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**GENERAL ATOMIC**  
DIVISION OF  
**GENERAL DYNAMICS**

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GA-5009, Vol. I

This document consists of 65 pages.  
No. 12 of 175 copies. Series A.

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NUCLEAR PULSE SPACE VEHICLE STUDY

Vol. I--SUMMARY

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

Contract NAS 8-11053

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## 1. INTRODUCTION

The Nuclear Pulse Space Vehicle Study performed for the National Aeronautics and Space Administration by the General Atomic Division of General Dynamics Corporation, with interdivisional assistance by both General Dynamics/Astronautics and General Dynamics/Convair, is summarized in this report. Most of the work reported was accomplished during the seven-month period from July 1963 through January 1964, although there was a large background of technical data available from previous Air Force-funded efforts. The study also drew upon and reports on parallel study efforts supported by the Air Force, particularly on propulsion-module design and developmental techniques.

The interdivisional assistance provided by General Dynamics/Astronautics dealt with mission velocity requirements and comparisons with other space propulsion systems. The assistance effort by General Dynamics/Convair consisted of a preliminary analysis of crew-compartment noise.

### 1.1 PROPULSION-SYSTEM BACKGROUND

Nuclear-pulse propulsion had received over five years of continuous analytical and experimental research prior to beginning this NASA study. Technical studies were initiated by General Atomic in 1957 with early Government support awarded in 1958, initially from the Advanced Research Projects Agency (ARPA). From 1960 to the present, the research support has been continued by the United States Air Force and by the General Dynamics Corporation.

A nuclear-pulse-propelled vehicle is shown conceptually in the frontispiece. Briefly, the propulsion system operates as follows: Low-yield nuclear pulse units are detonated consecutively external to and behind the vehicle. A substantial fraction of the mass of each pulse unit - the propellant - is directed toward the base of the vehicle as a high-velocity, high-density plasma which is intercepted by a large circular metallic plate - the pusher. The momentum of the propellant is transferred to the pusher and the resulting high accelerations are smoothed out by shock-absorbing devices to levels of a few g's in the upper vehicle, well within human tolerances. The propulsion-system performance is characterized by both high thrust-to-weight ratios and large specific impulses.

About half of the research effort to date has been experimental and was directed initially to demonstrating scientific feasibility. More recently, since the concept appears to be feasible without need of any scientific "breakthroughs," the efforts have included the determination of

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engineering practicability of the concept. This is being done through integrated propulsion-system design studies and applied research programs to provide technical information relating to pulse-unit design, pusher ablation, and structural integrity of the pusher, pusher attachment, and shock-absorbing systems.

Earlier design studies concentrated on vehicles of large sizes (4,000-ton gross weight and some 100 ft in diameter) and quite high specific impulse (4,000 sec and over). \* Such vehicles were intended primarily for nuclear-pulse operation starting just above the atmosphere and with initial thrust-to-weight ratios of  $\sim 1.25$ . At the conclusion of the parametric phase of the NASA study, it became apparent that very significant mission performance, under the less-demanding NASA mission constraints, at least, became available using much smaller and lower-specific-impulse ( $I_{sp}$ ) vehicles if operated at lower initial thrust-to-weight ratios. At the same time more detailed studies of the nuclear-pulse unit and its interaction with the vehicle eased the constraints previously applied to the smaller-sized vehicles. As a consequence, the current contractual design effort, part of which was to investigate a smaller, early developmental engine, has subsequently investigated designs of the 10-m-diam size. †

## 1.2. STUDY OBJECTIVES

This study was performed to explore the mission potential of the nuclear-pulse space vehicle concept in the accomplishment of missions meeting the requirements for lunar transportation, lunar logistic, and exploration or logistic missions to the planets, including Mars, Venus, and Jupiter. Both "favorable" and "unfavorable" departure years were to be investigated and fast-transit missions, as well as those more optimum from a mission-velocity viewpoint. A number of vehicle sizes and operational modes were to be considered. Operational requirements and problems were to be investigated and a development program outlined. The study was therefore of broad scope and limited depth and encompassed three primary and three secondary objectives:

### Primary Objectives

1. Determining the mission potential of nuclear-pulse space vehicles for lunar and planetary missions in the time period between 1975 and 1995 for a variety of mission profiles and operational objectives.

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\*Technical Summary Report, Nuclear Pulse Propulsion Project (Project ORION), Air Force Weapons Laboratory, RTD TDR-63-3006, Vols. I - IV, 1963 (S- ).

†Technical Summary Report, Nuclear Pulse Propulsion Project (Project ORION), Air Force Weapons Laboratory, Air Force Contract AF29(601)-6214 (to be published).

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2. Defining expected operational systems, operational problems, and possible approaches for solution to the problems.
3. Establishing typical development programs directed toward reaching major planetary mission capability in the 1975-1995 period.

Secondary Objectives

1. Developing parametric propulsion-vehicle-system design data in a form useful for the various mission studies.
2. Defining possible areas of growth or improvement in technology which would have a substantial influence on future performance, operations, or development programs.
3. Providing nuclear-pulse-vehicle performance techniques capable of simulating flight and indicating performance for any given vehicle.

1.3. STUDY APPROACH

The study was divided into two phases: a parametric phase to explore a very broad range of sizes and mission capabilities and then a specific-conceptual-system phase to investigate in greater detail the mission capability of two selected sizes of nuclear-pulse vehicles.

1.3.1 Parametric Study Phase

Four tasks were performed during this phase of the study.

1. Parametric characteristics defining the performance and operation of nuclear-pulse propulsion modules as functions of effective thrust were derived from earlier propulsion-system design studies over a wide range of thrust.
2. Vehicle systems were defined and "exercised" by computing their performance for a range of mission velocities encompassing the simpler and more difficult Mars explorations, lunar missions, and selected Jovian explorations. Concurrently, mission payload requirements were compiled. Three modes of operation were considered: (a) self-boost to orbit mode, called operational Mode I, which requires an effective thrust-to-weight ratio of  $\sim 1.0$  (after initial acceleration is provided by a chemically fueled rocket) to escape the earth's gravity. (b) An orbital start-up mode, Mode III, in which the nuclear-pulse vehicle is initially carried to orbit by a chemical

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booster; the thrust-to-weight ratios for Mode III can be well under 1.0. (c) An intermediate Mode II in which the propulsion module is loaded in orbit with additional propellant and/or payload after self-boosting.

3. Comparative direct operating costs and the major cost components were computed in a simplified cost analysis to derive the more economical operating modes and vehicle sizes over the broad range of nuclear pulse systems being explored.
4. Operational problems and hazards unique to nuclear-pulse propulsion were explored so as to define and quantify, to first order, the magnitude of the problem and to identify those problem areas that require further attention.

The most useful guidance from the parametric study phase came from the realization of the performance potential of the orbital start-up, reduced thrust-to-weight ratio, mode of operation. The performance capability of single-stage vehicles in the smaller sizes (~10 m in diameter and thrusts well under one million pounds) was found adequate for significant planetary explorations with comfortable performance and payload margins. A 10-m propulsion module with a dry weight compatible with the orbital delivery capability of Saturn V was therefore selected as one of the two sizes to exercise during the specific-conceptual-systems study phase.

#### 1.3.2. Specific-conceptual-systems Study Phase

During the second study phase, five tasks were performed; the major portion of the contractual effort, however, was devoted to the first three.

1. Two specific nuclear-pulse-propulsion modules were defined in conjunction with specific manned payloads for a variety of Mars and Jupiter explorations, for Mars logistic delivery, and for lunar logistic and personnel transport. The propulsion modules were sized to be compatible with earth-launch vehicles planned for the same time periods. The module compatible with Saturn V, a 10-m-diam (32.8 ft) configuration, received particular emphasis after it appeared capable of more than adequately performing most of the exploration and space logistic tasks of early interest. The second module, of 20 m diam (65.6 ft), is compatible with post-Saturn design concepts.
2. Performance and approximate direct costs were determined for the space missions mentioned above.

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3. A sensitivity analysis was made by varying, one at a time, the more suspect vehicle-performance or unit-cost inputs and recomputing the total mission performance or costs.
4. A tentative development plan and schedule was generated for an orbital start-up 10-m propulsion module.
5. Advanced versions of nuclear-pulse vehicles and their performance and economic potential were reviewed. These data are based on performance characteristics predicted by exploiting known fundamental properties of nuclear fission and fusion devices.

## 2. MISSION REQUIREMENTS

The major mission requirements for the specified missions of this study fell into two categories: mission velocity and mission payload. Both were handled parametrically during the early study phase; i. e., a wide range of undefined payload mass was required to be transported through mission velocities encompassing the range of mission interest (~30,000 to 300,000 fps). Near the midpoint of the study, a number of more specific requirements were defined so that the selected 10-m and 20-m nuclear-pulse vehicle systems could be exercised in specific missions. The more important of these specific mission requirements are summarized here.

Velocity increments for the various maneuvers of representative missions to the moon, Mars, Venus, and a moon of Jupiter were compiled primarily from the study assistance provided by General Dynamics/Astronautics.\* A 3 percent performance reserve allowance was applied to most of the mission velocity increments and 5 percent was applied to the lunar-orbit-to-lunar-surface shuttle maneuvers and to terminal maneuvers on lunar landing missions.

The velocity increments tabulated in this study were based on impulsive-maneuver calculations. These velocity increments, with the 3 percent reserve allowance, were used when earth-orbit-departure thrust-to-weight ratios were approximately 0.5 or greater for typical modest  $\Delta V$  earth-departure maneuvers, since gravity losses are then insignificant. When lower departure thrust-to-weight ratios were used or the earth-departure maneuver  $\Delta V$  was high, additional gravity loss penalties (over that buried in the performance reserve allowance) were applied. The earth-departure gravity-loss curves used for the 10-m propulsion module ( $I_{sp} = 1,850$  sec) are shown in Fig. 2.1. Still higher gravity losses were applied for the 20-m module ( $I_{sp} = 3,150$  sec), since at any given initial thrust-to-weight ratio a higher  $I_{sp}$  results in somewhat greater penalties (this is because the thrust-to-weight ratio of the lower  $I_{sp}$  systems increases more rapidly as the velocity increment is attained because of the more rapid propellant consumption).

### 2.1. LUNAR-MISSION VELOCITIES

The lunar missions considered were (1) lunar ferry missions, for which the vehicles are manned, reusable, and designed to accommodate both passengers and cargo, and (2) lunar logistic missions, for which the vehicles are used to deliver cargo only.

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\*See Vol. IV.

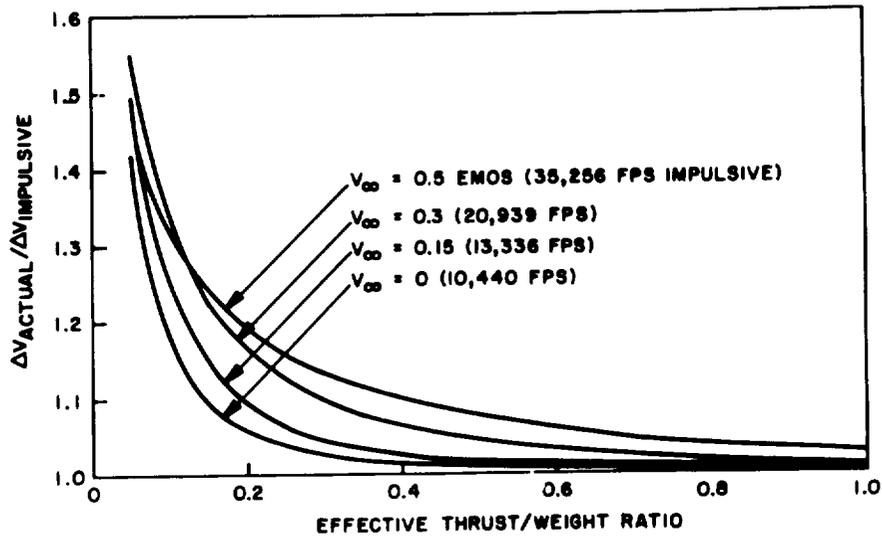


Fig. 2.1--Earth-departure gravity losses vs thrust-to-weight ratio ( $I_{sp} = 1,850$  sec) for 10-m propulsion module

Two operational plans considered for the lunar ferry missions were earth-orbit-to-lunar-orbit and earth-orbit-to-lunar-surface. For earth-orbit-to-lunar-orbit and return, nuclear-pulse propulsion is used and then chemical-rocket-powered shuttle vehicles are used to transfer cargo and passengers from the lunar orbit to the lunar surface. The velocity requirements for the nuclear-pulse lunar ferry are shown in Table 2.1. The earth-departure and earth-orbit-capture requirements were based on operation from a 325-km earth orbit and a 72-hr transit time. A 15° plane change capability was provided to permit rendezvous with the lunar shuttles in a lunar equatorial orbit. The velocity requirements for the chemically-powered lunar shuttles totalled 2,474 m/sec (8,119 ft/sec) for descent and 1,982 m/sec (6,501 ft/sec) for ascent.

Table 2.1

**LUNAR-FERRY VELOCITY REQUIREMENTS  
FOR ORBIT-TO-ORBIT MODE**

<u>Maneuver</u>	<u>ΔV Requirements</u>	
	<u>(m/sec)</u>	<u>(ft/sec)</u>
Earth departure (without losses) . . . . .	3,234	10,610
Outbound midcourse correction . . . . .	152	500
Lunar-orbit capture . . . . .	910	2,987
Plane change . . . . .	439	1,440
Lunar-orbit departure . . . . .	910	2,987
Return midcourse correction . . . . .	152	500
Earth-orbit capture . . . . .	<u>3,234</u>	<u>10,610</u>
Total Velocity Required . . . . .	<u>9,031</u>	<u>29,634</u>

The velocity requirements for the lunar ferry operating between earth orbit and the lunar surface are given in Table 2.2. In this system the nuclear-pulse operation is terminated at an altitude of approximately 6 km (20,000 ft) above the lunar surface. During descent, the final velocity increment of 1,012 m/sec is provided by a chemical rocket system. During ascent, the chemical rocket system provides the initial 640 m/sec velocity increment, after which nuclear-pulse operation is begun.

