics on the development include the practicality of separating development problems such that a large percentage of development can be carried out on the ground utilizing high-explosive loading and underground nuclear testing; the handleability of test specimens after exposure to nuclear-energy pulse units (low induced radioactivity); and no cryogenic or high-temperature environment or subsystem need be developed. All of these conditions simplify the development and certainly help to keep the development cost down.

Although the development program presented here is designed around a 10-m-diam nuclear-pulse module, it should be emphasized that all of the development techniques employed, with suitable modifications, can be equally useful in the development of other size, both larger and smaller, propulsion modules. For any pulse period there is a maximum module diameter for which a repetitive high-explosive (HE) pulsing technique can be utilized to simulate synchronized operation; this diameter could very well be more than 20 m for repetitive rates of 1 to 2 sec.

The cost data presented were based on estimated costs for (1) the design, (2) construction and fabrication of all test specimens, (3) test facilities and test materials, (4) direct flight costs, which include lofter and booster costs as anticipated for the scheduled flight time, (5) launch complex modification required for this program, and (6) special test personnel training and life support for in-orbit tests. Furthermore, no allowance was made for inefficiencies resulting from

121

crash programming or vacillating project direction and no escallation was considered.

7.1. <u>OBJECTIVE: 10-M VEHICLE</u>

The aim of this development plan is to bring to a point of initial operational capability (IOC) a propulsion module for a nuclear-pulse vehicle (NPV) that has the specific impulse required for carrying out a variety of manned space tasks.

Recent technical developments and applications studies have resulted in a nuclear-pulse-propelled-vehicle size which appears to be a reasonable (economical) development objective that would result in an impressive early operational capability for certain classes of manned space missions. Both the 10 m diameter and the gross weight of this nuclear pulse propulsion module (NPPM) are compatible with a Saturn V for launch to orbit. The 10-m NPPM chosen for the development goal has been used in studies to establish schedule and cost criteria.

7.2. <u>DEVELOPMENT APPROACH</u>

A logical step-by-step development program progressing from the development of components to preliminary flight rating and qualification can be carried out in fairly well defined predevelopment and development phases, and each phase will provide answers to definite operational problems. Fig. 7.1 is a representation of the relationship between the major development areas and the development phases.

The basic development philosophy is that each element, system, and total assembly of systems which make up the module shall be tested and proved under practical and realistic conditions in ground facilities to the maximum extent. Furthermore, because of the unique operating conditions of the nuclear-pulse system, the nuclear, thermal, and mechanical effects can be treated separately; therefore, the conditions of loading can be applied to separate components and then to full propulsion modules with meaningful results. Different specific experimental techniques can be applied to each problem area with redundant results. The known basic problem areas and the experimental techniques which may be applied to study and prove each component are shown in Fig. 7.2. The separation of nuclear, thermal, and mechanical effects makes separate, but related, problems of the development of the pulse unit and of the propulsion module. Fig. 7.3 shows schematically the separability of the NPPM development problems. The pulse unit and the propulsion module must be developed concurrently, however, because
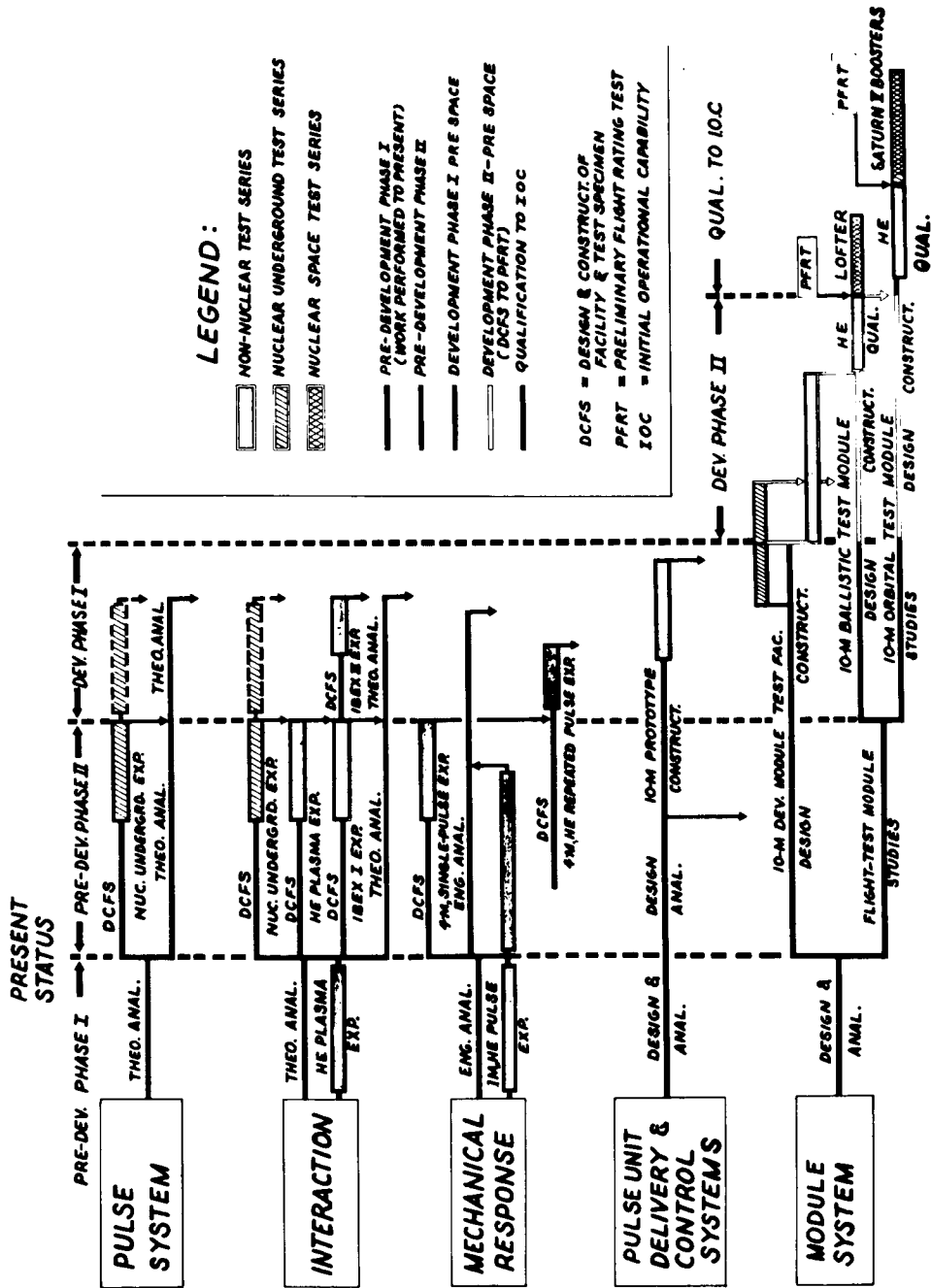
Fig. 7.1--Development phases

EXPERIMENTAL TECHNIQUES

DEVELOPMENT AREAS	HE PLASMA	EM PLASMA	COMP. TEST (VACUUM)	HE PULSE	HE REP. PULSES	NUCLEAR UNDERGROUND PUL.UNIT	NUCLEAR UNDERGROUND SCALED	NUCLEAR UNDERGROUND MODULE	NUCLEAR REP. BALLISTIC	NUCLEAR REP. ORBIT
1. PULSE UNIT EXPANSION										
1a. PLASMA/PUSHER DIAGNOSIS						●●		●	●●	
2. INTERACTION EFFECTS	●	●					●	●	●●●	
2a. REP. INTERACTION EFFECTS	●	●								
2b. PUSHER ABLATION PROTECTION			●				●	●	●●●	
3. PUSHER RESPONSE				●	●		●	●	●	
3a. OFF-DES PUSHER RESPONSE				●	●					
3b. REPEATED PUSHER RESPONSE					●				●	
4. S.A. & ATTACH. RESPONSE			S.A. ONLY	●	●		●	●	●●	
4a. REP. S.A. & ATTACH. RESPONSE			S.A. ONLY		●					
4b. S.A. COOLING			●	●	●				●●	
5. PULSE UNIT HANDLING			●		●				●●●●	
5a. PULSE UNIT DELIVERY			●		●					
5b. PULSE UNIT POS. EFFECTS					●					
5c. ARMING & FIRING			●		●	●			●●●	
6. MODULE DYNAMICS					●			●	●●	
6a. MODULE FLIGHT CONTROLS					●					
7. THRUST VECTOR CONTROLS					●				●	
8. VEHICLE MISS. QUALIFICATION					●			●	●	●
YEAR REQUIRED	0	2	1 TO 6	0	4 ON	2 TO 5	2 TO 5	6	10	11
YEAR AVAILABLE	0	2	1	0	2	1	1	3	4	7

Fig. 7.2--Development approach

124

the results of the development of one are design parameters for the other. Each is developable to a high degree of reliability prior to costly nuclear space tests. The experimental techniques to be used in this program are discussed in the following sections.
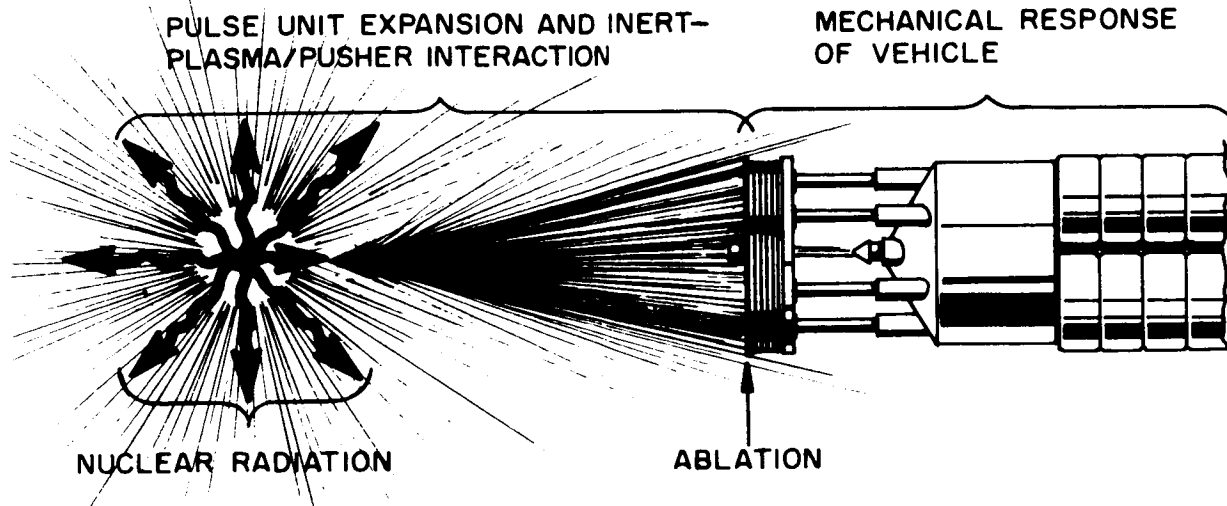
Fig. 7.3--Separability of nuclear-pulse problems

7.3. DEVELOPMENT TECHNIQUES

7.3.1. Ablation Experiments with High-explosive Plasma Generator

A high-velocity, high-density plasma generator has been developed to investigate experimentally plasma-target interactions similar to those of the propellant-pusher interactions of the NPPM. The test setup is shown in Fig. 7.4. The plasma is produced by a high-explosive (HE) cylindrical implosion of a thin-walled metal (usually lead) tube. Stagnation of the high-velocity plasma against a target produces hydrodynamic impulses (pressure as a function of time) similar to those expected from a nuclear pulse unit. The velocity of the plasma is lower than that of a pulse unit, but the temperature of the stagnated plasma is high enough (7 to 8 ev) that radiation is the dominant mode of energy transfer from the plasma to the target, and thus the interaction can be applied to the propellant-pusher interaction conditions for which theoretical calculations apply.

The two main objectives of this experimental program are
(1) to develop techniques and instrumentation with the HE plasma generator for use in nuclear tests,(2) to check theoretical calculations of

the ablation process and to study effects (such as radial flow and con-
vective instability) that may not be amenable to calculation.
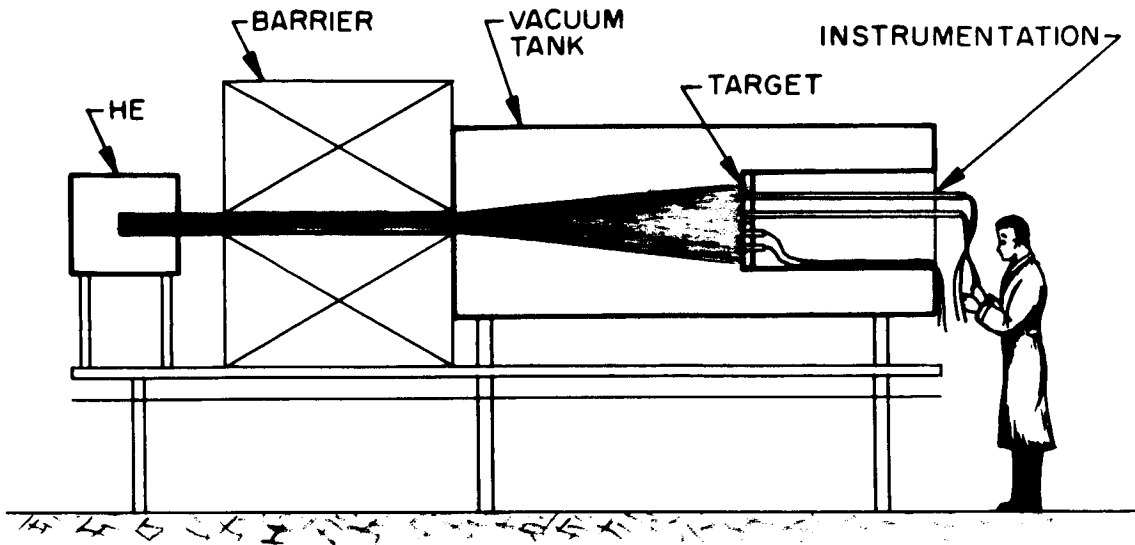
Fig. 7.4--HE plasma generator