Table 2.2

**LUNAR-SHUTTLE VELOCITY REQUIREMENTS  
FOR ORBIT-TO-ORBIT MODE**

<u>Maneuver</u>	<u>ΔV Requirements</u>	
	<u>(m/sec)</u>	<u>(ft/sec)</u>
<b>Descent:</b>		
Retrothrust . . . . .	146	479
Descent ellipse midcourse corrections . . . . .	18	58
Theoretical ideal (supersynchronous pericyynthion) . . . . .	1,746	5,728
Gravity losses . . . . .	114	375
Lunar-surface speed . . . . .	5	16
Pilot requirements . . . . .	236	775
Performance reserve (5 percent) . . . . .	<u>118</u>	<u>388</u>
Total descent velocity . . . . .	<u>2,474</u>	<u>8,119</u>
<b>Ascent:</b>		
Lunar-surface speed . . . . .	5	16
Theoretical ideal (Hohmann ellipse pericyynthion) . . . . .	1,711	5,615
Gravity losses . . . . .	116	380
Midcourse corrections . . . . .	17	55
Rendezvous . . . . .	38	125
Performance reserve (5 percent) . . . . .	<u>95</u>	<u>310</u>
Total ascent velocity . . . . .	<u>1,982</u>	<u>6,501</u>

In the unmanned lunar logistic missions studied, nuclear-pulse operation is initiated suborbitally, but after the vehicle has been lofted above the earth's atmosphere by a chemical-rocket first stage. Two modes of operation near the moon were explored, with the nuclear-pulse vehicle staged and expended in each mode. In the first mode, nuclear-pulse operation is terminated in the lunar orbit; in the second mode the switchover to chemical propulsion is accomplished near the lunar surface to utilize more fully the nuclear-pulse capability. A high-energy liquid-propellant chemical stage is used for landing the cargo on the lunar surface in both modes. The velocity requirements for the lunar logistic vehicles, for both modes of operation, are summarized in Table 2.3.

**2.2. EXPLORATION-MISSION VELOCITIES**

A relatively small number of specific planetary missions were selected from the number of possible variations. Most of those selected are round-trip exploration missions that start from an earth orbit, remain for a few weeks in a circular orbit of the target planet, and return to earth by one of several methods. The specific earth-approach method used causes greater variations in the mission-velocity requirements than do other variables, such as the year of departure, and hence several earth-approach conditions were considered.

Table 2.3

LUNAR-LOGISTIC-VEHICLE VELOCITY REQUIREMENTS

<u>Maneuver</u>	<u>ΔV Requirement</u>	
	<u>(m/sec)</u>	<u>(ft/sec)</u>
Earth surface to earth orbit		
Solid-propellant lofted . . . . .	10,674	35,020
Saturn S-1C boosted . . . . .	10,360	33,990
Earth-orbit departure . . . . .	3,234	10,610
Midcourse correction . . . . .	152	500
Lunar-orbit capture . . . . .	910	2,987
Lunar descent		
Lunar-orbit termination of nuclear pulse propulsion* . . . . .	2,475	8,119
Near-surface termination of nuclear pulse propulsion:		
Initial phase . . . . .	1,341	4,400
Final letdown* . . . . .	1,012	3,320

\*Maneuvers by chemical-propellant rocket stages.

Mars missions were investigated in several variations since Mars is an exploration target of major interest and since departure year as well as trip duration has a significant effect on mission velocity requirements. A reference departure year of 1975 was selected for variations in other parameters.

A breakdown of the mission velocity requirements for Mars missions having four selected earth-approach conditions is given in Table 2.4. For the minimum mission velocity condition (zero ΔV for the arrival maneuver) a hyperbolic rendezvous with an earth-launched pickup vehicle was assumed as earth is approached. Thus, for this condition, neither earth-approach retrothrust nor a reentry system is required of the interplanetary vehicle (although rather demanding requirements are then placed on the pickup vehicle). Two of the earth-approach conditions require retrothrust sufficient to decelerate to a given reentry velocity: 50,000 ft/sec (fps) for one and 36,000 ft/sec, or approximately APOLLO conditions, for the other. The fourth earth-approach condition requires deceleration to a circular-orbit capture velocity.

For the effect of the selected departure year, one full cycle of mission velocity for favorable through unfavorable years (1973 through 1984) was selected. These mission velocity requirements are summarized in Table 2.5.

Table 2.4

1975 MARS MISSION VELOCITIES FOR VARIOUS EARTH APPROACHES

Maneuver	Earth-approach Condition							
	M-1, Hyperbolic Rendezvous		M-2, 50,000-fps Reentry		M-3, 36,300-fps Reentry		M-4, Circular-orbit Capture	
	m/sec	(ft/sec)	m/sec	(ft/sec)	m/sec	(ft/sec)	m/sec	(ft/sec)
Earth departure	4,458	14,625	4,419	14,498	4,419	14,498	4,419	14,498
Outbound midcourse correction	305	1,000	305	1,000	305	1,000	305	1,000
Mars capture	3,932	12,899	4,912	16,117	4,912	16,117	4,912	16,117
Mars departure	6,077	19,939	6,026	19,772	6,026	19,772	6,026	19,772
Return midcourse correction	457	1,500	457	1,500	457	1,500	457	1,500
Earth arrival	0	0	6,085	19,963	10,477	34,373	13,554	44,468
Total velocity requirement	15,229	49,963	22,204	72,850	26,596	87,260	29,673	97,355
Reference Departure Date	10-3-75		9-11-75		9-11-75		9-11-75	

Note: All velocities include 3 percent performance reserve. Mission duration (days) = 160 transfer + 30 capture + 20 departure window + 230 transfer = 440 days.

Table 2.5

MARS MISSION VELOCITIES FOR SELECTED DEPARTURE YEARS

Year	Departure Date	Mission Velocity Requirement					
		Outbound		Return		Total	
		m/sec	(ft/sec)	m/sec	(ft/sec)	m/sec	(ft/sec)
1973	1-28-73	12,650	41,500	9,850	32,300	22,500	73,800
1975	6-7-75	11,950	39,200	14,150	46,400	26,100	85,600
1977	10-24-77	8,400	27,600	19,600	64,300	28,000	91,900
1979	12-23-79	8,300	27,200	18,000	59,200	26,300	86,400
1982	1-11-82	8,750	28,800	15,250	50,000	24,000	78,800
1984	1-21-84	11,550	37,900	11,450	37,600	23,000	75,500

Note: 450-day missions with 30-day Mars capture plus 20-day departure window; return earth-approach deceleration to subparabolic velocity (36,300 fps).

Faster transit Mars missions than the nominal 440- to 450-day duration were investigated. Velocity requirements were found to increase rapidly as durations decreased below 300 days. For this reason, a 300-day mission was selected as a typical fast-transit mission for the 10-m exploration vehicle. A 150-day Mars mission was selected for the 20-m vehicle since this vehicle has a higher single-stage mission potential due to its higher specific impulse. Velocity requirements for these two fast-transit missions are given in Table 2.6.

Venus exploration missions were also exercised. As also indicated in other studies, the Venus missions are generally of shorter duration, require less mission velocity than Mars missions, and are less perturbed by the selected departure year. Three earth-approach conditions were investigated: a <50,000-fps reentry velocity, a ~36,300-fps reentry, and a circular-earth-orbit capture. The velocity requirements were 16,970 m/sec (55,600 ft/sec), 20,930 m/sec (68,600 ft/sec), and 23,990 m/sec (78,600 ft/sec), respectively.

Table 2.6

**SELECTED FAST-TRANSIT MARS EXPLORATION MISSIONS**

Maneuver	300-day Mission		150-day Mission	
	m/sec	(ft/sec)	m/sec	(ft/sec)
Earth departure	9,340	30,600	12,400	40,800
Outbound midcourse correction	300	1,000	300	1,000
Mars capture	6,560	21,500	21,900	72,000
Mars departure	11,140	36,500	17,800	58,500
Return midcourse correction	450	1,500	450	1,500
Earth arrival	<u>4,030</u>	<u>13,200</u>	<u>10,700</u>	<u>35,000</u>
<b>Total velocity requirement</b>	<b>31,820</b>	<b>103,300</b>	<b>63,550</b>	<b>208,800</b>
Reference departure date	3/29/75		11/28/77	
Outbound transfer time	180 days		60 days	
Capture/window time	20/13 days		10/7 days	
Return transfer time	87 days		73 days	

Jupiter missions have velocity requirements and mission durations that are typically at least double those of Mars missions. The Jupiter mission selected for sizing a nuclear-pulse-vehicle system is an exploration mission in which the Jovian moon, Callisto, is orbited. The mission velocity requirements and durations data are shown in Table 2.7.

Table 2.7

**VELOCITY REQUIREMENTS FOR A SELECTED JUPITER  
(CALLISTO ORBIT) MISSION**

Maneuver	$\Delta V$ Requirement	
	(m/sec)	(ft/sec)
Earth departure	25,000	81,800
Outbound correction	610	2,000
Callisto capture	12,300	40,400
Callisto departure	8,900	29,200
Return correction	610	2,000
Earth arrival	<u>16,320</u>	<u>53,600</u>
<b>Total velocity required</b>	<b><u>63,740</u></b>	<b><u>209,000</u></b>
Reference departure date	2/20/82	
Outbound transfer time, days	400	
Capture and window times, days	30, 20	
Return transfer time, days	<u>460</u>	
Mission duration, days	910	

**2.3. EXPLORATION-MISSION PAYLOADS**

Mission payloads were considered in two distinct ways in determining the performance of the specific nuclear-pulse-vehicle systems:  
 (1) For the planetary exploration missions, given numbers of expedition personnel and given missions were specified. Payloads were therefore

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subject to reasonably accurate estimation procedures and hence were specified in some detail. (2) For lunar missions and for other logistic delivery missions, the cargo and personnel payloads were treated parametrically. The payload gross characteristics were of interest, e. g., payload density and whether of bulk, fluid, cryogenic, etc. Suitable accommodations for personnel were provided, but other than a selected operating crew on manned systems, the number of personnel carried was treated as a variable.

Weight breakdowns for six different exploration payloads are summarized in Table 2.8. These are for 8-man and 20-man expeditions to Mars or Venus using vehicles propelled by the 10-m propulsion module. A similar breakdown of considerably larger payloads (for 20-man and 50-man explorations, with total operational payloads from 179,000 to 290,000 kg and carrying destination payloads of from 100,000 to 1,500,000 kg) was used to exercise the 20-m vehicle designs.

Table 2.8

PAYLOADS AND WEIGHTS FOR MARS AND VENUS EXPLORATION MISSIONS WITH 10-M VEHICLE  
(Weights in kilograms except as indicated)

Payload Components	Exploration Payloads					
	A, Nom. 8-man System 750-kg Payload	A <sub>1</sub> , (EMPIRE <sup>a</sup> Comparable)	B, Same as A plus 75,000-kg Payload	C, Same as B plus Convoy Ecology	D, Nom. 20-man 750-kg Payload	E, Same as D plus 150,000-kg Payload
<b>Operational payload:</b>						
Vehicle hardware						
Life-support system (dry)	20,700	19,020		20,700	42,100	
Radiation shielding	18,170	18,170		18,170	27,200	
Ecological system	2,977	2,977		3,360	5,870	
Total hardware	<u>41,847</u>	<u>40,167</u>		<u>42,230</u>	<u>75,170</u>	
Expendable hardware						
Food and ecological system	12,723	11,402	Same as Payload A	17,165	26,850	Same as Payload D
Abort propellant	4,500	4,500		4,500	9,000	
Spin propellant	4,540	4,540		4,540	6,800	
Checkout instrumentation	500	500		500	500	
Spare and repair equipment	3,400	3,400		3,400	3,400	
Taxi capsules and propellant	1,450	1,450		1,450	2,900	
Reentry vehicle and maneuver stage	(b)	(b)		(b)	(b)	
Personnel	725	725		725	1,810	
Total expendables and personnel	<u>27,838</u>	<u>26,517</u>		<u>32,280</u>	<u>51,260</u>	
Contingency (~5 percent)	3,315	3,316		3,490	6,570	
Total operational payload <sup>b</sup>	<u>73,000</u>	<u>70,000</u>	<u>73,000</u>	<u>78,000</u>	<u>133,000</u>	<u>133,000</u>
In-transit payload allowance	250	250	1,000	1,000	250	2,000
<b>Destination payload:</b>						
Mapping equipment	450		900			1,800
Data-handling and - storage system	250	Same as Payload A	500	Same as Payload B	Same as Payload A	1,000
Environmental satellites (4 nom.)	---	as	4,000	as	as	8,000
Unmanned landers (3 nom.)	---	as	6,060	as	as	12,120
Unmanned returners (3 nom.)	---	as	11,800	as	as	23,600
Manned excursion vehicles (3 nom.)	---	as	48,000	as	as	96,000
Contingency (~5 percent)	50		3,740			7,480
Total destination payload	<u>750</u>	<u>750</u>	<u>75,000</u>	<u>75,000</u>	<u>750</u>	<u>150,000</u>
<b>TOTAL MISSION PAYLOAD<sup>b</sup></b>	<u>74,000</u>	<u>71,000</u>	<u>149,000</u>	<u>154,000</u>	<u>134,000</u>	<u>285,000</u>
Total mission payload, lb	<u>163,000</u>	<u>157,000</u>	<u>328,000</u>	<u>339,000</u>	<u>296,000</u>	<u>628,000</u>
<b>Payload reduction during mission:</b>						
Outbound correction maneuver	5,700	5,400	5,700	5,700	11,000	11,000
Mars approach maneuver	6,400	7,400	6,400	6,400	13,000	13,000
Mars departure maneuver	7,700	8,600	81,500	82,100	15,200	163,000
Return correction maneuver	15,000	15,100	88,800	92,000	28,900	176,000
Earth approach	18,100	21,000	91,900	96,400	35,100	182,000

<sup>a</sup> Life-support structural arrangement (expendable modules) comparable to those of the EMPIRE Study (A Study of Early Manned Interplanetary Missions, Final Summary Report, General Dynamics Astronautics, Report AOK 03-0001, January 31, 1963).

<sup>b</sup> If drag reentry vehicle and its maneuver stage is required for ~3b, 300 fps earth-approach velocity, 4,000 kg is added to operational payload for 8-man missions and 10,000 kg for 20-man missions. For 50,000-fps approach, 7,000 kg is added for 8-man missions and 17,000 kg for 20-man missions.

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### 3. VEHICLE DESIGNS

The complete nuclear-pulse vehicle consists of the propulsion module plus an "upper vehicle." The upper vehicle, at the minimum, carries and protects the payload and provides a guidance and control system. The propulsion module provides, in rocket terminology, the equivalent of an engine, tankage, plumbing, and thrust structure.

For the early parametric performance calculations, the upper vehicle was defined very simply; it was considered to have a cone or ogive configuration and its weight was a function of the propulsion-module weight and the weight of payload. During the later phase specific design configurations were considered.

This section on over-all vehicle designs first reviews the propulsion module characteristics, then summarizes the specific conceptual designs generated during the last phase of the study. Finally, the compatibility with these designs of Saturn V and other ELV's or lofters, is summarized.

#### 3.1. PROPULSION-MODULE CHARACTERISTICS

At the conclusion of the parametric study phase, two sizes of propulsion modules were chosen to be the basis for the subsequent specific design effort. The smaller, 10-m-diam vehicle has a dry weight of just under 91,000 kg (200,500 lb); it was selected, in part, because of its compatibility with the boost-to-orbit capability of Saturn V. The 10-m vehicles will be seen to have a performance capability that is more than adequate for most now-contemplated planetary exploration or lunar missions, so most attention was given to this size. The 20-m vehicles are primarily of interest for the more ambitious Mars missions and for explorations of Jupiter, etc.

The design of the 10-m propulsion module, as defined and "frozen" for this study, is shown in Fig. 3.1. The characteristics of this module were derived primarily by scaling laws, since it is smaller than those that had previously received significant design effort (by a factor greater than 2 in thrust and approximately 1.5 in dimensions). Subsequent design effort on 10-m modules have to date confirmed its performance potential and the general feasibility of even smaller modules.

The basic propulsion module shown in Fig. 3.1 is 21 m long; on the right are the payload structure and provisions for supplemental propellant. The 10-m-diam base of the vehicle on the left is a heavy steel pusher that weighs some 6/10 the dry weight of the complete module. Much of the

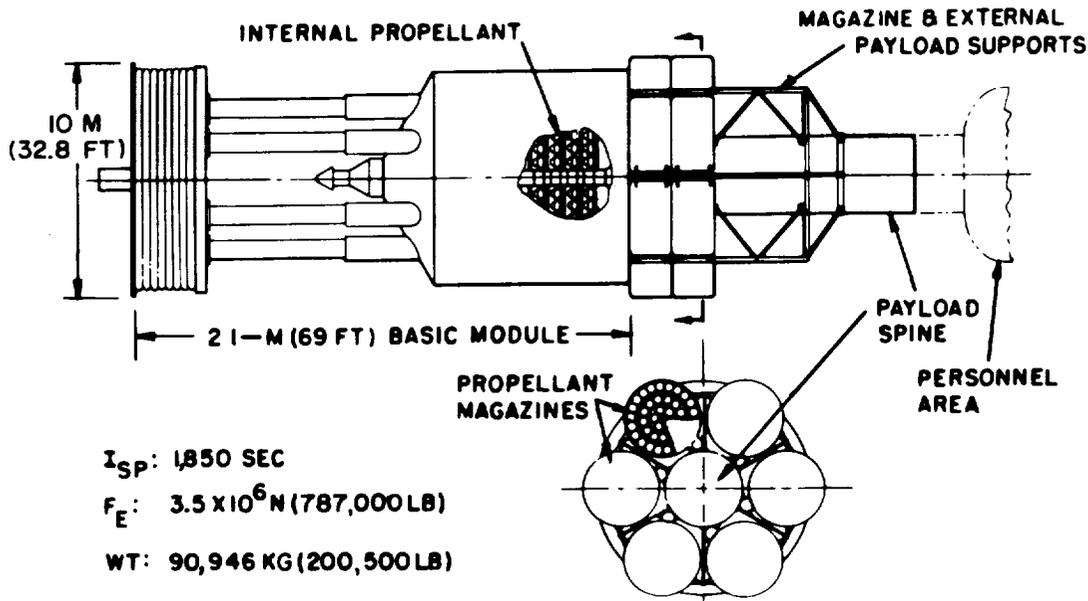


Fig. 3.1--Study configuration of the 10-m nuclear pulse propulsion module

propulsion module above the pusher consists of shock-absorbing stages to smooth out the high acceleration received by the pusher during propellant interaction. The first-stage shock absorber, immediately ahead of the pusher, is composed of concentric multilayer, gas-filled tori. It serves a function analogous to a tire on an airplane landing gear, smoothing the pusher acceleration to levels acceptable to the long-stroke, cylindrical second-stage shock absorbers.

The approximately 1 pulse/sec acceleration received by the pusher are attenuated, by the shock absorbers to levels in the module body and payload region as shown in Fig. 3.2. The pulsing acceleration profile is one of the unique operational characteristics of pulse propulsion. The maximum accelerations, however, even in an unloaded, near-burnout condition ( $F_E/W = 3.1$ ), are not large, and typical periods of powered flight last only some 5 to 15 min. (It is of interest to note that both the pulse frequency and the  $F_E/W = 1.25$  pulsing acceleration profile are reasonably well simulated by a child's backyard swing operating through an arc some  $65^\circ$  each way from vertical.) It will also be noted that negative accelerations are experienced during shock-absorber damping following shutdown (requiring some 15 to 20 sec) or in the event of a misfire. Hence, personnel as well as payload need to be properly restrained during periods of powered flight.

The body section of the propulsion module contains several floors of pulse units (propellant); the module designs of this study carry up to 900

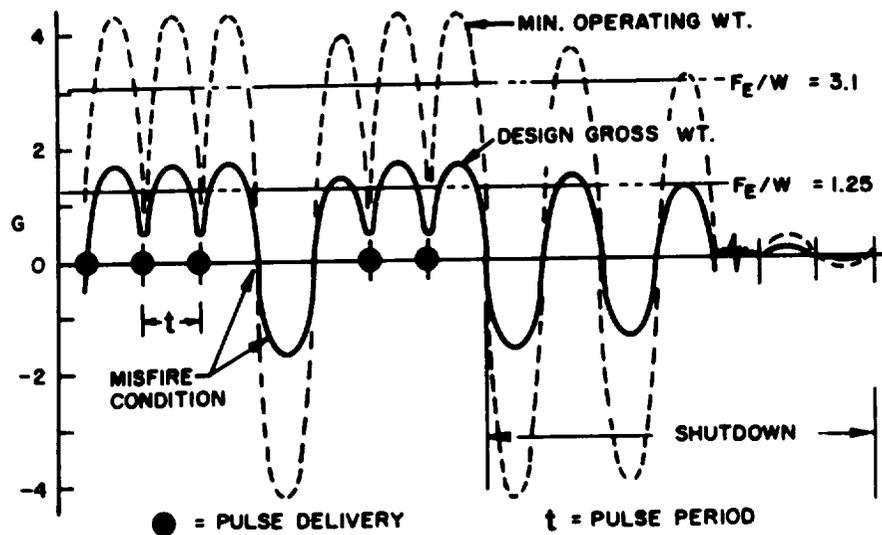


Fig. 3.2--Typical nuclear-pulse-vehicle acceleration profile

pulse units internally in "ready-to-fire" racks. The body section also contains the pulse-unit ejector tube, breech loading mechanism, and a control system. The ejector tube fires the pulse units through a shock-absorber protection tube, seen extending beyond the pusher along the centerline of the module.

The design of the 20-m module is very similar to the 10-m module shown. It is twice the diameter and approximately 1-1/2 times the length of the 10-m design and has a larger number of cylindrical second-stage shock absorbers.

Two other rather unique characteristics of nuclear-pulse propulsion have a strong influence on the performance and operating costs of the vehicles as a function of size. One is the variation of specific impulse with size, or with thrust if other operating conditions remain fixed. The curve of specific impulse versus effective thrust used for this study is shown in Fig. 3.3, with the selected 10-m module ( $I_{sp} = 1,850$  sec, thrust =  $3.5 \times 10^6$  N, or 787,000 lb) and 20-m module ( $I_{sp} = 3,150$  sec, thrust =  $16 \times 10^6$  N, or 3,600,000 lb) indicated. The curve indicates obvious performance advantages with increasing size or thrust.

The cost characteristic that varies with vehicle size is the over-all cost of nuclear-pulse propellant, which decreases rapidly with increasing thrust. For example, propellant for the 10-m module is estimated to cost \$320/kg, for the 20-m module \$120/kg, and for a still larger design (about 30-m diam. with a thrust of  $44 \times 10^6$  newtons) \$64/kg. The reason is that

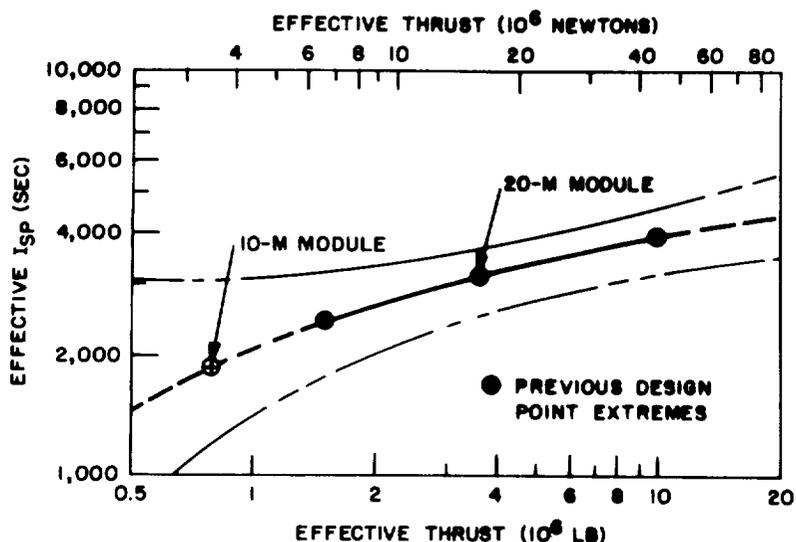


Fig. 3.3--Propulsion-module specific impulse

the relatively high cost of the pulse unit's nuclear explosive device (presently based on current state-of-the-art weapons technology) does not change significantly from the smallest sizes considered until propulsion module thrusts reach some  $28 \times 10^6$  newtons ( $6 \times 10^6$  lb). Conversely, the mass of the other pulse-unit components and associated fluids increases with increasing thrust, so the cost per kilogram decreases.

### 3.2. EXPLORATION VEHICLES

The specific conceptual vehicles of the second part of the study phase were designed to meet the specific requirements of both the missions and the propulsion systems. For exploration missions, many of the requirements stem from the needs of mission personnel. Personnel accommodations, radiation protection, and safety considerations therefore had much to do with configuring the over-all vehicles.

Shielding against nuclear radiation during powered flight is one personnel requirement. Some  $120 \text{ g/cm}^2$  of hydrogenous material plus some lead for direct radiation shielding (crew compartment bottom) and  $\sim 25 \text{ g/cm}^2$  of scatter radiation shielding (sidewalls and top) are required to keep a typical mission dose from propulsion to 50 rem. Because of the low neutron fluxes from the pulse units, the vehicle undergoes only mild activation, however, and propulsion shielding for personnel is required only during actual propulsion periods (typically lasting 5 to 15 min). The minimum thickness of side and top shielding provided ( $25 \text{ g/cm}^2$ ) is sufficient to attenuate probable solar-flare radiation to values similar to the 50 rem allowed for propulsion.

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Figure 3.4 shows the shielded powered-flight station for the personnel of an 8-man exploration mission. The shielded volume was designed large enough and adequately equipped to serve as a storm cellar during solar-flare disturbances of a week or more duration. The powered-flight station, plus an unshielded supply-storage and navigation room, were made separable from the basic vehicle as an emergency escape vehicle. The escape rockets and post-escape maneuver propellant (enough for  $\sim 2,000$  ft/sec  $\Delta V$ ) partially serve a dual purpose as shielding.

The 8-man personnel accommodations are shown in Fig. 3.5. Two-man staterooms are provided for sleeping and periods of privacy, with several work, laboratory, and recreation areas. Pressure floors and airlocks provide for three pressure-tight volumes if required. The total volume of continually pressurized personnel accommodations, excluding passageways, is just over  $200 \text{ m}^3$  ( $7,100 \text{ ft}^3$ ) or approximately  $25 \text{ m}^3$  ( $880 \text{ ft}^3$ ) per man. This volume does not include the relatively large (6 m high by 3.2 m diam.) repair bay/spares storage room provided in the central payload support spine below. It will be noted that the personnel accommodations are shown "upside down" in this and subsequent figures, owing to the present plan for obtaining artificial gravity, during coast periods, by rotating the entire vehicle at about 4 rpm.