7.3.2. Pulsed High-energy Electromagnetic Plasma Accelerator

A proposed advanced plasma acceleration system which should
accurately simulate in one dimension the propellant-pusher interaction
will be used to test target plates up to 6 in. in diameter. The basic
acceleration mechanism is magnetohydrodynamic in nature. Explosive-
electric generators (IBEX) are to be used to obtain the high total plasma
kinetic energy. These generators, which are being designed and
developed at General Atomic for specific applications, will provide high
electrical energy which will then be converted into plasma kinetic
energy by a multiple-rail-gun system. The necessary high-current
switching system has already been developed. The plasma densities
within the separate coaxial rail guns should be comparable to those
which are presently used in an operative rail-gun system at General
Atomic. By focusing the plasmoids generated by the separate rail guns,
the plasma densities required for the interaction tests may be obtained;
the energy losses upon convergence are not expected to be large   This
technique has been experimentally established. The advanced IBEX
system is shown in Fig. 7.5.
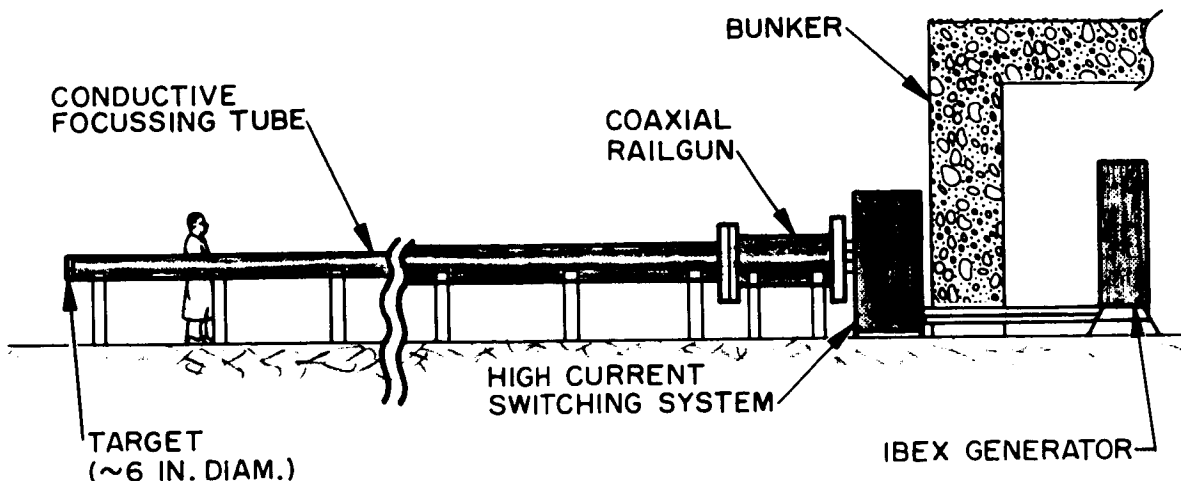
Fig. 7.5--Pulsed high-energy electromagnetic plasma accelerator

7.3.3. Space-simulated Component Tests

The development of the NPPM will require a broad range of
tests of materials, elements, components, and systems in various
degrees of simulated atmospheres prior to the final assembly of a pro-
totype. These tests will be in addition to the very specialized tests for
pulse-unit expansion, pusher ablation, and propulsion-module mechanical
response, which are discussed separately. Most of the more or less
standard tests can be made with standard techniques on existing lab-
oratory equipment. Strength tests of tensile, compression, and torsion
specimens under one-time fatigue and nuclear radiation conditions, tests
of sliding surfaces for friction and radiation conditions, and tests of
subsystem operation under simulated conditions can be accomplished
using standard techniques.

Some tests of certain major systems of the module will require
installation of the system in a large vacuum vessel. Examples of such
tests are (1) shock-absorber seals for friction and fatigue, (2) ablation-
oil distribution, and (3) pulse-unit transfer from storage to delivery
tube, firing from delivery tube, trajectory accuracy, and "detonation"
timing accuracy. Existing space-simulation chambers with special
modifications should fulfill the facility requirements for these tests.

7.3.4. Single HE Pulse Tests

The internal arrangement of the pulse unit for the nuclear
pulse propulsion system is such that the inert propellant essentially
shields the vehicle from all of the nuclear radiation and all of the fission

products of the nuclear explosion. Most of this inert propellant is inter-
cepted by the pusher plate of the propulsion system. The momentum
of the propellant is transferred to the pusher when the propellant
stagnates against the pusher. The duration of the propellant stagnation
is on the order of a 100 μsec, during which time the propellant tem-
perature rises to ~100, 000°K and then rapidly cools. The structural
material of the pusher (a medium-strength steel) is insulated from this
short burst of temperature by a layer of oil. Calculations* and tests†
show that a few thousandths of an inch of oil will keep the metal of the
pusher at temperatures below 600°F, even after a long powered flight
of a few thousand pulses. The mechanical subsystem of the module,
being isolated from the nuclear and thermal effects of its energy source,
responds to the impulse in a purely mechanical way.

A simulation of this mechanical response is necessary for the
development of the pusher plate and shock-absorber designs and for the
eventual fatigue tests of the assembled propulsion module. A technique
for simulating the nuclear-driven propellant-pusher interaction impulse
(pressure as a function of time) has been developed. For example, the
technique will simulate a pressure distribution which has approximately
100, 000 psi maximum at the center of the pusher and diminishes radially
to the edge of the pusher. The pressure-pulse simulation technique
employs sheet HE separated from the pusher plate by a layer of pressure-
attenuating material of the required thickness and density. The HE pulse
unit for single-pulse tests can be assembled on the pusher plate of an
inverted propulsion module or portion(s) thereof and subsequently
detonated by an electric blasting cap to produce a single-impulse load.

7.3.4.1. <u>Tests at Existing Facilities</u>. Many tests have been
made on both small-scale and small-diameter plates and shock-absorber
structures. Results from these tests indicate that there are practical
solutions to the many unique problems connected with the pusher-plate-
shock-absorber mechanism proposed for nuclear-pulse propulsion.**
Existing facilities for testing parts up to 1.5-min diameter are a cable
stand and a shock-absorber test stand.

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*Technical Summary Report, Nuclear Pulse Propulsion Project,
Vol. II—Interaction Effects (U), RTD-TDR-63-3006, Sections 2 and 3,
Air Force Contract AF29(601)-2207. (S/RD)
† Ibid., Sections 5 and 6.
**Technical Summary Report, Nuclear Pulse Propulsion Project,
Vol. IV—Experimental Structural Response (U), RTD-TDR-63-3006,
Sec. 2, Air Force Contract AF29(601)-2207. (S)

The experimental study of full-thickness portions of some parts of the pusher plate and scaled plates weighing up to 400 lb can be undertaken at the cable stand. These tests are necessary and adequate to prove the structural integrity of materials and shapes, especially of attachments, to the extent that a sound decision can be made for the construction of large-scale test pusher plates. However, early tests of a full-size (10-m-diam) pusher plate would be desirable for the most economical development of the propulsion system because some scaling questions need to be answered, and the pusher design needs to be proven before the entire set of shock absorbers are assembled for integrated tests. The shock absorbers, on the other hand, appear to be more readily scalable than is the pusher plate.

7.3.4.2. <u>Four-meter Test Stand</u>. The second-stage shock-absorber system for the 10-m module consists of a circular array of six identical, rigid-walled, double-acting shock absorbers. It appears feasible and most desirable economically to do the dynamic development on a single full-scale shock-absorber unit. This can be accomplished by equipping the full-scale unit with a suitable first-stage shock absorber, pusher plate, and the other necessary hardware and subjecting the assembly to high-explosive impulses. A pusher diameter of approximately 4 m would be compatible with the energy-storage capability of a full-size second-stage shock absorber. A test stand suitable for the shock-absorber development would be similar to the existing 1-m test stand. A sketch of such a test setup drawn approximately to scale is shown in Fig. 7.6. In addition to testing a single full-scale second-stage shock absorber, this 4-m setup will test the center section of a full-size 10-m first-stage shock absorber. It is also possible to scale or section the pusher plate in various ways so that valuable information on pusher design may be gathered concurrently with the shock-absorber data.

7.3.4.3. <u>Ten-meter Test Stand</u>. More than one test setup will be required to obtain the data necessary to prove the design and fabrication of the two shock-absorber systems and the pusher. The 4-m test-stand arrangement illustrated in Fig. 7.6 will be especially useful for shock-absorber development. The pusher, on the other hand, should be proven in full size before subjecting the very costly shock-absorber assembly to full dynamic tests because a pusher failure at that time would surely result in very serious damage to the shock absorbers. The pusher should therefore be well developed and proven on a special test setup before the integrated testing of pusher and shock absorber. Furthermore, the development problems of the two systems are to a certain extent separate. The development of the pusher will be carried out on a 10-m-diam test setup for which a special dissipative shock-absorbing system will be designed to simplify the test setup and the test shock

absorber. A sketch of the test setup for the 10-m-diam pusher is given
in Fig. 7.7. The shock absorbers will be developed on the 4-m test
stand and the pusher on the 10-m test stand, and both will be tested with
single HE pulses.

SHEET HE PULSE UNIT
PUSHER PLATE
FIRST-STAGE SHOCK ABSORBER
(FULL CROSS-SECTION TOROIDS
FOR IO-M MODULE)
SECOND-STAGE SHOCK
ABSORBER (ONE FULL-SIZE
UNIT FOR IO-M MODULE)
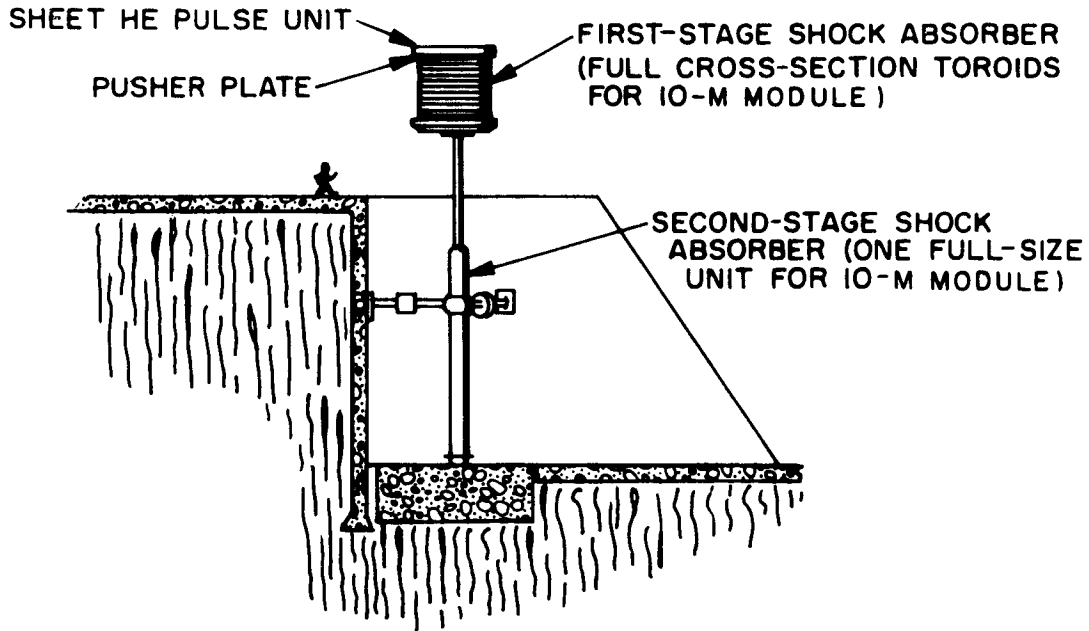
Fig. 7.6--4-m-diam single-pulse HE test facility

SHEET HE UNIT
PUSHER
FIRST-STAGE
SHOCK ABSORBER
SECOND-STAGE
SHOCK ABSORBER
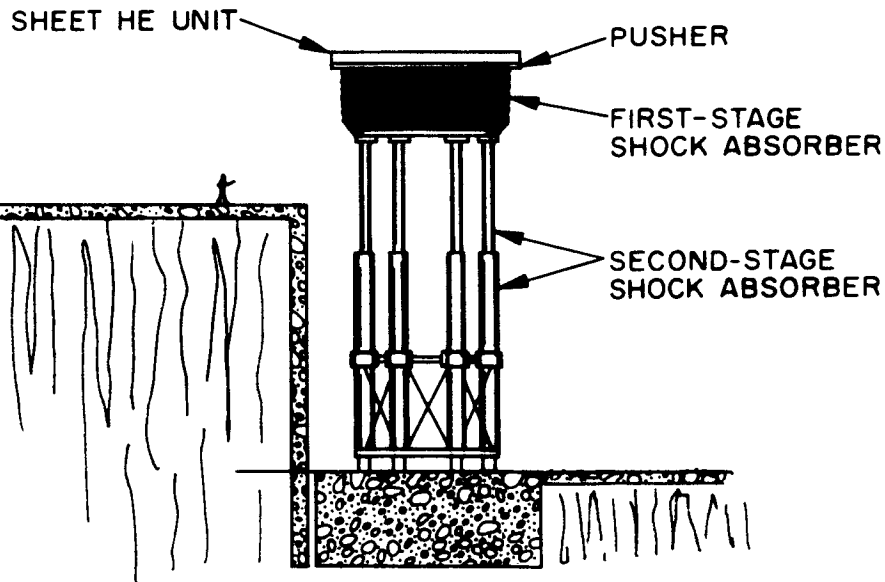
Fig. 7.7--10-m-diam single-pulse HE test facility

130

After the pusher has been tested extensively with single pulses and the single shock-absorber unit tested with single pulses and then with repetitive pulses, a complete assembly of shock absorbers and pusher will be tested, first with single pulses and then with repeated pulses on a repetitive 10-m test stand.

7.3.5. Repetitive-HE-Pulse Tests

The impulse expected from the nuclear pulse unit should be reproducible with HE on a full-size module at the design operating frequency while the vehicle is secured to a test stand. The value of this technique, which can play a strong role in proving the design in preliminary flight rating and in preflight ground qualification, cannot be overemphasized. This testing technique has become practicable because of several developments, such as using sheet HE and pressure attenuators for impulse simulation[*] and chemically boosting the modules into orbit. Preliminary estimates indicate that delivery and detonation of HE pulse units for the 10-m module can be accomplished at frequencies of approximately one per second. Sufficient HE pulse units (~100) can be stored in a magazine-silo for subjecting a module to a sufficient burst of pulses to bring each critical working component to its operating temperature equilibrium. By reloading the magazine-silo and repeating such bursts, a development specimen can be effectively life tested or an operational vehicle can be ground qualified before delivery to orbit. This repetitive-HE-pulse test installation, when fully developed, will be invaluable for component life testing, module fatigue testing, and vehicle preliminary flight-rating and ground qualification.

The repetitive-HE-pulse installation is illustrated in Fig. 7.8. The installation will be used for testing the vehicle in an inverted position for two basic reasons: vehicle support is simplified and, more important, the arrangement provides free expansion of the pulse-unit explosion products and strong shock waves.

The development plan envisions two sizes of repetitive-HE-pulse test installations: a 4-m and a 10-m test facility. The smaller-size facility can be developed economically and will be useful for inexpensive development work on individual full-size shock absorbers.