The complete 10-m exploration vehicle is shown in Fig. 3.6. The personnel accommodations and the shielded powered-flight station-escape vehicle are seen atop a central payload spine that connects them and provides passage to the propulsion module. Around the spine near the propulsion module are numerous propellant magazines, each of which carry 90 pulse units. These are in addition to the 900 carried within the module body. Empty propellant magazines are jettisoned during coast periods. Above the propellant magazines are canisters that carry destination payload for use in the vicinity of the target planet, for descent to the surface, or for other purposes. An 8-man earth reentry vehicle, with a chemical maneuver stage capable of  $\sim 1,000$  ft/sec  $\Delta V$ , is carried as shown for missions requiring its use at earth return.

A 20-m configuration, as arranged and loaded for a high-velocity ( $\sim 200,000$  ft/sec) exploration mission, is shown in Fig. 3.7, with the 10-m configuration shown for reference. The configuration of the larger vehicle is similar to the 10-m design. It is designed for a personnel complement of 20, with  $29.5 \text{ m}^3$  ( $1,040 \text{ ft}^3$ ) of normally pressurized volume per man. The 20-m vehicle, as will be seen in the section on performance, is considered for very large or fast missions to Mars or for explorations of a Jovian moon.

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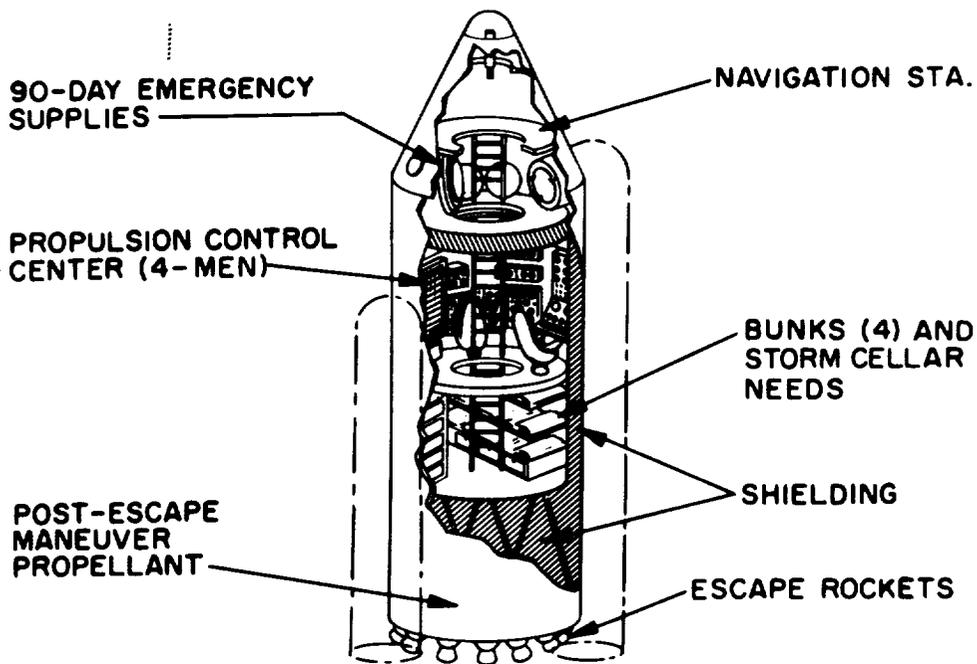


Fig. 3.4--Powered flight station-escape vehicle for 8-man exploration missions with 10-m configurations

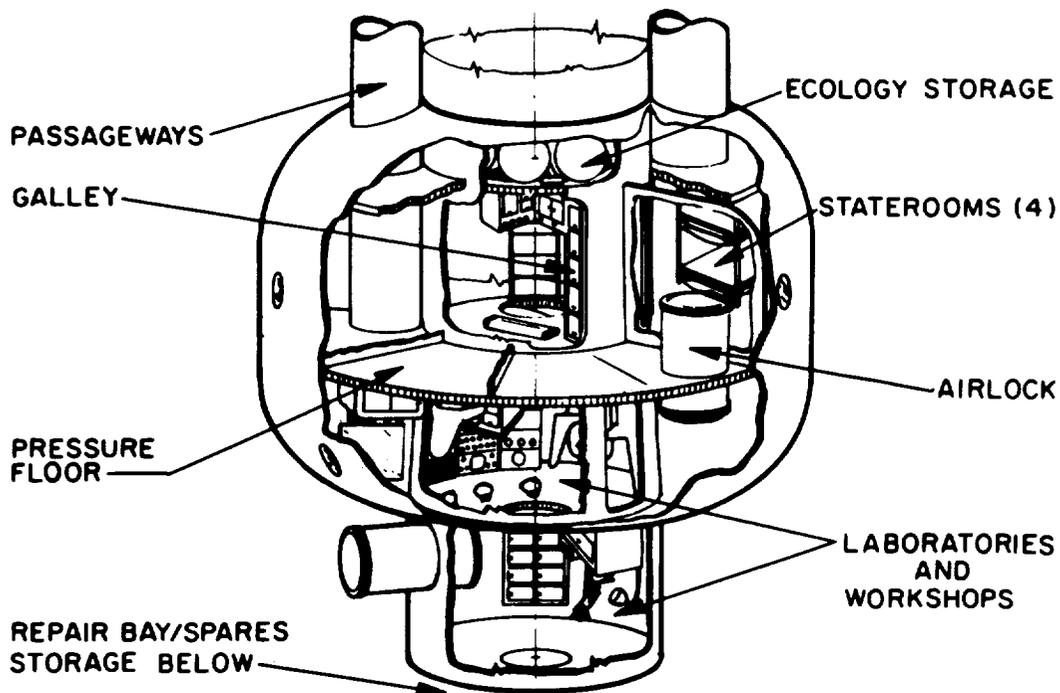


Fig. 3.5--Exploration-mission personnel accommodations for an 8-man complement

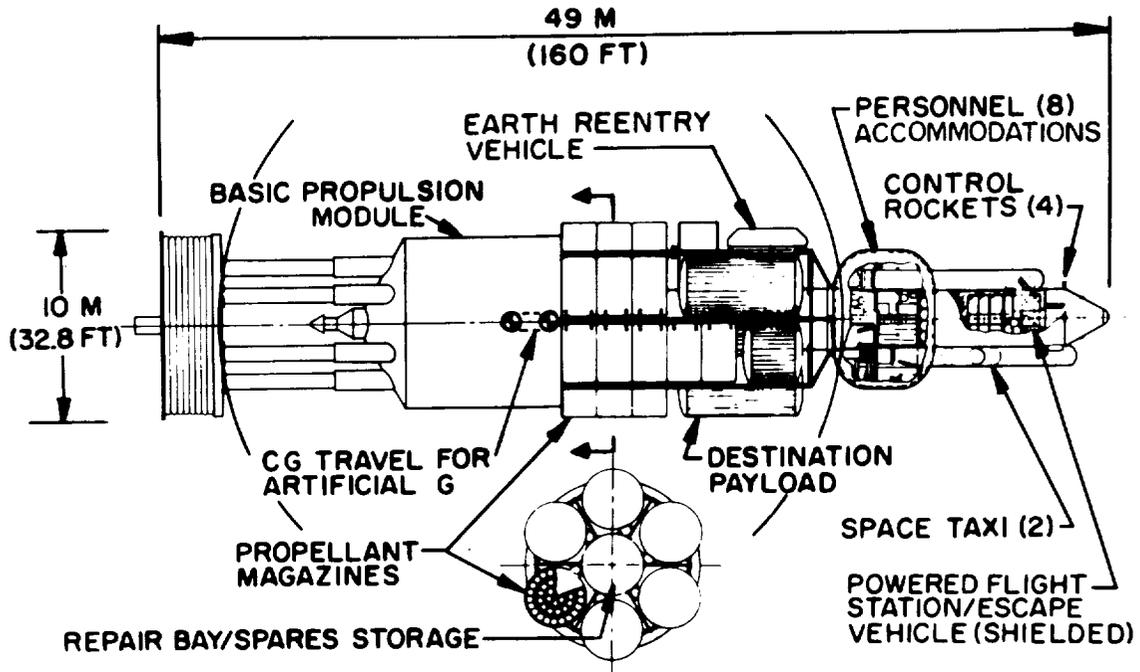


Fig. 3.6--Basic 10-m exploration configuration

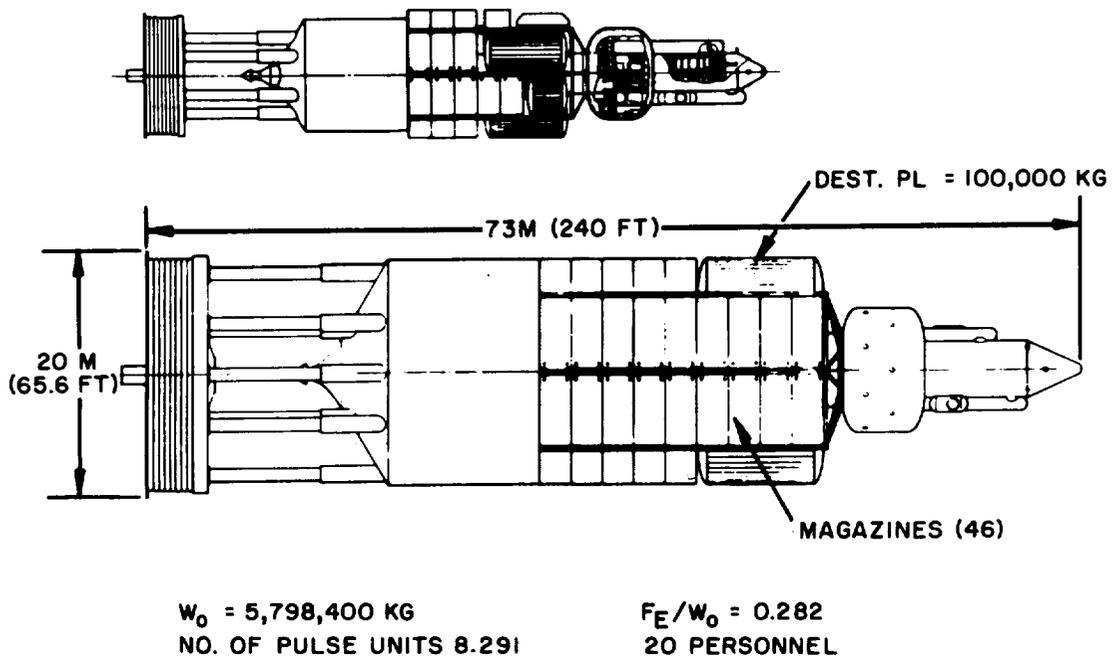
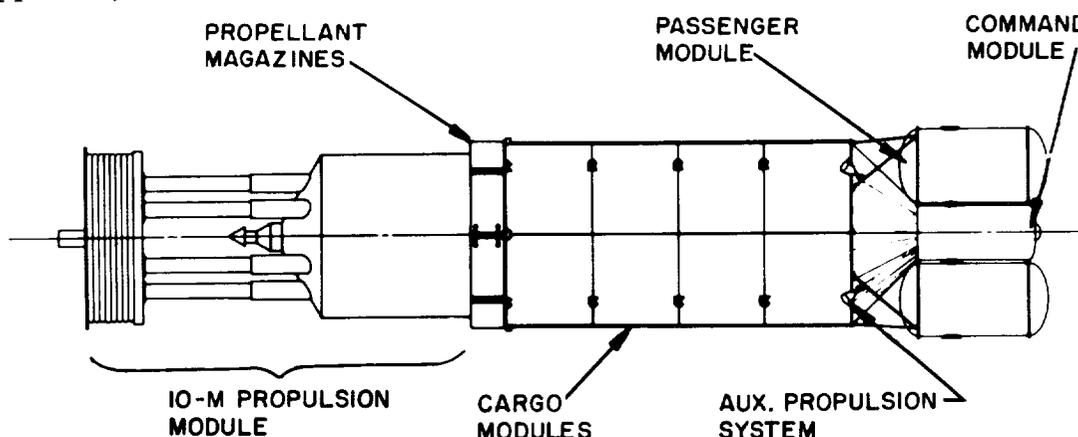


Fig. 3.7-- The 20-m exploration configuration, with the 10-m Mars exploration system shown for comparison

### 3.3. LUNAR VEHICLES

Vehicles using the 10-m propulsion module were designed for reusable lunar ferry systems and for one-way expendable logistic delivery systems. Several vehicle variations and different operational modes were considered. Since the 10-m designs provide as much lunar transportation capability as is presently of interest to planners, the 20-m modules were not considered for lunar systems.

The vehicle concept adopted for the earth-orbit-to-lunar-orbit ferry operation is shown in Fig. 3.8. Using an assumed payload density of  $320 \text{ kg/m}^3$  ( $20 \text{ lb/ft}^3$ ), the cargo modules were sized for a total mass of 100,000 kg (220,000 lb) and a 10 m diam to be compatible with Saturn V orbit delivery capability. The command module is a shielded compartment with two levels, an upper flight control station for a crew of three and a lower section for added crew accommodations. The lower section also provides shielded protection for the passengers during powered flight, transit through the earth's radiation belts, and solar flare encounters. The reference design shown accommodates 20 passengers in two modules; each module has its own power and environmental and life-support systems.



$$F_E = 3.5 \times 10^6 \text{N}$$

Fig. 3.8--Earth-orbit-to-lunar-orbit ferry vehicle concept

The design concept for the earth-orbit-to-lunar-surface ferry vehicle is shown in Fig. 3.9. This system is similar to the orbit-to-orbit ferry except that landing gear and a chemical ( $\text{O}_2/\text{H}_2$ ) landing propulsion module are provided. Both the landing struts and chemical-rocket thrust chambers are retractable to avoid impingement loads during nuclear-pulse operation.

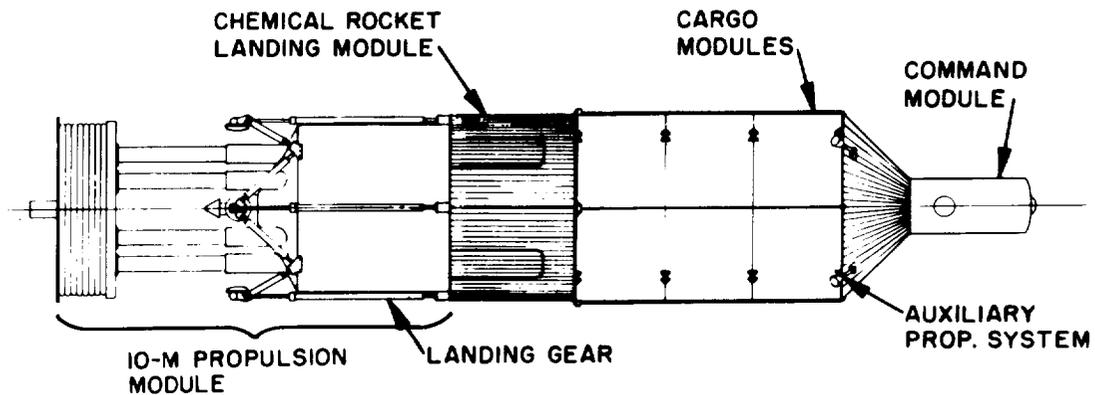


Fig. 3.9--Earth-orbit-to-lunar-surface ferry concept

Both solid-propellant and liquid-propellant booster stages (lofters) were included in the conceptual designs of unmanned earth-launched lunar logistic systems, as illustrated in Fig. 3.10. In these systems, the nuclear-pulse-propulsion module, as well as the chemical rocket stages, are expendable. In configurations IA and IB, a cluster of six solid-propellant motors accelerates the vehicle to just above the atmosphere prior to nuclear-pulse initiation. A thrust-to-weight ratio of 1.25 was used for the nuclear-pulse vehicle after staging the solid-propellant lofters. In configurations IIA and IIB, the Saturn S-1C is the booster stage. For these systems, a thrust-to-weight ratio of 1.0 was used for the nuclear-pulse vehicle after staging because of the higher staging velocity of the S-1C. With these loading conditions, an ideal velocity increment of 4,330 m/sec (14,200 ft/sec) is indicated for the S-1C.

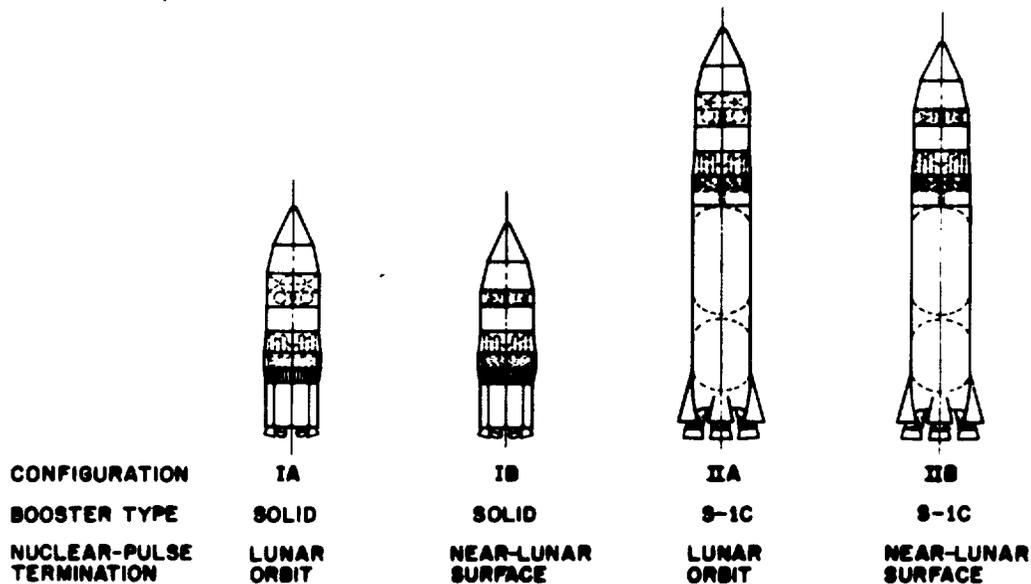


Fig. 3.10--Earth-launched lunar logistic systems

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The nuclear-pulse module is used for propulsion from booster separation to a lunar orbit for configurations IA and IIA and to near the lunar surface for configurations IB and IIB. Chemical-rocket stages ( $O_2/H_2$ ) are used, after staging the nuclear-pulse module, for the lunar descent from orbit (configurations IA and IIA) and/or for the lunar landing (all configurations).

An unmanned earth-orbit-to-lunar-surface logistic system was also conceptually designed as illustrated in Fig. 3.11. The expendable nuclear-pulse module is staged at near-lunar surface for this design and an  $O_2/H_2$  stage provides the last 2,000 ft/sec descent velocity increment plus allowances for landing maneuvers. Cargo modules, identical to those for the lunar ferry, are located forward of the chemical landing stage.

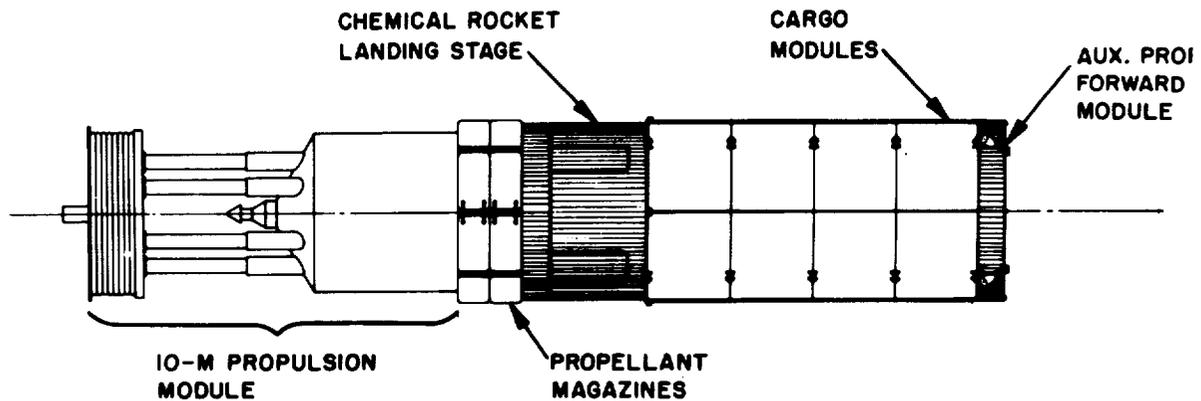


Fig. 3.11--Orbit-launched lunar logistic vehicle

### 3.4. SATURN V SYSTEM COMPATIBILITY

The two-stage Saturn V (S-1C and S-II stages) was considered the appropriate earth-launch vehicle (ELV) for almost all operational situations involving orbital deliveries. Larger post-Saturn ELV's were considered only for some missions involving the 20-m designs. The 10-m propulsion module, its operational payload or logistic cargo modules, and its propellant magazines were all conceptually designed to fit the desired payload envelope and weight limitations of Saturn V. Neglecting the more detailed considerations of dynamic control requirements and bending moments, all three appear to be fully compatible with the ELV. Figure 3.12 shows the three classes of boost-to-orbit payloads in launch position on the two-stage Saturn V. It also shows, at the right, a complete 10-m exploration vehicle atop the S-1C stage for a self-boost-to-orbit operation after being boosted well above the atmosphere by the S-1C (as discussed for the earth-launched lunar logistic vehicle of Fig. 3.10).

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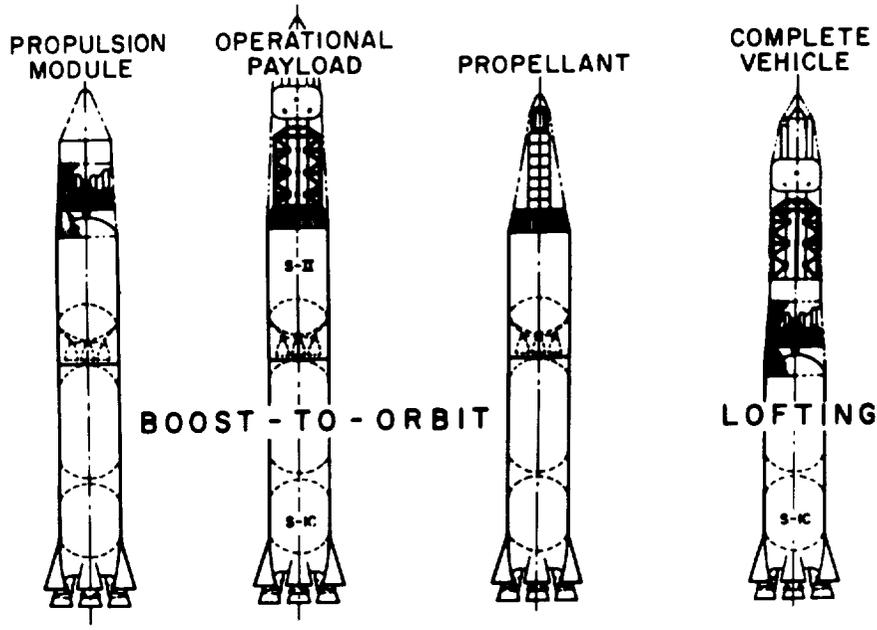
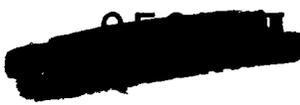


Fig. 3.12--Saturn V with three exploration mission payloads for orbital rendezvous and the S-1C stage as a complete-vehicle lofter

Other chemically propelled "lofters" were considered in a broad range of sizes for Mode I and II operations. In such operations relatively low-performance chemical rockets are required to loft a nuclear-pulse vehicle above the atmosphere and to a velocity of a few thousand feet per second prior to nuclear-pulse initiation.

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#### 4. PERFORMANCE AND OPERATING COSTS

The performance capability of the smaller (e. g., 10-m-diam) propulsion modules was shown during the parametric study phase to be quite adequate for most of the exploration missions to be investigated and to be more-than-adequate for most lunar missions. The cost-effectiveness of the smaller modules suffered in comparison to that for the larger ones because of the combination of increasing  $I_{sp}$  and decreasing unit cost of propellant with increasing size, but was found still favorable in an overall system sense or in comparison with other systems.

##### 4.1. PARAMETRIC STUDY INDICATIONS

The results of the parametric study conducted in the first phase of this study program indicated a pronounced performance benefit for the orbital start-up and reduced thrust-to-weight ratio (F/W) mode of operation (Mode III) in performing high-energy missions. These benefits are seen in Fig. 4.1, where the single-stage payload-delivery capability of typical nuclear-pulse vehicles is shown for initial F/W = 1.25 and 0.25. At an initial F/W = 0.25, mission velocities ( $\Delta V$ 's) of over 100,000 ft/sec are attainable with the 10-m module carrying appreciable payloads and using but one stage. Similarly, with thrusts in the range of the single-stage 20-m module at the lower F/W's, 200,000-ft/sec mission velocities are attainable; the more modest Jupiter missions and the very fast Mars mission requirements are in this category.

These parametric performance data, plus the compatibility of the 10-m module with Saturn V and of the 20-m module with a nominal million-pound-to-orbit post-Saturn ELV, decided the vehicle sizes for the specific-conceptual-system study phase.

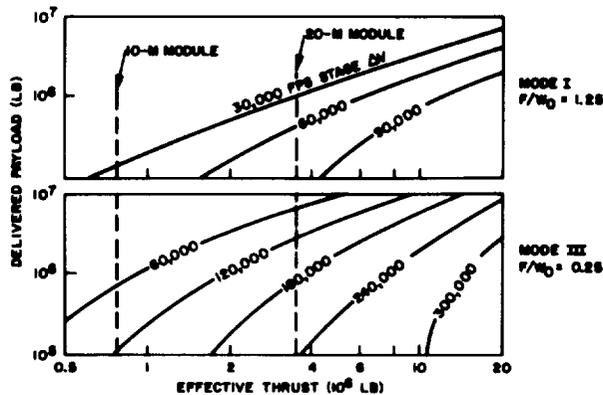
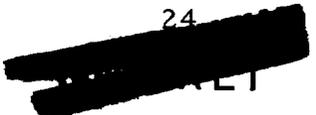


Fig. 4.1--Performance benefits of lower thrust-to-weight-ratio mode of operation



#### 4.2. EXPLORATION MISSIONS

The 10-m exploration vehicles were exercised in performing a wide variety of planetary explorations, using the configuration arrangements of Section 3 in fulfilling the mission requirements summarized in Section 2. The  $\Delta V$  requirements for Mars and Venus ranged roughly from 50,000 to 100,000 ft/sec. Throughout this range the same basic configuration was used. Different amounts of propellant are required, of course, which results in more propellant magazines being required for the more difficult missions. Such propellant loading differences are shown in Fig. 4.2. The four configurations represent earth-departure loading conditions for the four earth-approach conditions computed for 1975, 450-day Mars missions (see Table 2.4). The earth-departure weight is shown under each vehicle; a summary weight statement for each is given in Table 4.1.