The size of vehicles for which this technique becomes impractical is not established. Certainly, such modifications as dual storage

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[*]Technical Summary Report, Nuclear Pulse Propulsion Project, Vol. IV, Experimental Structural Response (U), Report RTD-TDR-63-3006, Sec. 2, Air Force Contract AF29(601)-2207. (S)

131

and dual transfer of half-sectioned HE pulse units for testing a 20-m-diam module at approximately the same frequency as that of a 10-m vehicle could be made. Experience gained from the 4-m and the 10-m installations should provide the necessary test data to allow more accurate determination of size limits.
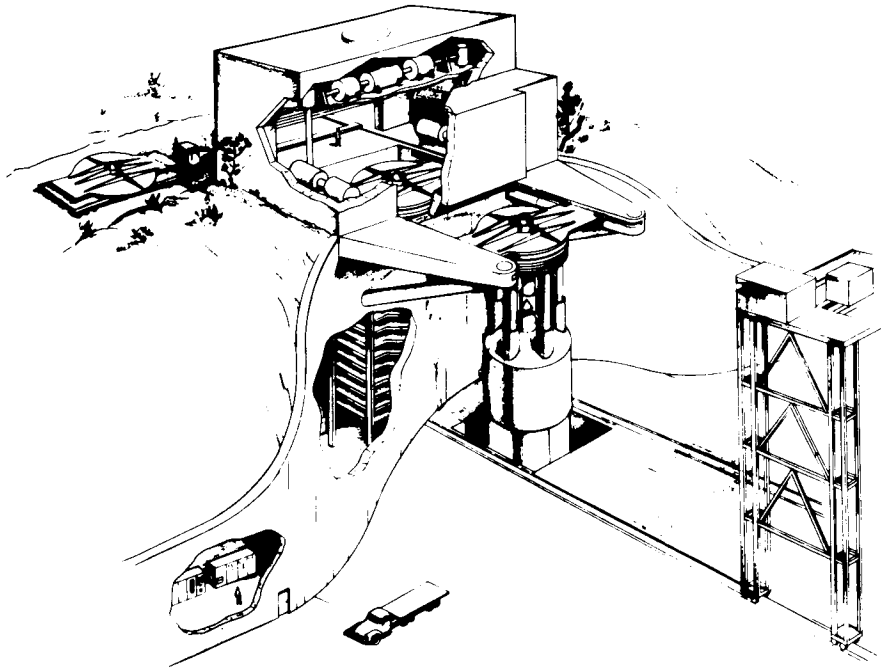
Fig. 7.8--Repetitive-HE-pulse test facility with inverted 10-m module in test position

7.3.6. Nuclear Underground Tests

7.3.6.1. Pulse-unit Development. A substantial portion of the expended effort of Predevelopment Phase I (see Fig. 7.1) has been applied to pulse-unit problems. Theoretical and analytical studies using and expanding the calculational techniques developed for weapon design are providing an understanding of the characteristics of various designs of nuclear-energy-driven "propellants." There is considerable promise that the pulse unit can be constructed in a desirable form with a minimum number of iterations. The development plan calls for two tests of pulse units early in the third year of Predevelopment Phase II. A third test is scheduled for the first year of Development Phase I. It is believed that information on calculational techniques received from the first two tests and supplemented with calculations of the expansion of other types of axisymmetric explosions will be sufficient to prove the practicality of producing a good design that can be tested in the third

scheduled test. The anticipated configuration of the pulse-unit test facility is shown in Fig. 7.9, where the pulse-unit propellant is shown expanding into a vacuum and interacting with an instrumented target plate. Essentially all of the yield energy of the nuclear device expands into the air-filled cavity, the requirement being that the energy density in the cavity be low enough to prevent reradiated energy from perturbing the state of the inert propellant on its flight to the target plate.
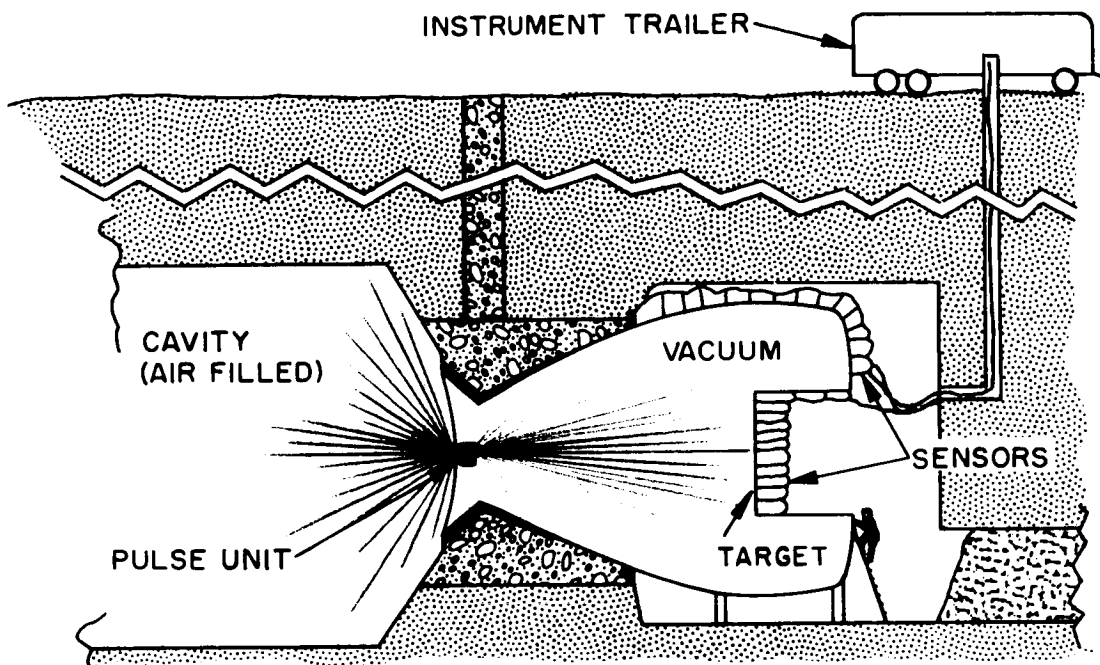
Fig. 7.9--Nuclear underground tests for pulse-unit development

After the pulse-unit expansion tests are satisfactorily completed and the design established, tolerance of device yield must be established. To accomplish this, a series of approximately 100 yield tests is planned. It is anticipated that these tests will pose minimum complication; they will be fired in well-shaft type of holes on a production-type schedule. These yield tolerance tests are scheduled for the second and third years of Development Phase I.

7.3.6.2. Interaction Studies. Experimental studies can be made of the interaction of the propellant and the pusher plate, in particular, the ablation processes. The test-plate sizes can range from a few feet in diameter to the full propulsion-module size. The propellant source planned for these tests is a low-yield nuclear

source—the LENS system, which is a very-low-yield "gun-type"
plutonium assembly (see Fig. 7.10). One or more "barrels" are
inserted into a reflector in such a way that the explosion will vent
through the barrel bores by expansion of the plutonium core. The
barrels consist of dense metal (uranium or tungsten) to retard the radial
expansion away from the barrel axis. By adjusting the barrel lengths,
the bore diameters, the yield of the core, and the distance to the target
plate along the barrel axis, a wide variety of propellant-pusher inter-
action states can be attained. With the low yield and the control of
momentum and energy of the LENS system, the recovery of exposed
target plates will be relatively certain, and with absorption of most of
the energy trapped in a vacuum tank, the recovered plates exposed
surface should show the true expected ablation and the effectiveness of
ablation-protection coatings. Small-scale tests (less than one-eighth
of a 10-m pusher plate) could be made on a noninterference basis as an
adjunct to other LENS tests by extracting as little as 1 percent of the
energy of the source. One-fourth scale and larger pusher plates would
use LENS designs specific to these test sizes.
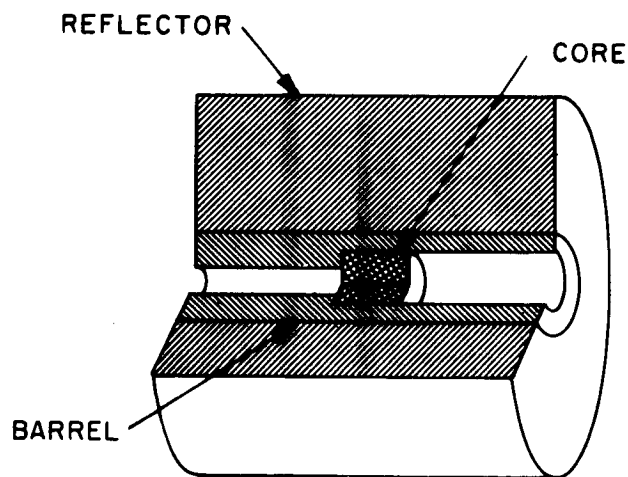
Fig. 7.10--Low-energy (yield) nuclear source (LENS)
for nuclear underground tests

A test facility for interaction experiments with a test plate
one-fourth the diameter of the 10-m pusher is illustrated in Fig. 7.11.
Directing most of the very-low-yield energy down the air-filled tunnel
will permit shallow burial of the test facility which, in turn, will allow
short instrument lines. It is estimated that containment will be kept

within and at the source end of the vacuum tank, thus making recovery of test samples favorable. The use of trailers for housing recording instruments will keep the cost of this phase of the tests to a minimum. Metal fins, properly situated and oriented, will be used to absorb the low-energy x rays (<1 ev) emitted from the propellant that stagnates against the target and the wall of the vacuum tank. This energy absorption rapidly lowers the temperature of the test-plate environment in simulation of space conditions. Consequently, the only thermal conditions that could affect the test-plate surface will occur during the initial propellant-pusher interaction, as would occur in a single pulse-unit operation in space. Therefore, the recovered test plate can be analyzed after the interaction for ablation effects resulting from a true source simulation of a single-pulse loading in the nuclear pulse propulsion system.
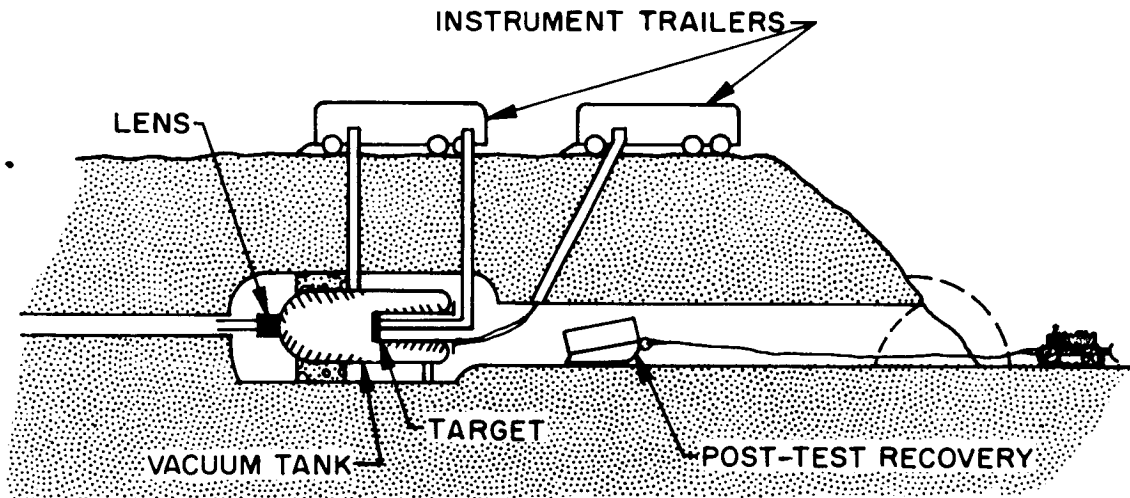
Fig. 7.11--Test facility for nuclear-underground-tests interaction studies

7.3.6.3. <u>Full-scale Module Interaction</u>. Subjecting a full-size nuclear pulse propulsion module to a contained underground nuclear pulse appears feasible with the LENS system. By using the LENS system, the test setup will not have to be buried very deeply and recovery probability will be very high. This test setup is depicted in Fig. 7.12. Experience gained from the small-scale test-sample experiments should improve the recovery confidence.

The main purpose of the full-scale tests is to verify the practicality of treating the nuclear and mechanical effects of the nuclear pulse propulsion system separately. If the results from the previous

tests and analyses will generate sufficient confidence, then these full-scale, complete module tests can be bypassed. The proof of validity of nuclear and mechanical-effects separation cannot be accomplished by any other means except space tests, which at a later time in the development program will firmly establish the validity of treating these two effects separately.
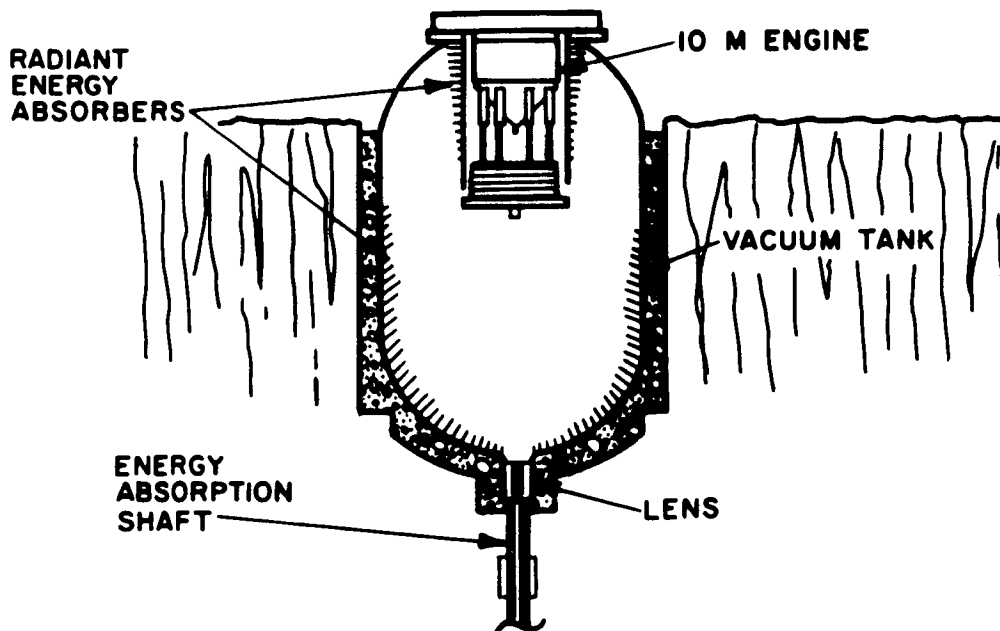
Fig. 7. 12--Nuclear underground tests full-scale interaction

7. 3. 7. Nuclear Ballistic Tests

The first space tests are scheduled after the propulsion module is thoroughly tested and qualified for space operation in the HE ground test facilities and by nuclear underground tests. Space tests will be made to obtain operational data with operational nuclear pulse units in actual space conditions. The first space tests are planned as a series of ballistic tests in which the propulsion module is lofted by a chemically fueled lofter to such an altitude that the module can be subjected to a series of nuclear pulses and recovered after the tests. To minimize or avoid contamination of the atmosphere, the tests will be performed above an altitude of 120 km (~400, 000 ft) and conducted in such a way that the fission products of the nuclear pulse units during the explosion expand away from the earth's surface. This means that the module should be lofted to an apogee which is determined by the distance the

module is traveling toward the earth's surface when propelled by the
selected number of pulse units for the test.

A typical trajectory of the ballistic tests is shown in Fig. 7.13.
After lofter burnout the lofter and test module will coast up to apogee
as a unit. At apogee this unit will be rotated so that the front end of
the module is pointing toward the earth, at which point the lofter and
module will be separated by a retrorocket on the lofter. After separation
to a distance greater than 76 ft (the stand-off distance of the pulse unit
from the pusher), the pulse units required for this test are sequentially
ejected and fired. The recovery system which will control the attitude
and the sink speed of the module is then activated. Flotation gear will
be employed to recover the propulsion modules from the water for
detailed inspection.

With regard to the radioactivity of the nuclear-pulse module
after the test, preliminary studies indicate that the amount of radiation
due to the explosion and discharge of nuclear debris on the module will
be very low ($\sim$20 mr/hr after 50 pulses) and that almost unlimited
access of personnel would be permitted to the module shortly after
module recovery. Therefore, the recovery from the water appears to
be a straightforward mechanical operation. A surface vessel equipped
with a crane with a lift capacity of 100 tons and a 20-ft radius and with
the required deck cargo capacity or towed barge could handle the pickup
and return of the module.

7.3.8. Orbital Tests

The final qualification to obtain initial operational capability
(IOC) will be achieved by a series of orbital tests. Two basic assumptions
have been made for the orbital test program: (1) orbital rendezvous
and assembly in orbit are routine operations and (2) man-in-space is
well established.

Throughout all previously performed experiments and tests,
where every single component, all subassemblies, assemblies, sub-
systems, and complete systems have been thoroughly tested under
simulated environmental conditions and where in the ballistic tests the
entire propulsion system has been tested in real space environment,
an extremely high level of confidence in the operational reliability of
the propulsion module will have been established. The first obvious
objective of the orbital tests, therefore, is to manifest the proper
functioning of the entire system complex of the propulsion module under
prolonged operational conditions. A second objective is to gain a
maximum of operational experience with the module, including orbital
startups and maneuvers in orbit. Concurrently, a series of measure-
ments much like those made for the ballistic tests will be made to

ROTATION

SEPARATION

START PULSE-
SYSTEM OPERATION

END OF PULSE-
SYSTEM OPERATION

APPROX. 120 KM ALT,
REQUIRED FOR MINIMUM
CONTAMINATION OF
ATMOSPHERE

POWERED
FLIGHT

AIR RECOVERY

LAUNCH PAD

FLOTATION GEAR FOR
RECOVERY FROM WATER
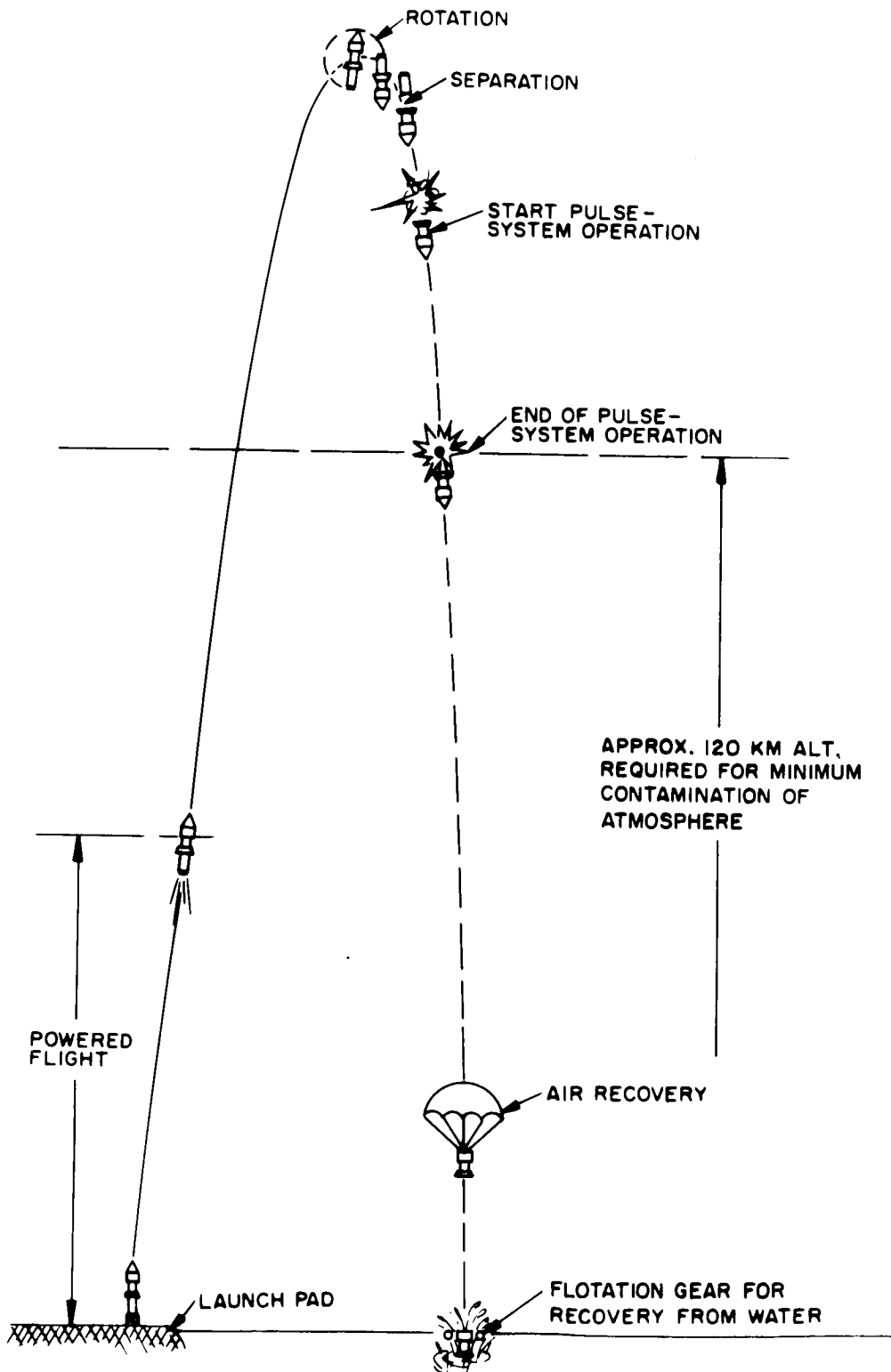
Fig. 7.13--Trajectory for nuclear ballistic tests

138

record temperature, pressure, and radiation profiles during operational cycles.

Each propulsion test module will be put into a low earth orbit by a two-stage Saturn V launch vehicle. The objectives of the orbital test program will require a test crew sufficient to conduct the tests and to monitor all other experiments scheduled for the test runs. Hence, a capsule (test-crew module) for the test crew, including a complete life support system, will have to be provided.

Because of the payload limitation of the Saturn V, the propulsion modules, test-crew modules, test personnel, test equipment, and the nuclear pulse units will have to be launched separately and assembled in orbit (see Fig. 7. 14).
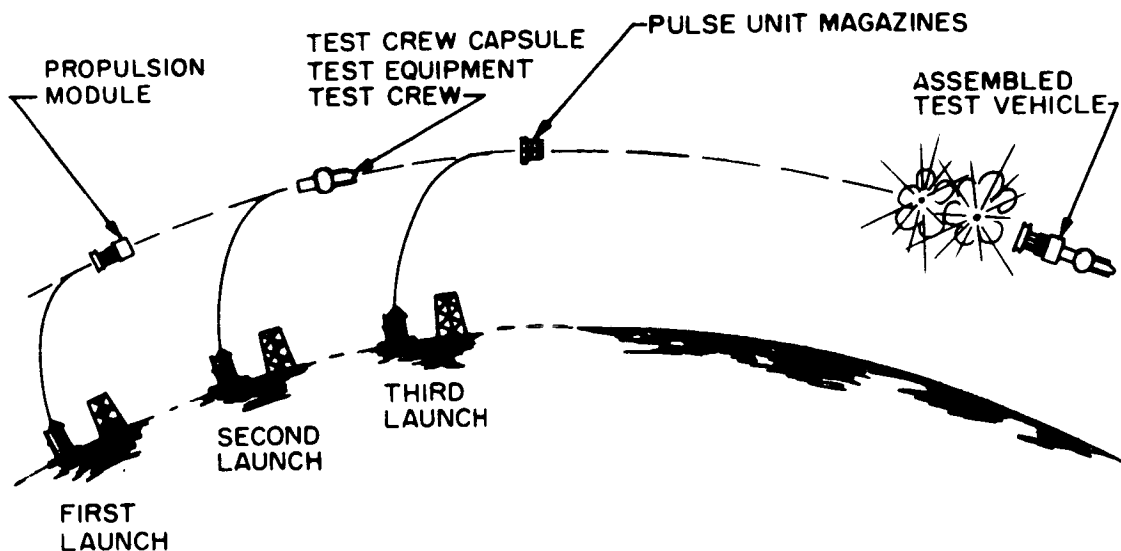
Fig. 7.14--Orbital tests

The launch schedule for this operation, as given in Table 7. 1, is based on the following considerations: With a reliability factor of 0. 75 for Saturn V orbital launches, four test vehicles (propulsion module plus test-crew module for each) are required for a successful launch of three test vehicles; i. e. , three of the vehicles will be launched into orbit and one will be kept as a spare. Six Saturn V's are required for this operation. Each test propulsion module will be equipped in orbit with 800 nuclear pulse units (this quantity should be sufficient to conduct orbital tests involving a few to a few hundred pulses); this

139

operation will require three Saturn V's. In order to perform the final
IOC tests, for which several thousand nuclear pulses are necessary,
five additional Saturn V launches are programmed. These five additional
Saturn V's can carry a total of 4,000 nuclear pulse units. In the event
of a launch failure, any of the remaining Saturn V's can be used as a
spare, still leaving sufficient pulse units for the IOC tests by orbital
transfer of pulse units to the vehicle selected for the IOC tests. There-
fore, a total of fourteen Saturn V's will be required to set up and equip
the orbital test program. However, depending on the allowed stay time
of the test personnel in orbit, additional launches may be required for
personnel exchanges at certain defined time intervals.

Table 7.1

SATURN V LAUNCH SCHEDULE FOR CONDUCTING IOC TESTS

Launch No.	Item to be Launched
1	Propulsion module No. 9
2	Test-crew module, personnel, test equipment
3	Nuclear pulse units (800)
4	Propulsion module No. 10
5	Test-crew module, personnel, test equipment
6	Nuclear pulse units (800)
7	Propulsion module No. 11
8	Test-crew module, personnel, test equipment
9	Nuclear pulse units (800)
10-14	{ Propulsion module No. 13, 14 } Spares { Test-crew module } { Nuclear pulse units (4,000) }

7.4. DEVELOPMENT AND QUALIFICATION SCHEDULE

A schedule for the efficient execution of the development and
qualification of the 10-m module is shown in a condensed form in
Fig. 7.15 and in more detail in Fig. 7.16. The schedule is divided
into four phases: Predevelopment Phases I and II and Development
Phases I and II. As indicated in Figs. 7.15 and 7.16, each phase
requires three years. At the end of the qualification program, which
is also a three-year period, it is expected that the initial operational
capability (IOC) will be obtained.
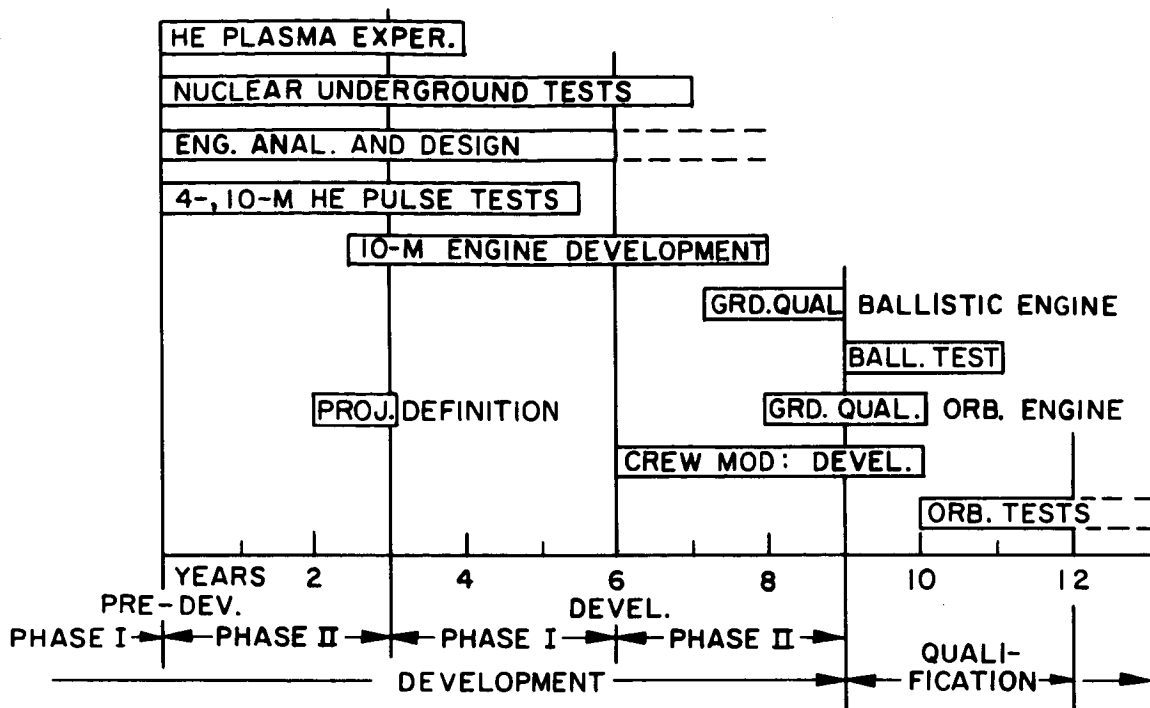
Fig. 7.15--Condensed development schedule

7.4.1. Predevelopment Phase I

All work up to present in the fields of analytical analysis, engineering design, and experimental testing, including a considerable amount of test development, response testing of small-diameter test plates, and HE plasma/target interaction tests, is included in this Predevelopment Phase I.