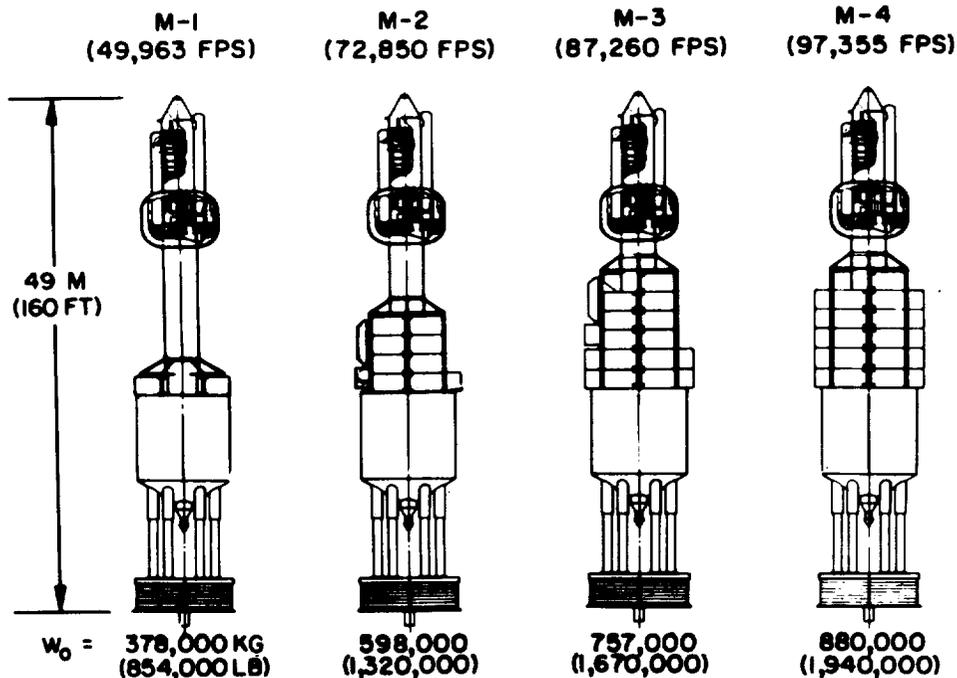


Fig. 4.2--Vehicle loading differences for different mission velocities

The effect of departure year on Mars mission  $\Delta V$  requirements is considerably less than the effect of the earth-approach condition. The range of mission velocities typical for a cycle of favorable-to-unfavorable years (Table 2.5) resulted in modest departure weight variations from those given for 1975. The most difficult year (1977) added 8 to 10 percent to departure weight; the most favorable (1973, 1984) decreased departure weight by 14 to 19 percent.



Table 4.1

WEIGHT SUMMARY FOR MARS MISSIONS FOR DIFFERENT MISSION VELOCITIES  
 Eight-Man, 450-day Missions  
 (In kilograms)

Component	M-1, 49,963 fps	M-2, 72,850 fps	M-3, 87,260 fps	M-4, 97,355 fps
Operational payload	73,000	80,000	77,000	73,000
In-transit payload	250	250	250	250
Destination payload	<u>750</u>	<u>750</u>	<u>750</u>	<u>750</u>
Total payload	74,000	81,000	78,000	74,000
Propulsion module	91,400	92,430	93,000	93,300
Propellant magazines	4,578	13,734	21,350	25,950
Guidance and start-up fluids	4,222	6,570	8,000	9,300
Propellant	<u>212,800</u>	<u>405,266</u>	<u>556,650</u>	<u>677,450</u>
Earth-orbit Departure Weight	387,000	599,000	757,000	880,000
Earth-orbit Departure Weight, lb	854,000	1,320,000	1,670,000	1,940,000

Note: Higher operational payloads for missions M-2 and M-3 due to required earth reentry vehicle and maneuver stage. Propulsion module weight differences are due to external propellant-magazine support columns; basic modules are identical.

A third factor significantly affecting mission  $\Delta V$  requirements is mission duration. Typical  $\Delta V$  requirements for a 300-day and 150-day Mars round-trip mission were given in Table 2.6. The 300-day mission ( $\Delta V = 103,300$  ft/sec) is still within the one-stage capability of the 10-m nuclear-pulse vehicle; a vehicle weight summary for this mission is shown in Table 4.2.

Table 4.2

WEIGHT SUMMARY FOR 300-DAY MARS MISSION  
 (In kilograms)

Payload	
Operational . . . . .	72,800
In-transit . . . . .	250
Destination . . . . .	<u>750</u>
Total payload . . . . .	73,800
Propulsion module . . . . .	95,380
Propellant magazines . . . . .	38,150
Guidance and start-up fluids . . . . .	14,940
Propellant . . . . .	<u>922,730</u>
Earth-orbit departure weight . . . . .	<u>1,145,000</u>
Earth-orbit departure weight, lb . . . . .	<u><u>2,522,000</u></u>



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Exploration-mission performance of the 10-m configurations was computed for six different categories of payload, as previously summarized in Table 2.8. Five of the payload configurations, on a vehicle loaded to perform the M-2, 72,800 ft/sec Mars mission, are shown in Fig. 4.3. The configuration on the left has partially expendable (and 2,000 kg lighter) personnel accommodations, ecological system, and supply storage compartments in the form of four cylindrical modules. The other configurations shown are self-explanatory; they simply accommodate different numbers of mission personnel and different destination payloads. A weight summary of the five configurations is given in Table 4.3.

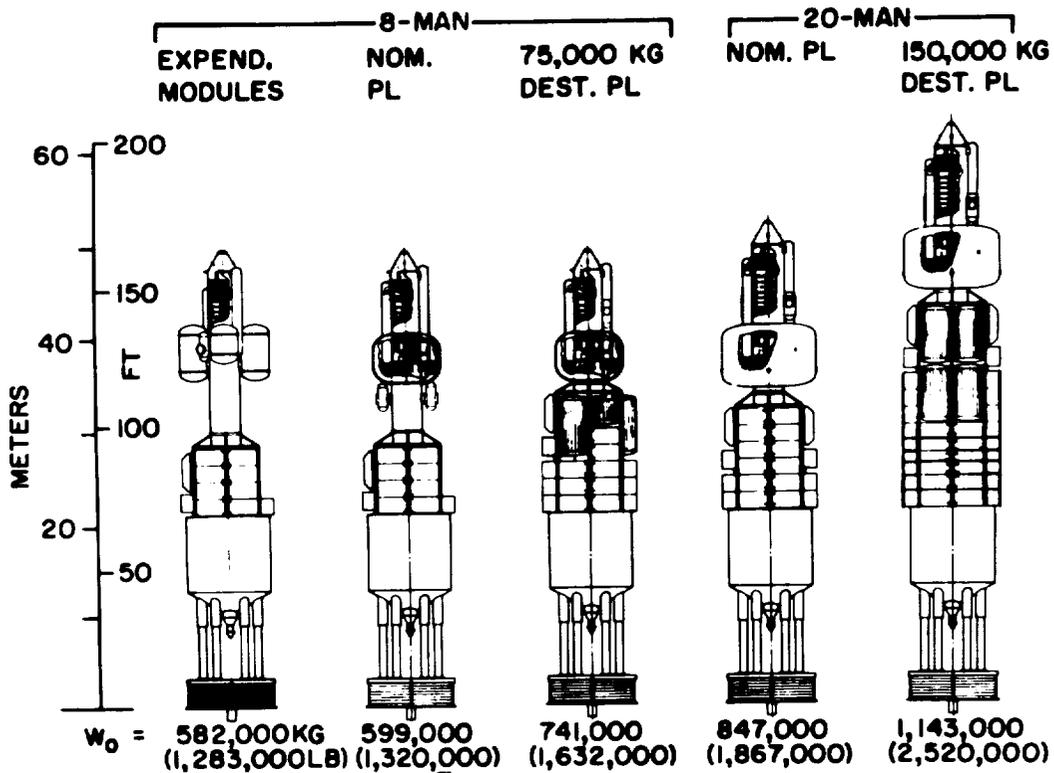


Fig. 4.3--Vehicle configurations and loading differences for various mission payloads

The performance data summarized above show that the 10-m propulsion module, as defined herein, has a very substantial capability in performing Mars explorations or the somewhat easier Venus missions. The capability appears such as to permit several forms of system redundancy or margin for error (as will be further indicated by the data on system sensitivities). For example, six or eight personnel do not constitute an upper limit--scientists as well as multiskilled astronauts can make the trip; it is not necessary to rely on a faster-than-Apollo reentry at the conclusion of the trip; extra shielding can be carried if required;

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Table 4.3

WEIGHT SUMMARY FOR MARS MISSIONS WITH DIFFERENT PAYLOAD CONFIGURATIONS  
(in kilograms)

Component	Eight-man Missions			Twenty-man Missions	
	Expendable Modules	Nominal (750 kg) Destination Payload	75,000 kg Destination Payload	Nominal (750 kg) Destination Payload	150,000 kg Destination Payload
Payload					
Operational	77,000	80,000	80,000	150,000	150,000
In-transit	250	250	1,000	250	2,000
Destination	750	750	75,000	750	150,000
Total Payload Weight	78,000	81,000	156,000	151,000	302,000
Propulsion Module	92,400	92,430	93,800	93,270	97,350
Propellant Magazines	12,210	13,735	16,785	21,350	27,470
Guidance and Start-up Fluids	6,440	6,570	7,235	8,245	9,580
Propellant	392,950	405,265	467,180	573,135	706,600
Earth Orbit Departure Weight	582,000	599,000	741,000	847,000	1,143,000
Earth Orbit Departure Weight, lb	1,283,000	1,320,000	1,632,000	1,687,000	2,520,000

Note: These data are for M-2 Mars missions having total  $\Delta V = 72,850$  fps. Propulsion-module weight differences are due to external propellant-magazine support columns; basic modules are identical.

and, if subsystem or payload weight requirements increase during the development period, these too can be accommodated.

Mission direct operating costs (DOC's) were estimated for the more significant missions investigated. Before presenting the exploration mission cost summaries, it is appropriate to briefly summarize the probability-of-success (or "reliability") inputs to the DOC, since they have the effect of increasing a "perfect system" DOC by a factor of 1.5 to 2.0, for example, for exploration systems, dependent on the number of ELV deliveries to an orbital rendezvous.

The delivery-system reliability assumptions are shown in Table 4.4, which includes not only the Saturn ELV for orbital deliveries, but also assumptions for lunar logistic systems employing a nuclear-pulse stage. For exploration systems which require a departure weight build-up by means of orbital rendezvous, the ELV delivery reliability assumption (0.85) is combined with an assumed probability of successful mating (0.97), of successfully loading additional payload (0.99), of successfully loading each delivery of propellant (0.98), and a desired probability of mission readiness ( $\geq 0.75$ ). The resulting required number of launches and the required procurement of mission hardware, propellant, etc., for a typical mission are shown in Table 4.5. This mission requires a total of six successful launches, but nine must be provided to assure the desired probability of readiness. Similarly, 1.5 times the required propellant, magazines, etc., must be procured and ready to launch if necessary, and a spare propulsion module and an extra operational payload assembly must be procured and standing by.

Table 4. 4

**DELIVERY-SYSTEM RELIABILITY ASSUMPTIONS  
FOR SYSTEM COSTING**

Delivery System	Probability of Successful Delivery			
	Stage 1	Stage 2	Stage 3	Over-all
Saturn V (S-1C plus S-II)	0.94	0.90	----	0.85
Nuclear-pulse Mode II operation (S-1C and 10-m module)	0.94	0.85	----	0.80
Earth-orbit-to-lunar-surface Logistic system (10-m module plus chemical landing stage)	0.85	0.88	----	0.75
Earth-surface-to-lunar-surface logistic system (S-1C plus 10-m module plus chemical landing stage)	0.94	0.85	0.88	0.70

Table 4. 5

**TYPICAL PROBABILITY EFFECT ON NUMBER OF LAUNCHES  
AND PROCUREMENT REQUIRED FOR EXPLORATION MISSION**

Launch-vehicle Payload	Successful Deliveries Required	Probable Number of Launches Required	Probability Procurement Factor
Propulsion module	1	1 or 2	2
Operational payload	1	2 or 1	2
Propellant and miscellaneous payload mix	4	6	(6/4 =) 1.5
<b>Total Launches Required</b>		9	

The total DOC for a number of 10-m-vehicle Mars explorations, with a breakdown showing the major cost elements, is shown in Fig. 4. 4. The differences shown illustrate the cost effect of varying total mission velocities (reflecting the different earth-approach conditions), of carrying destination payloads sufficient for planetary landings, and of increasing the personnel complement from 8 to 20 men. The ELV boost-to-orbit costs clearly predominate, with the cost of nuclear-pulse propellant being next in magnitude. One billion dollars is seen to be the approximate DOC for the more conservatively operated missions.