7.4.2. Predevelopment Phase II

Response testing of small-diameter plates and scaled primary shock absorbers will be continued as will the HE plasma—target interaction tests, and information derived from these tests will be utilized for the development and design of the 4-m and 10-m single-pulse test specimens and for the design and construction of their respective test facilities. In addition to the tests on small-diameter (6- to 12-in.-diam) targets, pulse tests on 1-m-diam targets will be made for the development of primary and secondary shock-absorber systems. In the latter part of the first year of Predevelopment Phase II, the first nuclear underground tests are scheduled (Milestone 1 of the program).
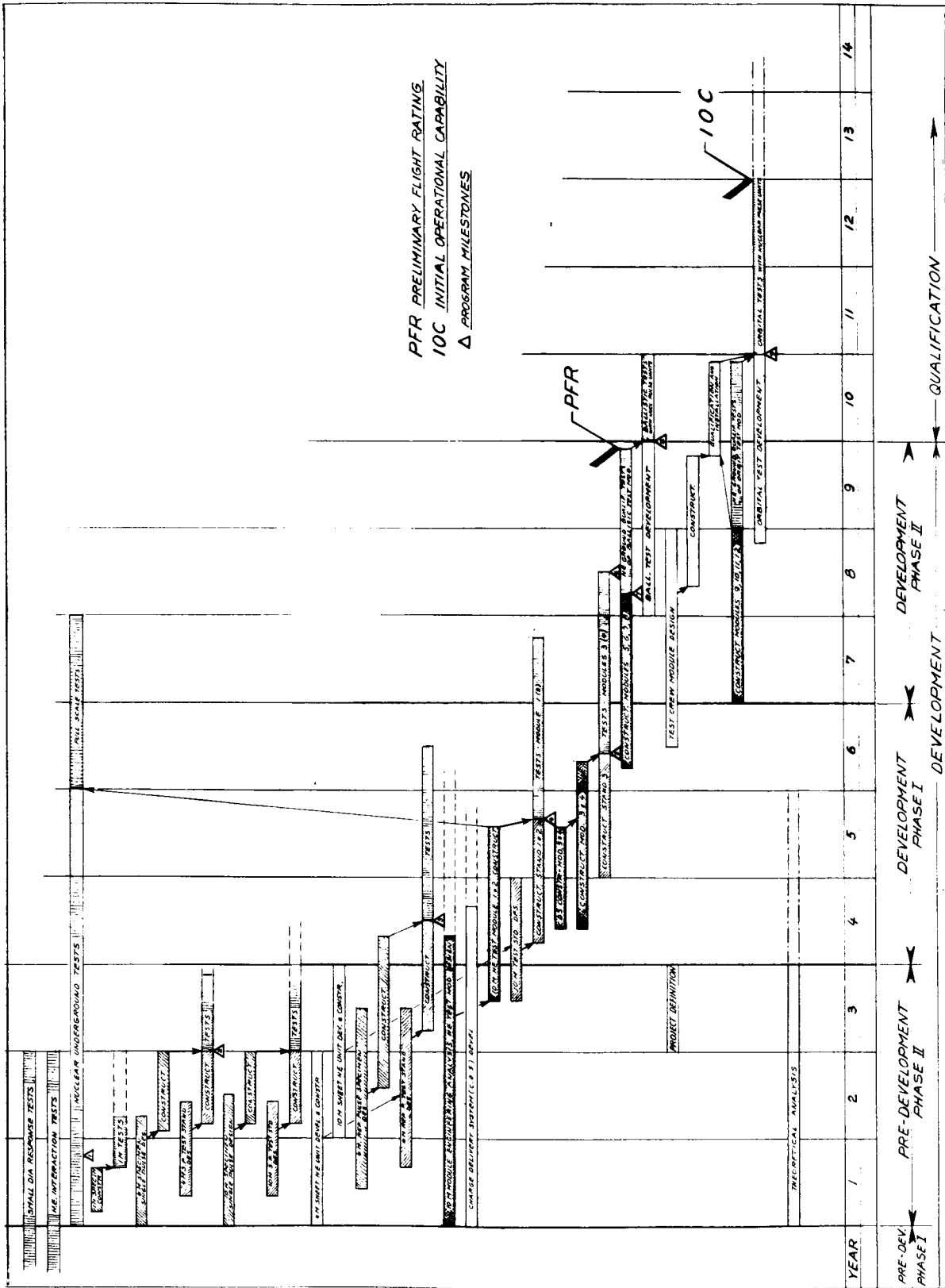
Fig. 7.16--Detailed development schedule

Extensive single-pulse tests will be performed with the 4-m
and 10-m test specimens (Milestone 2) to develop full-scale attachments
to the pusher plate and intermediate platform and to further develop
both the primary and the secondary shock-absorber systems. During
the time these tests are being carried out, and probably influenced by
their results, one of the major tools for the development program—the
HE repetitive-pulse test facility—will be developed. The first of these
facilities will be a test stand for the 4-m test specimen. The specimen
will be developed parallel with the test stand and the sheet HE propellant
units necessary for the tests. The design and construction of the facility
and test specimen will take the major part of Predevelopment Phase II;
however, construction of both will not be completed.

Also during this phase, an engineering analysis of a 10-m
prototype module and the design and development of the module and of
the charge (nuclear pulse unit) delivery system (CDS) will be brought
to such a stage that the construction of two 10-m test modules can be
started at the end of this phase.

The major portion of the nuclear underground tests will also be
performed during this Phase II. A total of eleven tests is scheduled
for this period, which will include two pulse unit tests, three interaction
research tests and three interaction design tests. The results of these
tests will be incorporated in the design of the test specimen and in the
design of the 10-m prototype modules.

During this entire 3-yr period, space-simulated tests of com-
ponents, subassemblies, and assemblies of the propulsion modules will
be performed in space-environment chambers. It is anticipated that
wherever possible existing facilities will be utilized and modified as
necessary to suit the test conditions.

Theoretical analyses of pulse systems and interaction problems
are to be performed throughout the Predevelopment Phase II and, if
required, will be continued into the development phase.

In Predevelopment Phases I and II, most of the basic tests
will have been performed to establish the feasibility of the system and
to build up a high level of confidence in its development. Therefore,
preceding Development Phase I and while other development tasks are
being performed, a full year's time is allotted for a Project Definition
Phase (PDP).

7. 4. 3.   Development Phase I

During the first year of Development Phase I, the 4-m repetitive-
pulse tests will start (Milestone 3). The first two 10-m test modules

and the two 10-m HE repetitive-pulse test facilities will be completed
in the second year of this phase. After the necessary trial runs of the
test stands, the testing of the 10-m modules will start (Milestone 4).
The first two test modules will be tested without a charge delivery
system. The construction of the CDS system, however, will be started
during the first year and will be installed into the third and fourth test
modules, which will be completed during the third year of this phase.
A third test stand will be constructed for tests of Module No. 3, which
will be the first complete propulsion module, and tests can start in the
first half of the third year (Milestone 5).

During the first year of this phase, the third pulse-unit
expansion test and the last two interaction—design tests will be performed.

In order to establish a reliable tolerance level of the yield of
the nuclear devices for the pulse units, a series of approximately 100
nuclear underground tests is scheduled for the second and third year
of this phase. It is anticipated that each of these devices will be fired
in vertically bored well shafts.

During the third year, two large-scale underground nuclear
interaction tests with a full-size propulsion module (Module No. 2) are
scheduled. These are to confirm the findings of the theoretical analyses
and computations and may have decisive influence on the construction
of the following series of prototype modules (Modules 5 to 12), which
is scheduled to start the last part of the third year.

The design of the test-crew module (powered flight station and
life support system) will also be started at the end of the third year.
For the design of this test crew module, it is assumed that by the time
the orbital tests have to be performed, sufficient experience with
manned space stations will be available so that no new developments
of life-support, environmental-control, and power-supply systems
will be required.

7.4.4. Development Phase II

The construction of test modules for ballistic and orbital tests,
the design and construction of the payload (test crew module), the
necessary developmental work for the ballistic tests, and the ground
qualification tests of the ballistic test modules are included in Develop-
ment Phase II. There will also be a large portion of the HE ground
qualification tests of the orbital modules accomplished, and at the end
of this phase the required developmental work for the orbital tests will
be well under way.

Three more milestones of the development program will have been reached in this phase: the completion of the HE ground tests of the 10-m modules (Milestone 6), the start of the HE ground qualification tests of the ballistic test modules (Milestone 7), and the completion of these tests, which will give the module a Preliminary Flight Rating (PFR) for conducting ballistic tests.

7. 4. 5. Qualification

The beginning of the qualification phase of the program is marked by Milestone 8, the start of the ballistic tests. These tests will be the first space tests of a complete propulsion module using nuclear pulse units. A series of three or more ballistic tests is contemplated, starting with approximately ten pulses and increasing the number of nuclear pulses for each subsequent test. The number of launches will depend on the amount and quality of information obtained from the first tests. The allotted time for the ballistic tests is one year; however, this time can be extended if more tests are necessary.

The HE ground qualification tests of the orbital propulsion modules, the preparation for the orbital tests, and the qualification of the crew capsule will be finished at the end of the first year of this phase. The start of the orbital tests is scheduled for the beginning of the second year of this phase (Milestone 9). During the second and third years, the orbital testing of the modules will be performed and it is expected that the initial operational capability (IOC) will be obtained at the end of the twelfth year of the program. For the orbital test program, three modules are intended to be placed into a low earth orbit by Saturn V boosters.

7. 5. DEVELOPMENT COST ESTIMATE

A development cost summary of all major development and test programs contemplated for the development of the nuclear pulse propulsion module is given in Table 7.2. The costs are given for two major items: (1) test hardware, which includes theoretical and engineering analyses and all developmental work and preparation required for the tests, and (2) testing, which includes the sheet HE units for the tests performed with high explosives, the nuclear devices for the nuclear underground tests, and the nuclear pulse systems for the ballistic and orbital tests. The testing costs also include the costs for engine qualification for the ballistic and orbital tests.

The total cumulative cost of the program is shown graphically in Fig. 7.17. It was assumed for this curve that approximately $10

145

Table 7.2

DEVELOPMENT-COST SUMMARY
($ Millions)

Item	Test Hardware	Testing
Small-diameter-target response tests		0.160
HE interaction tests		1.730
IBEX experiments	1.200	0.800
Underground nuclear tests	0.080	57.700
Nuclear devices	---	12,750
HE pulse tests		
Single pulse		
1-m diam	---	0.890
4-m diam	8.245	0.300
10-m diam	9.125	1.200
Sheet HE development		
4-m diam	0.100	---
10-m diam	0.100	---
Repetitive pulses		
4-m diam	8.191	4.600
10-m diam (Modules 1, 2, 3, 4)	65.690	42.600
Ballistic-engine qualification		7.600
Orbital engine qualification		7.600
Ballistics tests (remote site)	70.100	106.000
4 engines (Modules 5, 6, 7, 8)	52.000	---
Orbital tests to IOC (incl. 12 Saturn V launches plus 2 Saturn V spares)	25.900	905.600
4 engines (Modules 9, 10, 11, 12)	52.140	---
Space-simulated tests		14.120
Test-crew-module development	29.800	---
Test crew for orbital tests		2.000
Theoretical analysis	2.000	---
Engineering analysis	0.455	---
Totals	325.126	1,165.650
Total cost of test hardware		325.126
Total cost of testing		1,165.650
TOTAL COST OF PROGRAM		1,490.776

million will have been spent for Predevelopment Phase I. The curve is marked with major milestones. The end point of the development program is the achievement of the initial operational capability (IOC). On the left side of the diagram the total amount of expenditure for each phase of the program is shown in rounded-off numbers. The different phases of the program would therefore require the following funding:

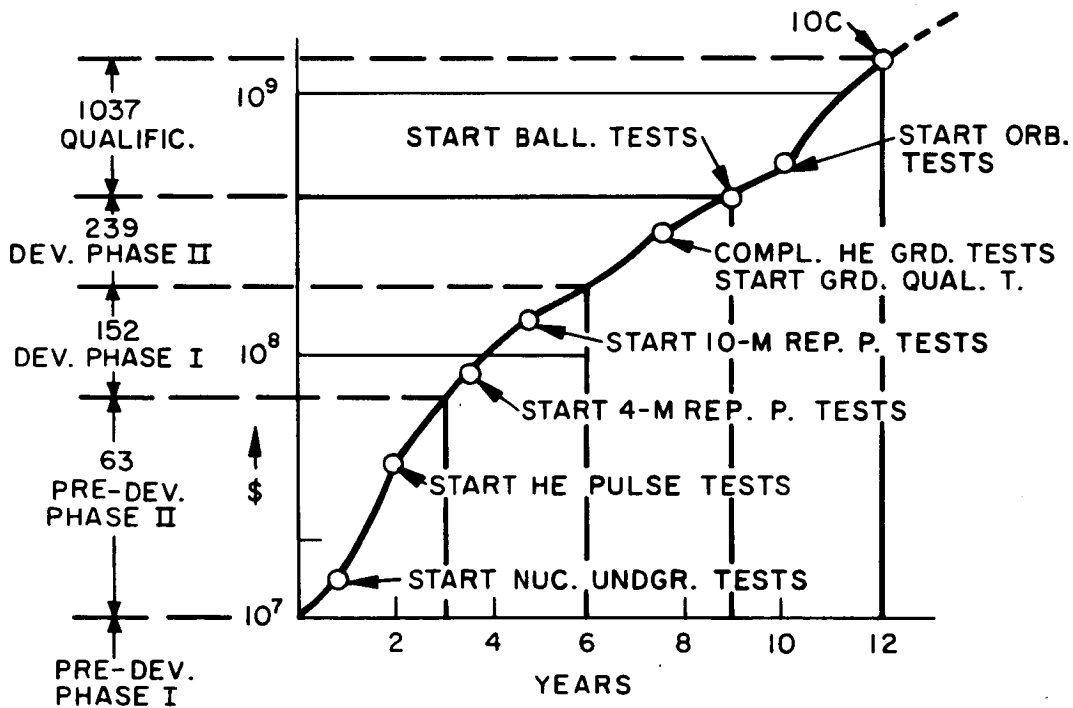|	$ Millions
Predevelopment Phase II	63
Development Phase I	152
Development Phase II	239
Qualification.	1,037
Total	1,491

Fig. 7.17--Cumulative cost summary

8. SYSTEM-COSTING CRITERIA

System-cost indications resulting from this study were reported in Vol. II, as were the system-cost sensitivities to changes in the more important performance and cost parameters. Some of the costing criteria were discussed in Vol. II in explaining the over-all direct operating costs (DOC), but additional details and background for the estimates are presented in this section. The three major components that comprised the total direct operating costs are discussed here: the vehicle hardware and mission-required payload, the nuclear pulse propellant, and the earth-launch vehicles (or the lofters) and their operation.

8.1. VEHICLE HARDWARE AND MISSION-REQUIRED PAYLOAD

For exploration missions, the payload is costed along with the propulsion module and other hardware since it is a (roughly) definable part of the mission requirements and typically acounts for a significant portion of the total DOC. Exploration-mission payloads are also difficult to separate from the vehicle hardware since the operational payload has been defined to include that portion of the vehicle structure that houses and provides for the mission personnel and mission operations.

A major and vital portion of the vehicle hardware is, of course, the propulsion module. It is considered first.

8.1.1. Propulsion-module Costs

The propulsion-module costs used in this study were taken from previous cost studies for which bigger propulsion modules were of major interest. A 33-m-diam module with a thrust of 44 meganewtons (MN) was a typical size of interest rather than the 10-m, 3.5-MN module prominent in this study. For those relatively massive propulsion modules, typical mission economics indicated that the cost per kilogram of the structure would remain fairly low (i.e., the system would not benefit by paying a much higher procurement cost to gain modest reductions in inert weight or modest improvements in performance). The cost per kilogram for all size modules, therefore, was predicated on relatively simple and rugged structural designs (a characteristic requirement for nuclear-pulse propulsion) and on the assumption that costly refinements or weight-saving measures would be avoided.

148

In this study two factors work to somewhat alter the above economic indications. First, the mission components, including the propulsion module and its propellant, are typically carried to orbit by a chemical ELV prior to employing nuclear-pulse propulsion. Second, the nuclear-pulse propellant is considerably more expensive per kilogram for the smaller 10-m vehicles than for larger ones, as will be discussed later. Therefore, as reported in the cost and sensitivity sections of Vol. II, the system would benefit by paying significantly more for the propulsion module, if so doing resulted in modest weight or performance benefits. Because of the basic characteristics of the propulsion system and the mass of even the small propulsion modules, however, any trend to higher cost per kilogram will stop far short of that paid for today's relatively small, delicate, and complex space-propulsion hardware.

The over-all cost per kilogram for propulsion modules, as used for this study, is shown as a function of the module dry weight in Fig. 8.1. The curve labeled "operational fleet" was used for the broad-range parametric data of the early study period, whereas the "first operational modules" curve was used for the specific conceptual designs of the later study phase. The over-all dry module unit cost was $132/kg for the 10-m module (dry weight ~92,000 kg) and $101/kg for the 20-m module (dry weight ~380,000 kg).

The cost data of Fig. 8.1 are based on a few point design studies, with the curve slopes (at least for the lower weights) established by comparison with various aerospace and industrial hardware items which are manufactured in a range of sizes. Dollar values for 1963 were used with no escalation or inflation factor.

The propulsion-module costs per kilogram are seen to be lower than the large aerospace vehicle average by a factor of perhaps two or so. (For example, the "first operational" curve if applied to a 127,000-kg Saturn S-1C stage would predict about $126/kg, or a procurement cost 16×10^6, whereas the actual expected cost is about 30×10^6.) One reason for the lower cost is that a large portion of the nuclear-pulse module's mass is concentrated in a few large conponents, the remaining mass being dense and compact due to the dense nature of its propellant. The steel pusher stands out as a single massive item (comprising some 6/10 of the entire propulsion-module mass) and it alone accounts for a considerable reduction in the module's average cost per kilogram, as shown in Fig. 8.2. The pusher is treated separately; the remaining hardware cost per kilogram is shown to be more compatible with other large vehicles such as Saturn.

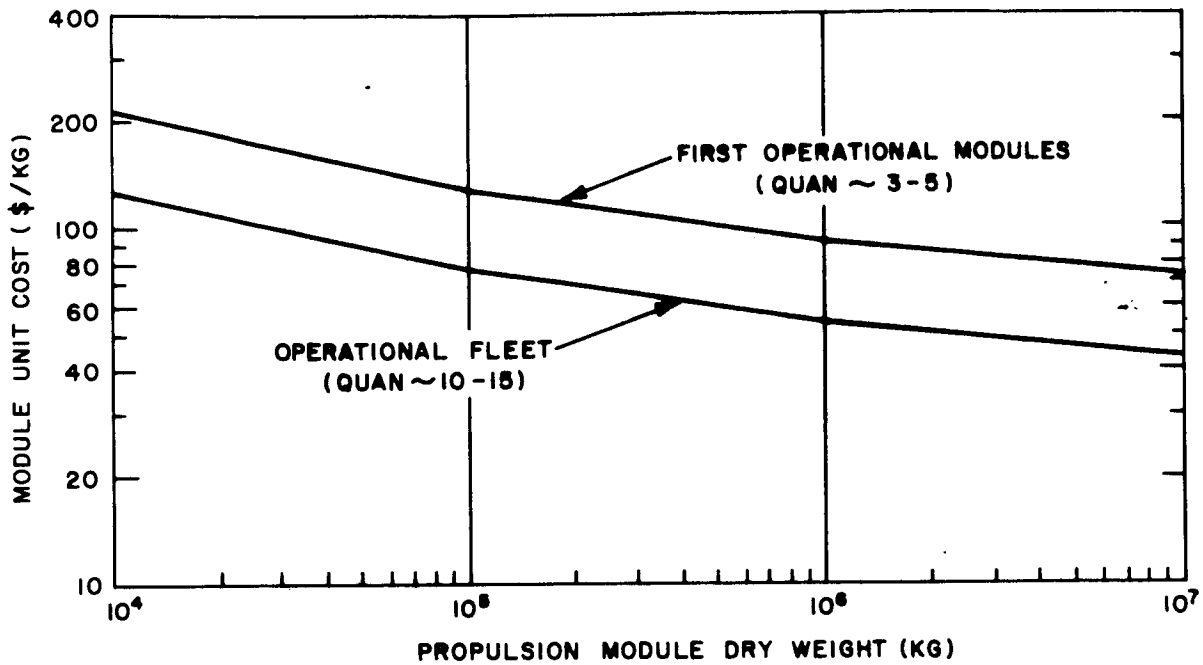Fig. 8.1--Estimated fabrication cost per kilogram
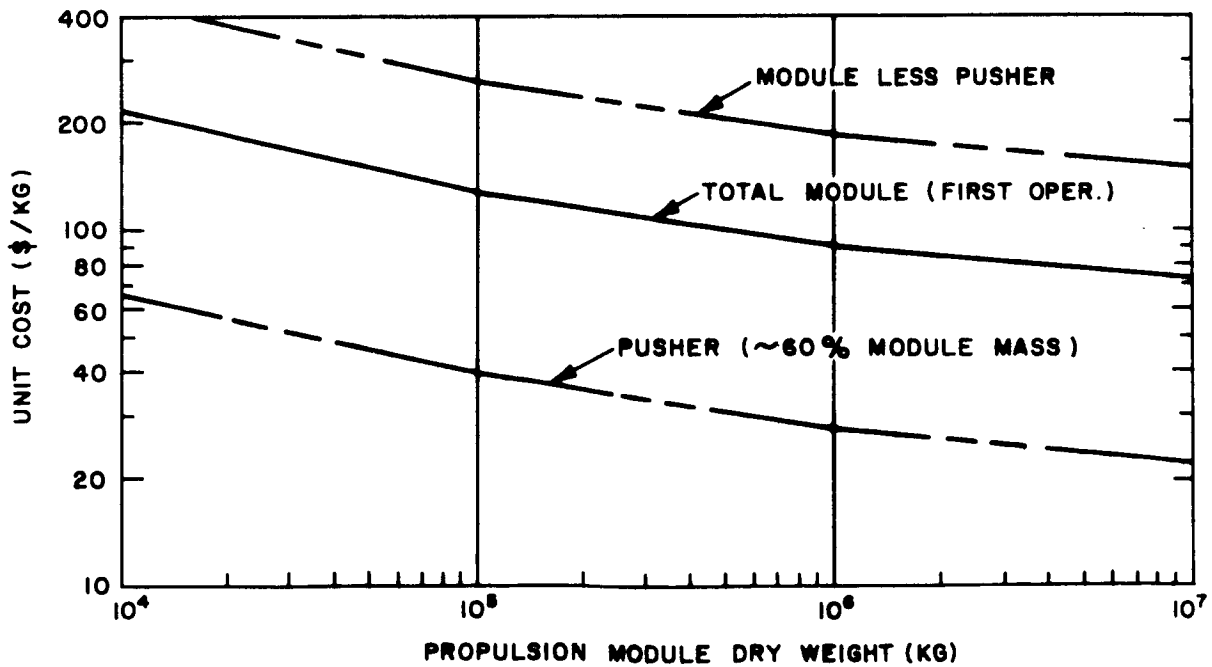for nuclear-pulse-propulsion modules

Fig. 8.2--Effect of pusher cost on estimated cost
per kilogram of complete propulsion module

150

8.1.2. Exploration-mission Payloads

The total payload for exploration missions (as discussed in Sec. 2 of Vol. II) is divided into three classifications: operational, in-transit, and destination payloads. The operational payload consists of personnel, their accommodations, furnishings, shielding, and other life-support items, plus items necessary to operate the vehicles. The operational payload is therefore largely dependent on the number of personnel and the mission duration and has consequently been fairly well defined. The in-transit and destination payloads are less understood at this time and are not well defined. However, the destination payloads, which for landing missions are large enough to be significant in total mission costs, were broken down for cost purposes by assuming a given payload mix (from payload data of Table 2.11, Vol. II).

The unit-cost values used to calculate the payload costs for this study are shown in Table 8.1, along with a brief rationale for their use. The resulting average unit cost for a 10-m vehicle's 8-man operational payload was $164/kg when no reentry vehicle was carried; $195/kg when an Apollo-velocity reentry vehicle was used; and $216/kg when a 50,000-fps reentry vehicle was used. A typical average unit cost for a 20-m vehicle operational payload was $140/kg with no reentry vehicle, the difference being largely due to a lower cost per kilogram for the life-support structure, equipment, and spares. (The 20-m vehicles normally were assumed to return to a circular earth orbit at mission completion, so reentry vehicles were not required.)

The costs of payloads for exploration missions accounted for some 5 percent to 10 percent of total DOC for the 10-m vehicles and to as much as 23 percent for the 20-m designs. Typically, the payload costs more than the remaining system hardware, exclusive of the earth-launch vehicles. The relation of payload costs to the other significant cost items is shown in the typical cost build-up data presented in Sec. 4 of Vol. II.

The costs of payloads for lunar logistic missions or lunar ferry missions were not computed. For such delivery or transportation systems, the cost data of primary interest are the costs per unit of payload delivered. Such data were presented in Sec. 4 of Vol. II, which also contains a tabulation of the basic unit-cost estimates for the components used in the complete lunar systems and hence these data are not repeated here.

Table 8.1

UNIT-COST ESTIMATES FOR EXPLORATION-MISSION PAYLOADS

Operational Payload Items

1. Life-support System

Structure (unit cost assumed similar to that of
military aircraft built in small quantities) $250/kg

Equipment not listed elsewhere

~26%	furnishings	$150/kg×0.26 = 39
~50%	main and emergency power supply	$400/kg×0.50 = 200
~ 5%	communications, navigation, autopilot, and guidance systems	$2,000/kg×0.05 = 100
~19%	control, spin system and abort chemical motors, tanks, and plumbing	$300/kg×0.19 = 57
100%		396 ≈ $400/kg

Structure for 20-m vehicles (generally same kinds of components
as for 10-m vehicles but larger or increased quantity; cost/kg
ratioed down in proportion to the respective propulsion module
unit costs: 101/132 × $250/kg) $191/kg

Equipment for 20-m vehicles (same basis as for structure:
101/132 × $400/kg) $306/kg

2. Radiation shielding (mostly borated polyethylene plus borated
steel or depleted uranium; high-weight, low-cost material,
therefore, much of the cost estimated is for fabrication) $50/kg

3. Ecological system and food

Hardware (main system including food storage and serving
equipment plus emergency and reserve equipment) $250/kg

Expendables (gases, water, food, filter materials, and
chemicals) $5/kg

4. Reentry vehicle and maneuver stage (Apollo-type reentry
vehicle plus approximately 1,000-fps maneuver
stage with storable propellant) $750/kg

5. Abort propellant $5/kg

6. Spin propellant $5/kg

7. Space-taxi capsules

Hardware only $600/kg

Including typical propellant allowance $300/kg

8. Spares and repair equipment (estimated at
approximately twice the cost of the
life-support-system structure or non-
pusher propulsion-module structure)

10-m vehicles $500/kg

20-m vehicles $380/kg

9. Checkout instrumentation $2,000/kg

In-transit Payload Allowance

(In-transit payload was assumed   a   mix of scientific/electronic
equipment and "guinea pig" experiments; modest total weight) $1,000/kg

Destination Payload Allowance

Nominal destination payload (modest total weights, mostly mapping
equipment plus data-handling and storage system) $2,000/kg

Landing missions; a typical payload mix was assumed to be

~ 1.3%	mapping equipment	$2,000/kg×0.013 = 26
~ 0.7%	data-handling and storage	$2,000/kg×0.007 = 14
~ 5.6%	environmental satellites	$5,000/kg×0.56 = 280
~ 8.5%	unmanned landers and propellant	$500/kg×0.085 = 42
~16.5%	unmanned returners and propellant	$300/kg×0.965 = 50
~67.4%	manned excursion vehicle and propellant	$500/kg×0.674 = 337
100.0%		749 ≈ $750/kg

Very large destination payloads (≥250,000 kg; a mix similar to the
above assumed, but with larger manned excursion vehicles
and more propellant; estimated 2/3 the above) $500/kg

152

8.1.3. Propellant Magazines and Miscellaneous Items

A remaining class of vehicle hardware, after considering the propulsion module and payloads, is the expendable propellant magazine. Still other components contributing to departure weight of the vehicle, but usually of insignificant cost, are guidance propellants and start-up fluids.

The propellant magazines are conceived as expendable containers for the pulse units (probably of aluminum construction). They contain a suitable mechanism for removing and feeding the pulse units into loading tubes during coast periods, from which the "ready-to-fire" racks within the propulsion module are restocked after a period of operation. The magazines are designed to be expendable and were estimated to be required in sufficient quantity to attain significant unit-cost reductions due to "learning." The procurement unit cost of magazines for the 10-m module was estimated as \$150/kg and the larger magazines for the 20-m module as \$115/kg.

The propellants used for vehicle guidance during periods of propulsion (as well as the artificial gravity spin-up and spin-down propellants accounted for as operational payload) are storable liquids which were costed at \$5/kg. The same unit cost was used for "start-up fluids," which are largely liquified gases used to bring the shock-absorber systems up to pressure prior to a propulsion period and to replace coast-period losses. The \$5/kg unit cost for these propellants and fluids is quite conservative if materials alone are considered (as it was also for foods, breathing gases, and other expendables in the operational payload's ecological system of Table 8.1); this value is used as a minimum to allow some expense for space-system packaging or handling provisions.