The 20-m configurations, as previously stated, were exercised in performing the larger Mars missions and an exploration mission to Callisto, a moon of Jupiter. Four of such configurations are shown in Fig. 4. 5. Their earth-orbit departure weights are indicated below the configurations. A weight summary for these vehicles and for vehicles to perform two more-difficult missions (a 150-day round-trip to Mars and a

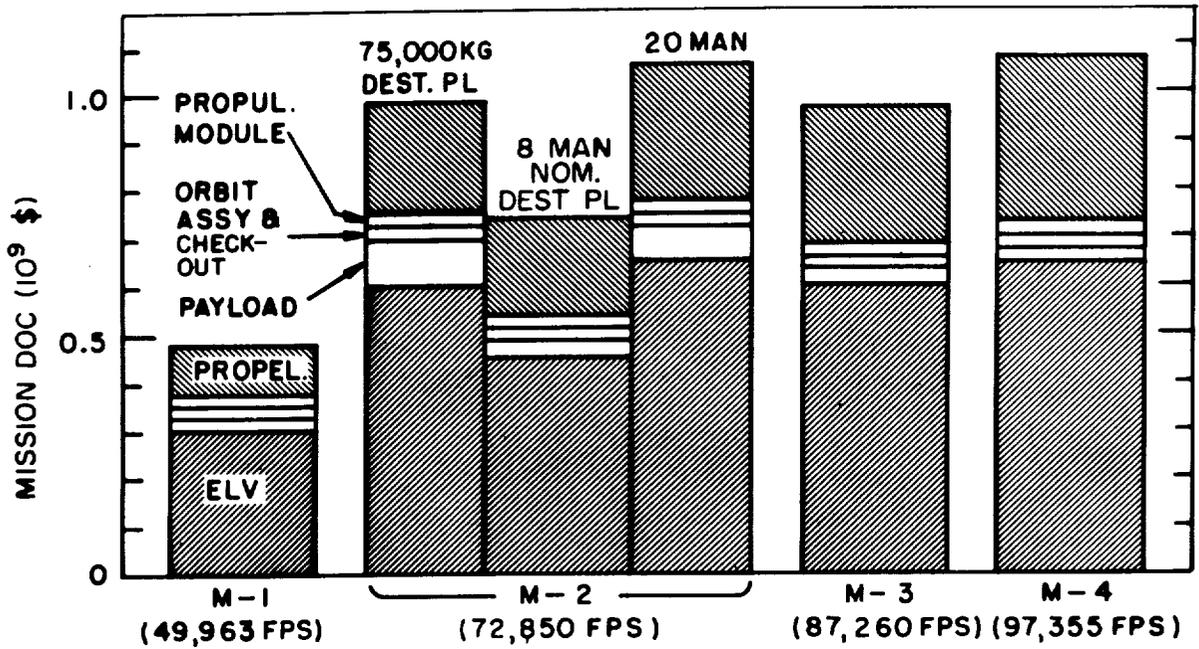


Fig. 4.4-- Direct-operating-cost components for different Mars exploration missions

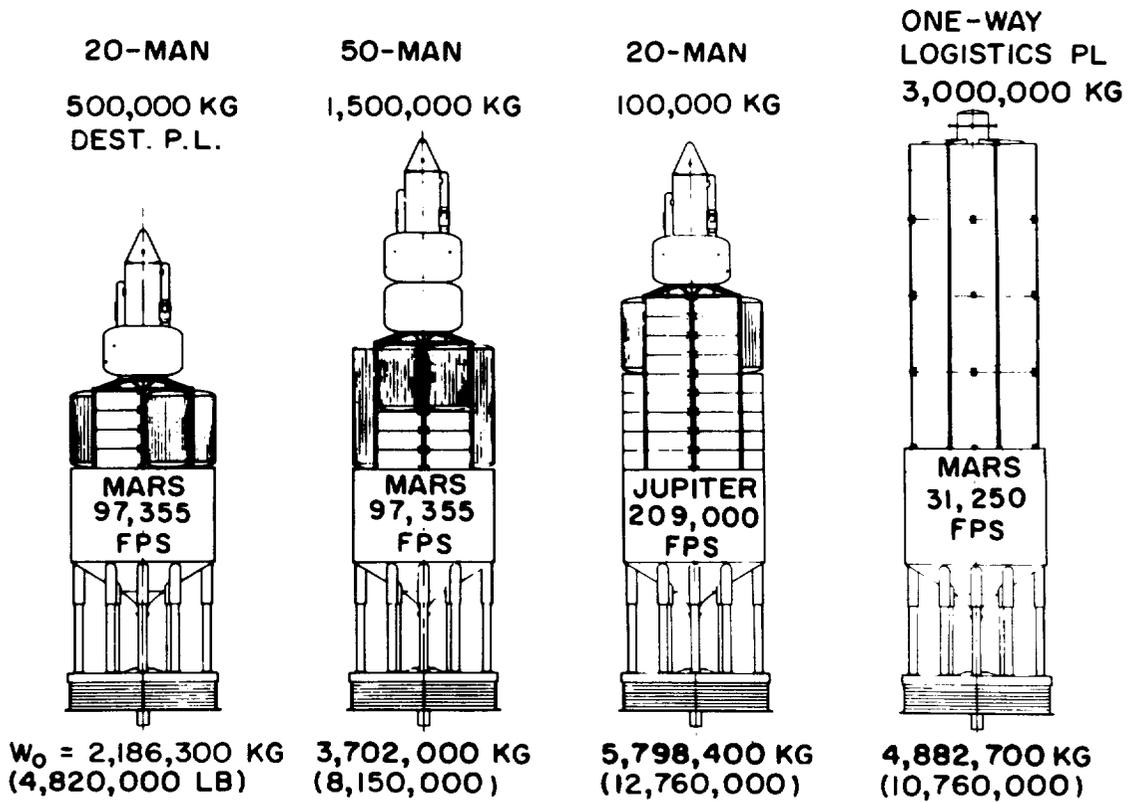


Fig. 4.5-- 20-m configuration and loading difference for three exploration missions and one logistics mission

Jupiter trip carrying 500,000 kg of destination payload) are shown in Table 4.6. Mission DOC's for the 20-m configurations varied from \$1.6 to \$4.0 billion if Saturn V was assumed for orbital deliveries, and from \$1.1 to \$2.3 billion if a postulated post-Saturn ELV was assumed.

Table 4.6

SUMMARY WEIGHT STATEMENT FOR 20-METER VEHICLES  
(In kilograms)

Components	Mars Missions				Jupiter Missions	
	450-day		150-day, 20-man, 208,800 fps	Logistic, One-way, 31,250 fps	910-day, 20-man, 209,000 fps	
	20-man, 97,355 fps	50-man, 97,355 fps			100,000 PL	500,000 PL
<b>Payloads</b>						
Operational	192,700	271,700	178,700	178,000	289,500	297,500
In-transit	1,300	1,300	1,300	1,300	2,500	2,500
Destination	500,000	1,430,000	500,000	3,000,000	100,000	500,000
<b>Total payload</b>	<b>694,000</b>	<b>1,703,000</b>	<b>680,000</b>	<b>3,178,000</b>	<b>392,000</b>	<b>800,000</b>
Propulsion module	363,500	365,400	383,100	360,000	376,000	380,400
Propellant magazines	35,500	53,400	267,000	44,500	204,000	267,000
Guidance and start-up fluids	15,500	20,400	64,240	15,250	53,000	60,700
Propellant	1,077,500	1,560,800	5,944,000	1,284,950	4,773,000	6,076,430
<b>Earth-orbit-departure weight</b>	<b>2,186,000</b>	<b>3,702,000</b>	<b>7,338,340</b>	<b>4,882,700</b>	<b>5,798,000</b>	<b>7,584,530</b>
<b>Earth-orbit-departure weight, lb</b>	<b>4,815,000</b>	<b>8,150,000</b>	<b>16,200,000</b>	<b>10,760,000</b>	<b>12,786,000</b>	<b>16,750,000</b>

#### 4.3. LUNAR MISSIONS

Only the 10-m propulsion modules were used for lunar ferry and logistic systems; the lunar configurations were described in Section 3 and their mission velocity requirements summarized in Section 2.

The performance of the earth-orbit-to-lunar-orbit ferry system (see Fig. 3.8 for configuration) is summarized in Fig. 4.6. The net payload mass delivered to the lunar surface is plotted as a function of the vehicle orbit-launch mass. The top curve indicates a cargo-only delivery, with the effect of the number of passengers shown by the lower curves. Good payload mass fractions (and hence good cost-effectiveness, as will be seen) occur at orbit-launch masses of ~700,000 kg and higher. The ferry system was considered primarily as a transport system for lunar-base support. Ideally, it should be capable of transporting an unbalanced cargo-personnel mix during the base buildup phase (structures and equipment delivery) and a balanced cargo-personnel mix during steady-state operations. The figure shows a locus of balanced cargo--personnel payloads based on a personnel stay time of six months and an annual requirement of 1,816 kg (4,000 lb) of supporting cargo per man.

Total direct cost of operating the ferry, with a breakdown of its major DOC elements, is shown in Fig. 4.7 as a function of the number of passengers (or number of lunar-base personnel, assuming two trips

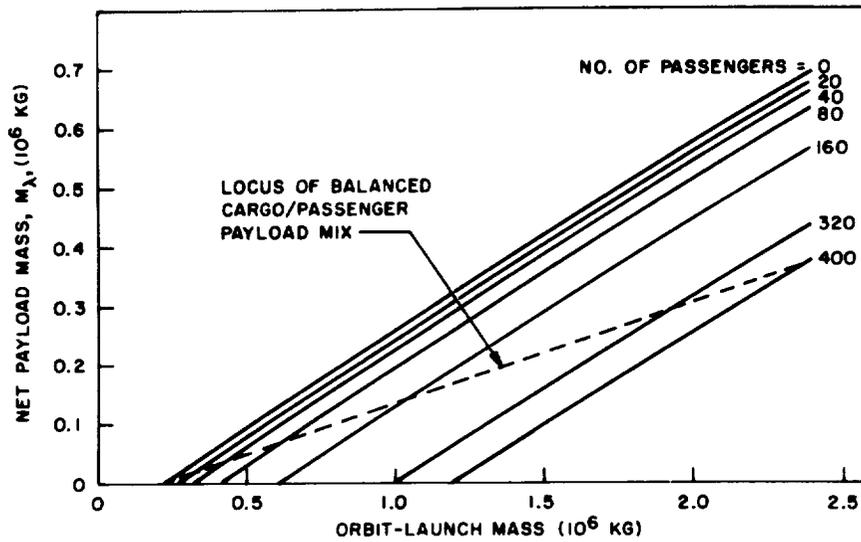


Fig. 4.6--Passenger transportation capability of earth-orbit-to-lunar-orbit ferry system

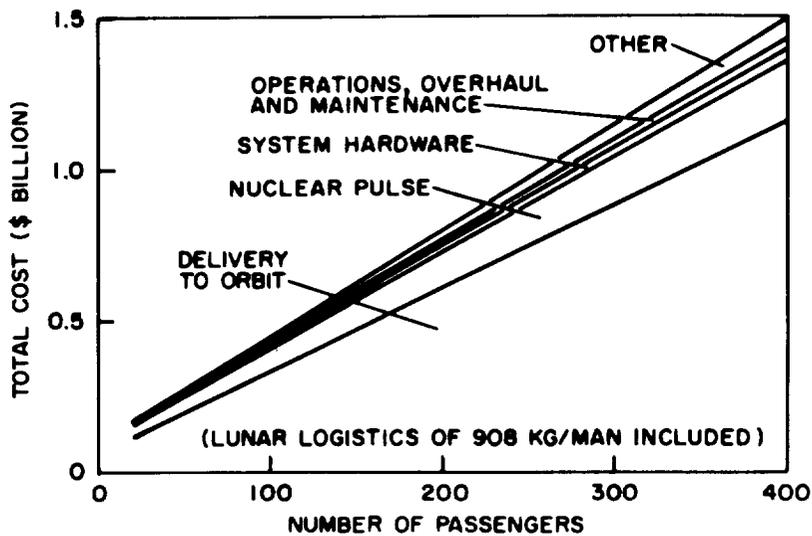


Fig. 4.7--Lunar-ferry direct operating cost for balanced passenger-cargo payload

per year). The costs are seen to be almost linear with the number of passengers. Delivery to orbit dominates the DOC even more than it did for exploration missions.

The lunar payload delivery capability of the earth-orbit-to-lunar-surface configuration (see Fig. 3.9) is shown in Fig. 4.8. This system's payload delivery capability appears better than the orbit-to-orbit system only at launch masses over  $10^6$ kg. Its more massive and complex lunar

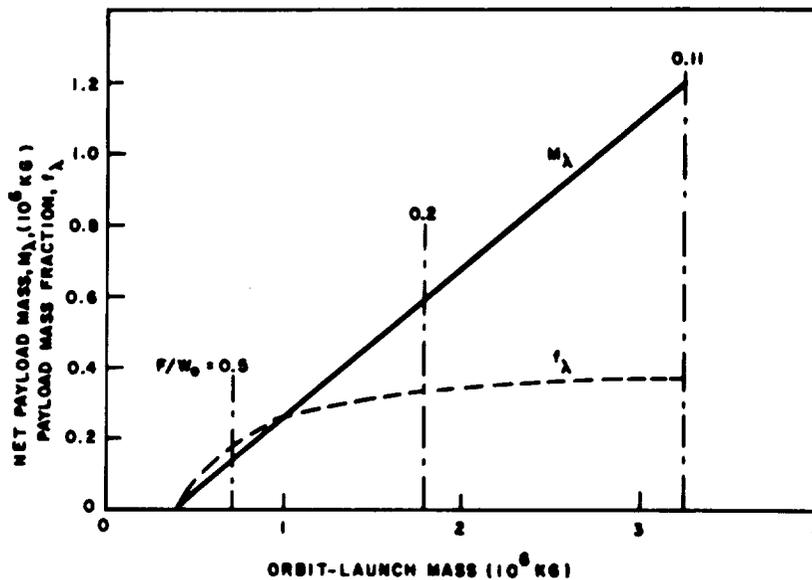


Fig. 4. 8--Performance of earth-orbit-to-lunar-surface ferry system

landing and take-off (compared to the chemical rocket orbit-to-surface shuttles of the orbit-to-orbit ferry system) does not therefore seem justified.

The performance of the earth-launched lunar logistic systems (see Fig. 3. 10 for configurations) is summarized in Table 4. 7. In this mode of using the 10-m module for lunar payload deliveries, the smaller payload capability is more compatible with present-day thinking about lunar operations. The high, first-stage,  $\Delta V$  of the S-1C boosted systems (the II's) results in a net payload more than twice that for the minimum  $\Delta V$ , solid-propellant-lofted system. For either class of vehicle, the "B" systems, with termination of the nuclear-pulse stage at near the lunar surface, provided approximately 50 percent more payload than the "A" systems, with termination in lunar orbit.

Table 4. 7

PERFORMANCE OF EARTH-LAUNCHED LUNAR LOGISTIC VEHICLES

Config-uration	Net Payload		Launch Mass		Net Payload Mass Fraction
	(kg)	(lb)	(kg)	(lb)	
I A	21, 538	47, 491	804, 900	1, 774, 900	0. 0268
I B	25, 026	55, 182	804, 900	1, 774, 900	0. 0311
II A	51, 027	112, 515	2, 614, 700	5, 765, 300	0. 0195
II B	67, 461	148, 752	2, 614, 700	5, 765, 300	0. 0254

The DOC for the four earth-launched lunar logistic systems is shown in Table 4.8. The S-1C boosted system is seen to have significantly better cost-effectiveness than the solid-propellant-lofted systems. (It should be mentioned that neither system was optimized for the best combination of chemical and nuclear-pulse  $\Delta V$  contribution or for initial stage F/W's. Work subsequent to this study indicates significant cost-effectiveness improvements are thereby available for these systems and to a lesser extent for the other lunar systems.)