8.2. NUCLEAR-PULSE PROPELLANTS

Propellant costs for nuclear-propelled vehicles are typically a larger fraction of the total direct operating cost (DOC) than are propellants for chemically propelled vehicles. Nuclear-pulse propellants provide no exception to this statement, especially for the smaller (10-m) vehicles of this study. For exploration missions using the 10-m propulsion module, the nuclear-pulse-propellant costs accounted for some 25 to 30 percent of the DOC; for orbit-launched lunar systems, the propellant accounted for 12 to 15 percent. The percentages would have been considerably higher if the DOC were not dominated by the cost of the chemical ELV, which typically

accounts for some 60 to 65 percent on exploration missions and 70 to 80 percent on orbit-launched lunar missions (see Vol. II, Sec. 4, "Mission Cost Indications").

The nuclear-pulse-propellant costs for this study, however, are considered conservative, as has been previously stated. They are based on the use of currently well-understood nuclear-explosive-device technology and thus they do not reflect any potential developments in explosive devices intended particularly for propulsion.

The propellant costs used in this study do, on the other hand, reflect a considerable amount of cost reduction due to "learning." The costs of concern here are propellant costs for the first operational vehicles, which will follow the production and use of some thousands of pulse units during development and qualification of the propulsion systems. (Current development planning estimates indicate 6,000 to 7,000 pulse units will be used, mostly in the flight qualification phase.)

The costs per pulse unit used for this study, with a breakdown displaying the major cost components, are shown in Fig. 8.3. Cost-breakdown bars are shown for pulse units defined for propulsion modules of various effective thrusts, which cover a wide range of module sizes. It will be noted that there is very little difference in the pulse-unit costs for the first three different thrusts, and there is no difference in fissionable-material costs for these three cases. It will also be noted that the smallest cost bar represents the pulse unit for the 10-m propulsion module ($F_E = 3.5 \times 10^6$ N) and that the second-to-smallest bar is nearly large enough to represent the pulse unit for the 20-m module ($F_E = 16 \times 10^6$ N).

There is no difference in the cost or amount of fissionable material shown for the smaller pulse units represented (for propulsion modules up to thrusts of some 28×10^6 N, or 6×10^6 lb). The range of yields required of the nuclear devices (less than 1 KT to approximately 15 KT), assuming current technology devices are used, reportedly do not change the amount of fissionable material required. The amount of fissionable materials used for the three lower-cost pulse units was the cost equivalent of 2.9 kg of plutonium. The plutonium cost used was $18,000/kg.

For the three lower-cost pulse units shown, and again using current nuclear-device designs, it is possible to use less fissionable material and produce lower-cost pulse units, but at an increase in the nuclear-device mass. In this manner the amount and cost of the fissionable material can be reduced to the amount indicated by the dotted line
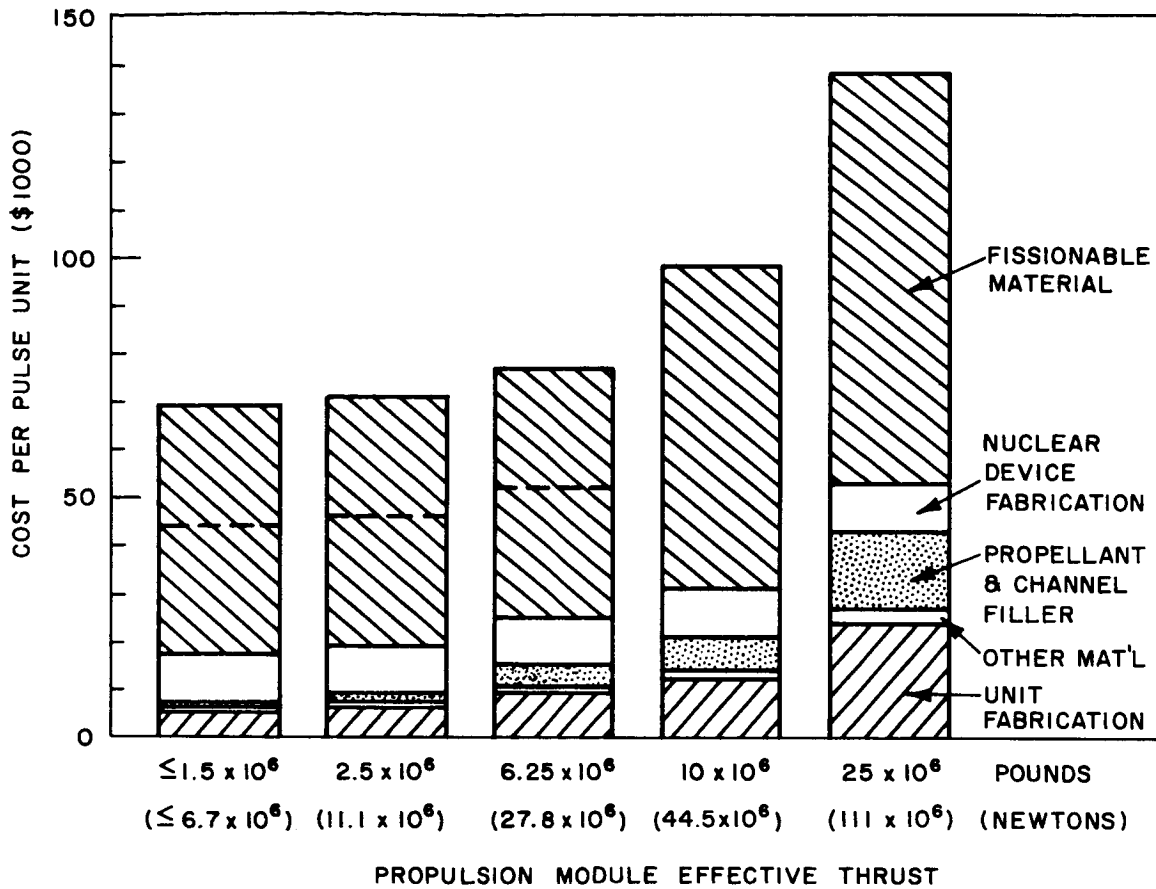
Fig. 8.3--Nuclear-pulse-propellant cost breakdown per pulse for
a range of propulsion module thrusts

shown in the three cost bars. The resulting increased mass of the pulse
unit, however, causes a reduction in I_{sp} such that the change was found
uneconomical from a systems viewpoint; thus, the higher cost, but higher
I_{sp}, data were used exclusively in this study.

The least-understood cost component in the pulse-unit breakdown
is that for the nuclear-device fabrication. The implosion system of the
device is understood to be complex and to require close tolerances. Its
trigger and circuitry obviously need to be highly reliable. A $10,000 per
unit allowance was made for fabrication, after considering, as previously
mentioned, a prior production of some thousands of units during the
propulsion-system development.

The remaining cost is largely for fabrication of the pulse unit,
exclusive of its nuclear device and the fissionable material therein.

The materials used in the pulse unit, relative to the fissionable material at least, are relatively common and inexpensive. They were costed at from \$2 to \$12/kg for nonfabricated materials. Modest quantities of material were required in all but the larger pulse units, which use large masses of propellant* and channel filler, which, in turn, cause a significant cost increment.

The mass of the pulse units increases rapidly with increasing thrust (although somewhat less than linearly with thrust as reflected by the increasing I_{sp}), whereas the pulse-unit cost, as shown in Fig. 8.3, increases only slightly, and hence results in a rapidly decreasing propellant cost per kilogram of over-all propellant with increasing thrust, as shown in Fig. 8.4. The solid curve shows the nominal nuclear-pulse-propellant costs used in this study. These costs per kilogram include the mass and cost of all material expended to attain the desired impulse: coolants, ejection gases, antiablation oil, etc., as well as the pulse units. The curve then represents total propellant cost (in the vehicle-system sense) and the cost values are properly applied directly to the expended propellant mass from "rocket" mass-ratio equations. The cost values are in 1963 dollars, as are the other cost data of this study.

The lower dotted curve of Fig. 8.4 shows the propellant unit cost currently predicted to result from a redesign of the nuclear device for propulsion purposes. It indicates a factor of 4 cost reduction for the thrust of the 10-m propulsion module and a 2.8 factor for thrusts of the 20-m module. The upper dotted curve reflects a factor of 2 cost increase over the nominal curve (presumed to be a conservative estimate of possible error in that direction). The area between the dotted curves represents a rather large area of uncertainty. The system-cost sensitivity to propellant-cost differences of this magnitude are shown in Sec. 5 of Vol. II.

8.3. ELVs, LOFTERS, AND LAUNCH OPERATIONS

Cost data on procurement and launch operations for earth-launch vehicles (ELVs) were for the most part taken from then-current information on Saturn V. These data were supplemented by postulated data on post-Saturn vehicles that were influenced by some known results of recent advanced- ELV studies. For lofters, several sources and methods were used. When using the Saturn S-1C as a lofter, the Saturn-V ELV data were modified to suit the situation. For lofters considered in the earlier broad parametric phase of the study, two concepts were used: (1) the

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*Propellant is used here in the more restricted sense of the pulse-unit design and refers to the slab of dense material which is vaporized and projected toward the pusher to produce the desired impulse (see Sec. 2 of this volume).
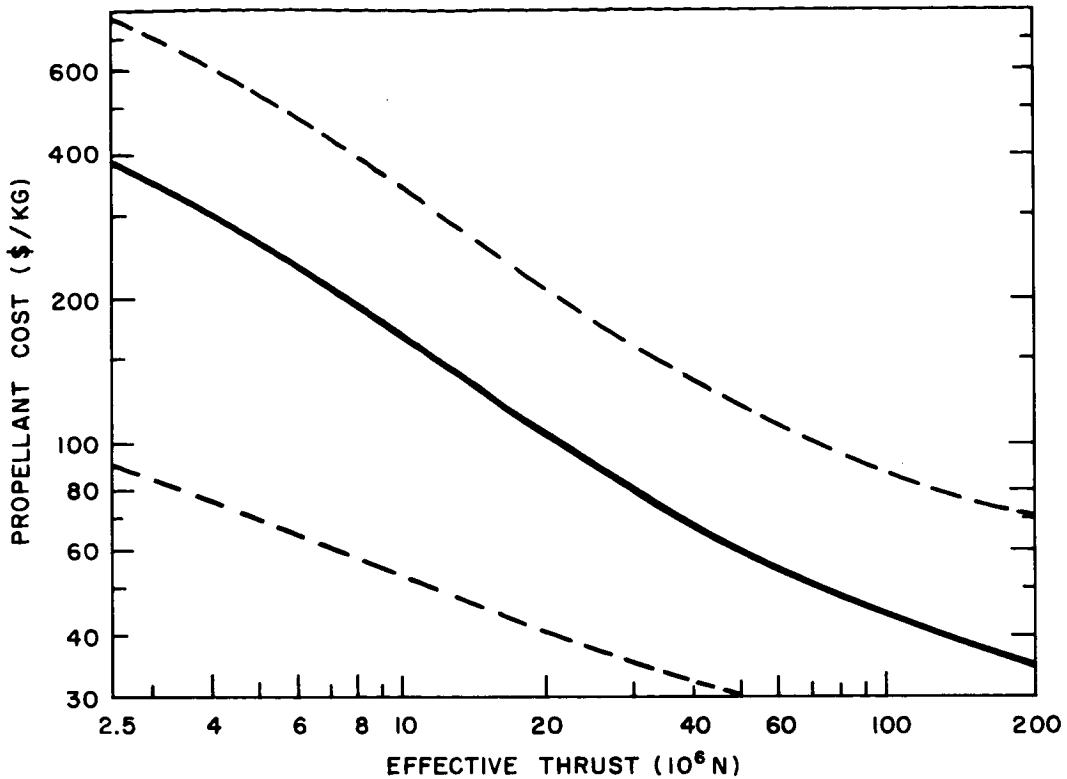
Fig. 8. 4--Nuclear-pulse-propellant estimated over-all cost
per kilogram as a function of effective thrust

adaptation of data from the General Dynamics/Astronautics Nova studies
and (2) the building up of lofters in many sizes by the clustering of solid-
propellant rocket motors.

8.3.1. The Saturn V Operating Costs

Saturn V operating-cost data were received in the form of a DOC
breakdown estimate (as of July, 1963) for the eleventh launch vehicle
(SA 511). A 90 percent learning curve was said to apply for estimates of
subsequent launch costs. The cost breakdown for the eleventh Saturn V
ELV was as follows:

S-1C stage, including engines	$30×10^6$
S-II stage, including engines	$16×10^6$
S-4B stage, including engines	$8×10^6$
Instrument unit and guidance	$6×10^6$
System integration.	$10×10^6$
Launch operations	$4×10^6$
Total	$74×10^6$

157

In the Saturn V ELV operations of this study, only the first two stages are used. The 8×10^6 estimated for the S-4B stage is therefore not required. This amount, however, is conservatively considered to be of the right order to defray the unique operational costs of separately orbiting the nuclear-pulse-propulsion module, its operational payload, and its propellant. The propulsion module, as discussed earlier in this section, requires a somewhat more complex interstage adapter than do most payloads. The ELV flights carrying nuclear-pulse propellant will require special nuclear safety precautions, probably under the direction of the Atomic Energy Commission, which will add to operational costs (estimates of several million dollars per launch have previously been made). Further, some means of attaining the final position closure prior to orbital rendezvous will add some cost to the over-all ELV operations. For these several reasons, the DOC total was assumed to remain as if the S-4B stage were used, but with its cost transferred to the system integration and launch operations headings.

The cost per launch as a function of the number of launches, displaying the 90 percent learning curve, is shown in Fig. 8.5. It is seen that at about the 140th launch the cost has dropped to 50×10^6 per operation. It was assumed that at least this many Saturn V operations will have taken place by the late 1970's so 50×10^6 per Saturn V operation was used for the 10-m vehicle systems of this study. (For the lunar logistic and ferry missions investigated, this cost was expressed as $500/kg of net payload delivered, since the net delivery capability to a desirable rendezvous earth orbit is 100,000 kg or slightly greater.)
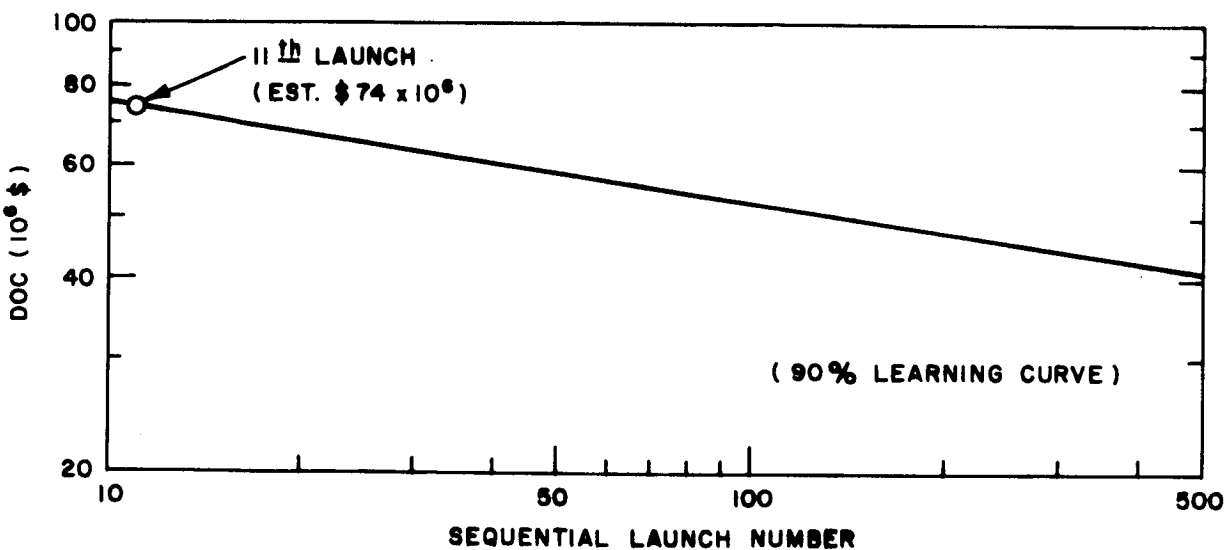

Fig. 8.5--Saturn V ELV direct operating cost per launch estimate versus launch number

For the larger 20-m exploration systems considered in this study, which were assumed in a later time period and which would require a larger number of launches, 40×10^6 per Saturn V operation was used rather than 50×10^6. From an extrapolation of the learning curve of Fig. 8.5, this corresponds to the predicted cost at about the 600th launch. Not only does this

158

assume a considerable use of Saturn V, but also a typical 20-m exploration mission requiring some 24 Saturn V launches (including seven redundant launches to account for success probabilities). Some sort of advanced Saturn or post-Saturn vehicle would appear highly probable in the event such 20-m vehicle operations were to take place. A post-Saturn vehicle, therefore, was also postulated. It was of the million-pound-to-orbit class (400,000 kg net payload delivery per launch) and was assumed to operate for 60×10^6 per launch.

The Saturn V DOCs discussed above are for procurement, systems integration, and operations only; no factors or cost allowances are included for probabilities of success. Success probabilities were applied to compute the number of launches and the redundant procurement required for exploration missions and to attain cost factors for repetitive logistic and ferry missions. These success probabilities and their applications are covered in detail in Sec. 4 of Vol. II.

8. 3. 2.   Cost Estimates for Lofters

The Saturn S-1C stage was used as a lofter for the earth-launched lunar logistic systems of this study.   For this lofter system the appropriate DOC elements were taken from the Saturn V ELV cost breakdown given above.   The 90 percent learning curve effect for the late 1970 time period assumed was applied as a factor equal to 50/74 (50×10^6 estimated Saturn V DOC in that time period compared to 74×10^6 for the eleventh vehicle breakdown).   Procurement of the S-1C stage was then 50/74 (30×10^6) or 20.25×10^6, while instrumentation and guidance became approximately 4×10^6 and systems integration became 6.75×10^6.   Launch operations were similarly ratioed and subsequently increased by more than 2×10^6 to allow for nuclear-safety precautions; the final estimate being 5×10^6 for the earth-launched logistic operations.

Earlier in the study, lofters were considered for a broad-range parametric investigation of system cost effectiveness.   Three kinds of lofters were considered in this investigation: those built up by clustering known sizes of solid propellant motors, those using the first stage of expendable ELVs being studied at that time under the Nova program, and recoverable ELVs then being studied.

The major cost component for the clustered-solid-motor lofters is, as would be expected, the solid-propellant motors themselves. (It should be remembered that the solid-motor lofters of this study, as discussed in Sec. 4, are conceived as relatively simple thrust producers only.   No guidance or thrust-vector control is required for the lofter since these functions exist in the nuclear-pulse vehicle for its own propulsion mode.) The procurement cost per pound used for these relatively simple solid motors, as a function of the individual motor's initial weight, is shown in Fig. 8. 6.   The cost curve shown is a weighted average of a

6

UNIT COST ($/LB)

120"

156"

260"

BASIS: SIMPLE, SINGLE NOZZLE MOTORS
SMALL QUANTITY PROCUREMENT (~24)

10^5 5 10^6 5
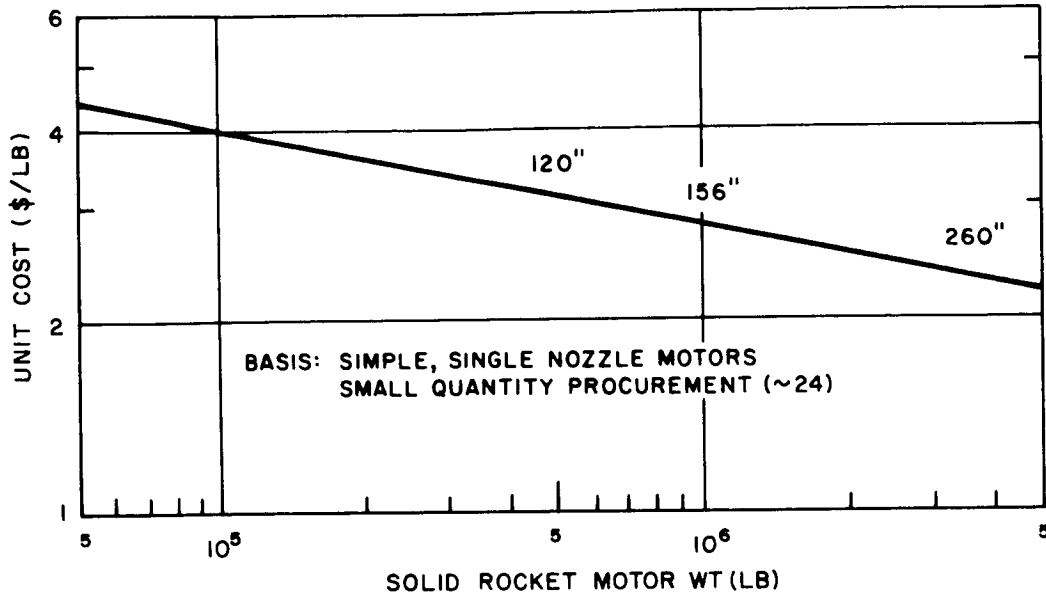
SOLID ROCKET MOTOR WT (LB)

Fig. 8.6--Estimated cost per pound of solid-propellant
rocket motors as a function of initial motor weight

rather wide variety of input. It is perhaps conservative by the standards
of most solid-propellant-motor producers and at the same time optimistic
by the standards of some other sources in the field of rocket motors.

The parametric data on complete operating costs for the solid-
motor lofters as a function of weight being lofted is shown in Fig. 8.7.
The lofter requirements were to loft a nuclear-pulse vehicle of a given
weight to a start-up altitude of not less than 90 km (approximately 300,000 ft)
and to a concurrent velocity not less than 3,000 fps. In addition to the
motors, costs were estimated for the clustering structure and a suitable
interstage adapter and for launch operations. The launch operations were
estimated to cost somewhat less than were the operations for a liquid-
propelled ELV first stage used as a lofter because of the simpler type of
motor.

The orbital-delivery cost data then available from the General
Dynamics/Astronautics Nova studies are summarized in Fig. 8.8 as a
function of orbital payload carried per launch. The raw data provided
included reliability effects which, as indicated, were factored out to attain
approximate cost per ELV launch (successful or not) so as to be comparable
with the other ELV and lofter data of the study (which have success prob-
ability data, when required, applied to the complete operational system).
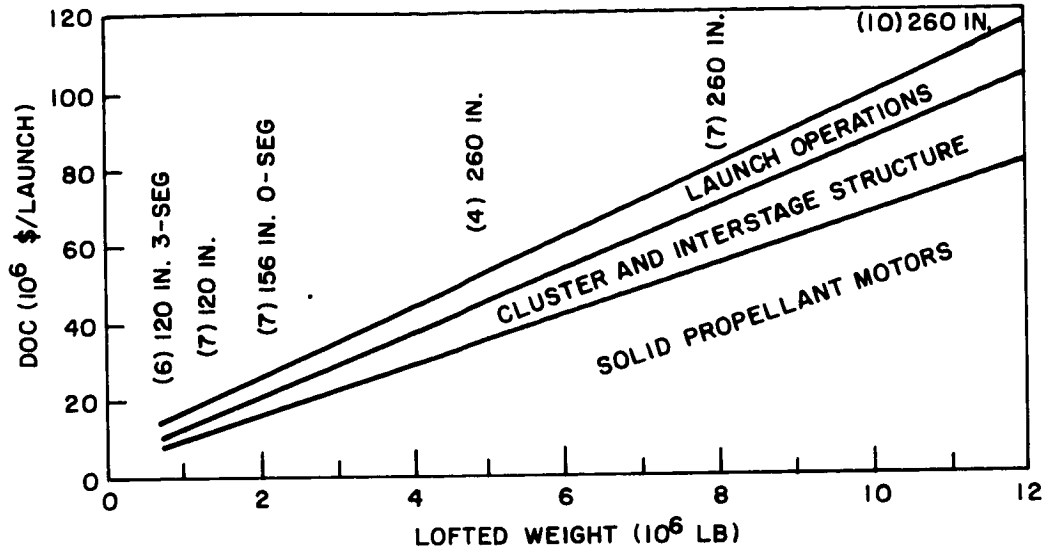It will be noted that the trends established by the relatively few design

160

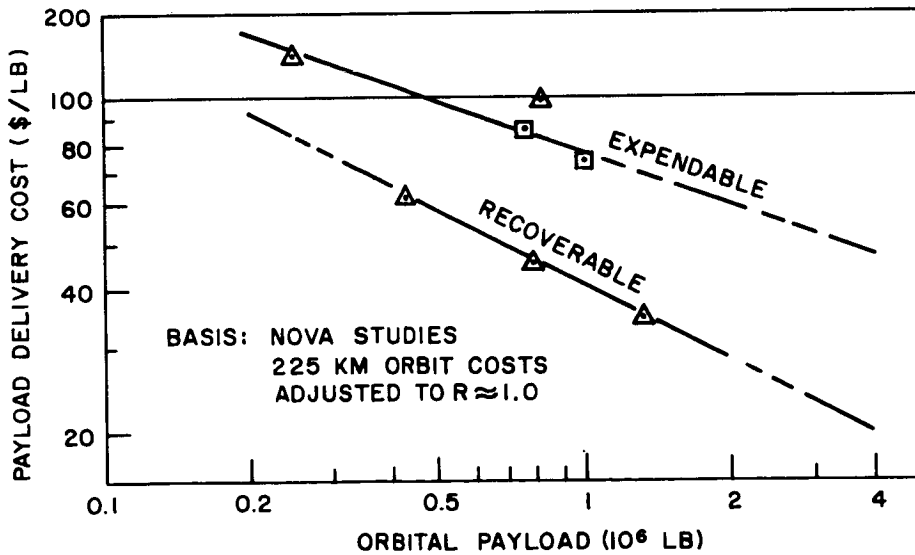Fig. 8.7--Estimated direct operating costs for solid propellant lofters and launch operations

Fig. 8.8--Estimated DOC per pound to orbit, using advanced Nova-class earth launch vehicles

points were rather freely extrapolated to cover the more extreme conditions of interest in the broad parametric investigation.

To apply the above data in estimating the direct cost for using the first stage of such ELVs as lofters, two simplifying assumptions were made: (1) After off-loading some chemical propellant from the first stage and adding structure to accommodate the larger mass to be lofted, the lofted-mass capability is approximately eight times the payload that the ELV can place in the 225-km orbit (this approximation followed several point-design estimates which resulted with this ratio). (2) The added direct cost for structural reinforcement, interstage structure, and nuclear-safety provisions was assumed equal to the cost of the ELV upper stage, which would not be used in the lofting operation.

All of the above lofter cost data for the broad parametric investigation resolved into two curves, one for recoverable and one for expendable lofters, as shown in Fig. 8.9. The cost difference indicated between the clustered solid lofters and the use of ELV first stages was considered lost in the noise level of the data and the input assumptions, at least for the purposes of the investigation at hand. (The data points for expendable lofters do indicate, however, that a somewhat steeper slope would be preferred for the clustered-solid-motor curve, whereas an ELV first-stage curve would more nearly parallel the recoverable curve. Clustered-solid lofters would then appear more economical than ELV first stages for the very small lofted weights and more expensive for the very large.)
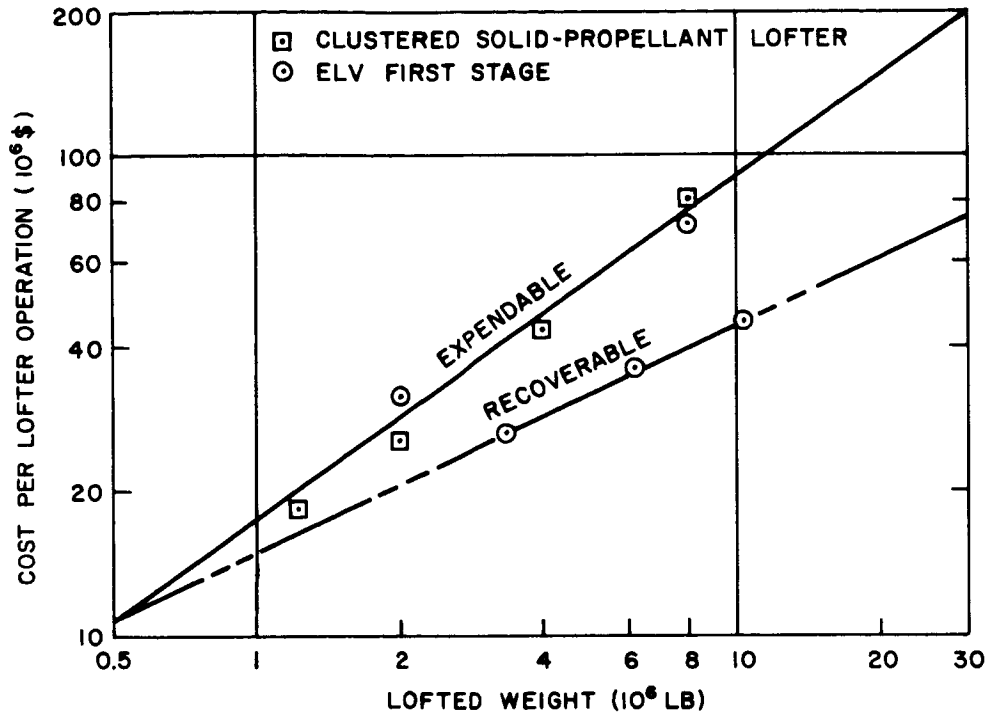
Fig. 8.9--Parametric study estimated direct operating costs
for expendable and recoverable lofter operations

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