Table 4.8

**DIRECT OPERATING COSTS FOR EARTH-LAUNCHED LUNAR LOGISTIC VEHICLE**  
(Costs per launch in thousands of dollars)

Cost Element	Solid-propellant First Stage		Saturn S-1C First Stage	
	IA Lunar-orbit Staging	IB Near-surface Staging	IIA Lunar-orbit Staging	IIB Near-surface Staging
First stage	10,020	10,020	20,250	20,250
Nuclear-pulse propulsion module	12,000	12,000	12,000	12,000
Nuclear-pulse propellant	48,000	51,500	49,500	54,300
Lunar landing stage	6,720	1,740	13,080	3,900
Lunar-landing-stage propellants	21	8	49	20
Instrumentation and guidance	4,000	4,000	4,000	4,000
Systems integration	6,000	6,000	6,750	6,750
Launch operations	5,000	5,000	5,000	5,000
<b>Total direct operating costs</b>	<b>91,761</b>	<b>90,268</b>	<b>110,629</b>	<b>106,220</b>
Net payload, kg	21,538	25,026	51,027	67,461
Unit DOC, \$/kg	4,260	3,600	2,170	1,580
Unit DOC at 75% reliability, \$/kg	5,680	4,800	2,890	2,110

The payload capability of the orbit-launched lunar logistic system (configuration shown in Fig. 3.11) is shown in Fig. 4.9. The net payload to the lunar surface and the net payload mass fraction are given as functions of orbit-launch mass. The maximum payload mass fraction is achieved at an  $F/W \cong 0.2$ , but is quite good at  $F/W = 0.5$  or smaller.

These data on lunar transport and logistic systems show that the 10-m propulsion module of this study has a large payload or passenger capability at delivery costs one-half to one-fourth that of other systems known to have been studied. The payload capability for all except the earth-launched configurations, however, is very large by present standards; to utilize this capability most efficiently would require a very substantial lunar program. It is known, however, that more optimum nuclear-pulse-propulsion modules and more optimum chemical-nuclear-pulse staging combinations are feasible for lunar missions.

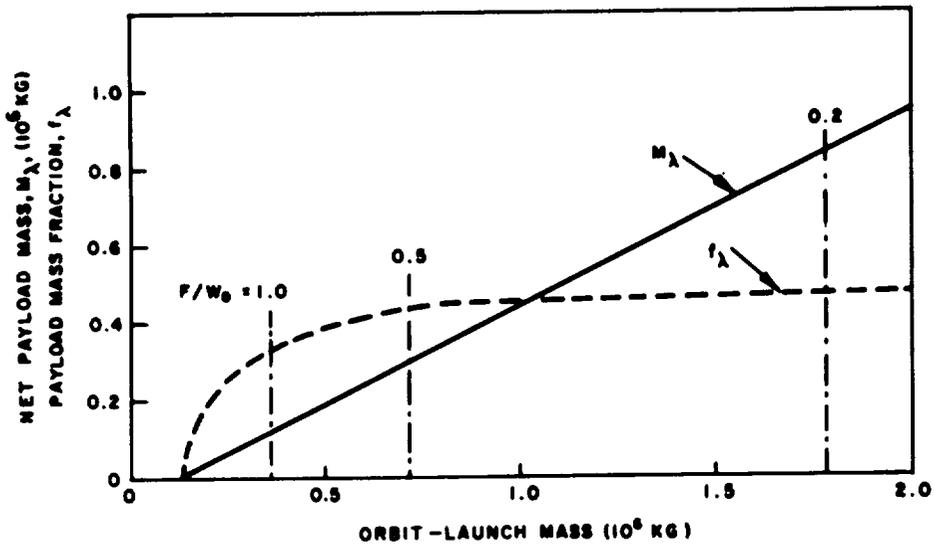


Fig. 4.9--Orbit-launched lunar-logistic-system performance

#### 4.4. PERFORMANCE AND COST SENSITIVITIES

In general, the sensitivity analyses show that the nuclear-pulse vehicle can tolerate more-than-typical developmental ups and downs in hardware, weight, and performance without such variations being critical to its presumed missions. The sensitivity data is briefly summarized here.

The performance factors that grossly affect mission performance, and therefore mission costs, are  $I_{sp}$  and the propulsion module weight or other inert weights. The  $I_{sp}$ , which for the 10-m module was considered 1,850 sec (but which subsequent study indicates may be nearer to 2,000 sec), was varied from 1,500 to 2,500 sec. The effect of such variation on the earth-orbit departure weight and on the mission DOC for a typical Mars exploration is shown in Fig. 4.10. It is seen that the higher  $I_{sp}$  now predicted will be beneficial, but also that considerable degradation can be tolerated for such missions. The lunar missions are considerably less sensitive to  $I_{sp}$  variations, as would be expected, because of their lower  $\Delta V$  requirements.

The sensitivity to propulsion-module dry weight was explored over a range from a 20 percent decrease (reduced 18,200 kg) to a 40 percent increase (up 36,400 kg). The effect on both orbit-departure weight and system DOC for a 72,850 ft/sec Mars exploration was essentially linear, permitting two useful generalizations. The effect of a change in non-expended inert weight (be it propulsion module, shielding, hardware, personnel, or other inerts) in departure weight to inert weight ratio

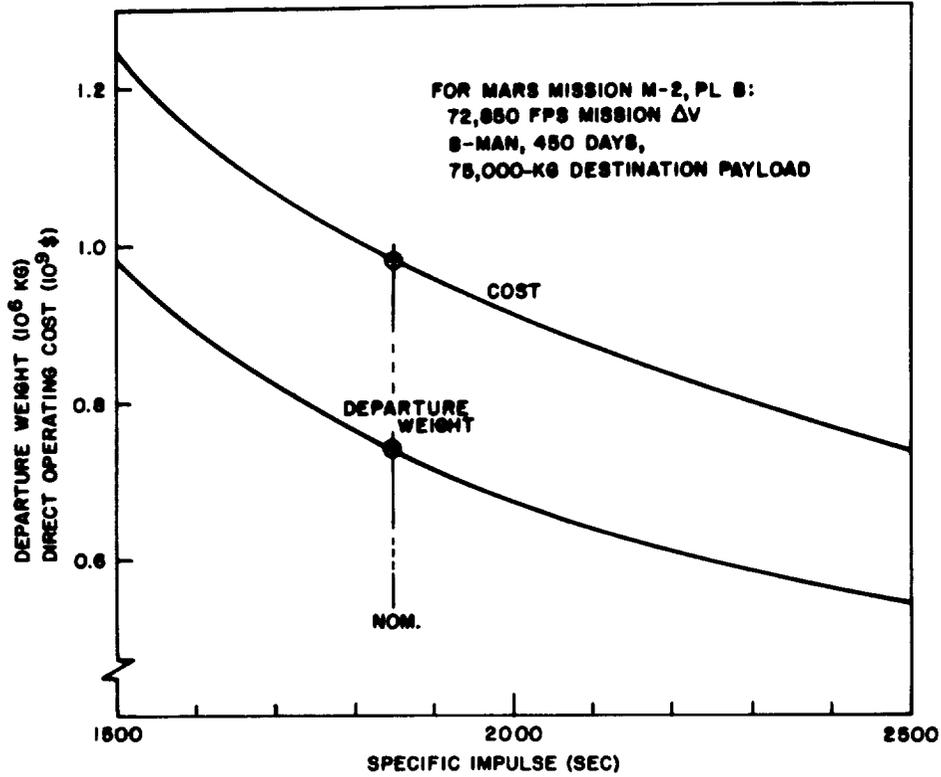


Fig. 4.10--Typical exploration-system sensitivity to specific impulse

( $\Delta W_o/\Delta W_i$ ) is  $\sim 3.7/1$ . Similarly, the operating cost to inert weight ratio ( $\Delta DOC/\Delta W_i$ ) is  $\sim \$3,300/\text{kg}$  ( $\sim \$1,500/\text{lb}$ ). Thus, for example, an added 1000 kg of inert weight increases departure weight by 3,700 kg and DOC by  $\$3.3 \times 10^6$ . These typical "growth factors" are quite low compared to most high-requirement transportation systems and reflect the considerable amount of reserve capability in the system.

#### 4.5. ADVANCED-VEHICLE POTENTIAL

Nuclear-pulse propulsion systems have a large performance growth potential over the systems exercised in this study. Because propulsion efficiencies of current designs are very low (a few percent) and means of improvement are known, potential improvements like factors of 5 to 10 are foreseen. Part of this improvement in performance potential ( $I_{sp}$ ) is coupled with increases in vehicle size, but much of it appears attainable in the smaller propulsion modules also.

It is expected that  $I_{sp}$ 's of 10,000 to 20,000 sec, at thrusts of  $10 \times 10^6$  lb to  $40 \times 10^6$  lb, could be available for second-generation systems. This prediction is predicated on several years of research and investigation beyond the development of the early modules described in this study. It is also based on the assumption that pusher ablation, as it

is now understood, is the dominant physical phenomenon limiting specific impulse.

Two advanced-version hypothetical vehicles, labeled A and B, were described for an investigation of their capability. Vehicle A is assumed to have an  $I_{sp}$  of 10,000 sec and a thrust of  $10 \times 10^6$  lb; B was assigned an  $I_{sp}$  of 20,000 sec and a thrust of  $40 \times 10^6$  lb. The dry vehicle F/W was assumed to be 3.0 in each case. Propellant costs were taken from those predicted to result from a redesign of the nuclear device especially for propulsion. For Mode I (self-boost to orbit) operations, the additional ground rule was the assumption of a near-earth-surface initiation of nuclear-pulse operation; thus lofter operating costs were modest but a fairly high F/W (1.25) was required. The resulting Mode I payload delivery capability and DOC per pound delivered is shown in Fig. 4.11. The larger vehicle, B, is seen to have significant payload capability at  $\Delta V$ 's in excess of 500,000 ft/sec; its DOC/lb is under \$300 until  $\Delta V$ 's reach 450,000 ft/sec. Quite large mission capabilities and relatively economical systems can therefore be expected if the predicted system potential is sought and realized.

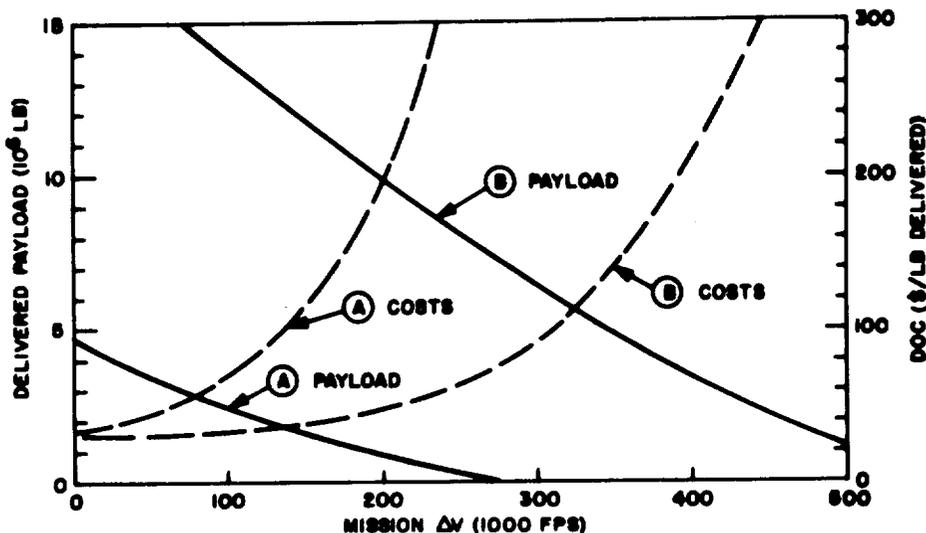


Fig. 4.11--Predicted performance and operating costs for advanced-version vehicles

## 5. OPERATIONAL CONSIDERATIONS

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

### 5.1. PULSE-CREATED NUCLEAR ENVIRONMENT

The calculated nuclear radiation levels due to a pulse-unit detonation are shown in Fig. 5.1 at various points of interest for the design of the 10-m vehicle systems. Personnel shielding (see Section 3) was calculated using earlier radiation estimates of higher values, making shielding weights now appear some 7 to 10 percent conservative. The radiation levels of equivalent points on larger, higher-thrust vehicles are lower because of attenuating masses and  $R^2$  effects, more than compensating for higher pulse-unit yields.

Pulse units for the 10-m vehicles generate a yield of approximately 1 KT per pulse. Earth-departure maneuvers for typical Mars missions (several hundred to over 1000 pulses) result in total yields of some 0.5 to 1.2 MT, producing a quantity of fission products that must be reckoned with. A number of countermeasures, however, can reduce fission-product trappage substantially from this amount. One somewhat helpful course is orbital start-up (Mode III) operations, which can provide reduction factors of 2 to 4. Improvements in the design of the nuclear devices (by reducing the fraction of total yield due to fission) might achieve reduction factors of  $10^2$  to  $10^3$ . Focussing the fission products and unburned fuel away from the earth (at a sacrifice in  $I_{sp}$  during the departure maneuver) is another possibility. Combinations of these measures can conceivably multiply the reduction factors. Finally, in the longer term, pure fusion devices would, by definition, eliminate fission products.

The creation of artificial radiation belts by electron injection is an important consideration in near-earth nuclear-pulse operations. Recent data obtained with artificial satellites indicate that the lifetime of trapped electrons is on the order of weeks on magnetic lines which are more than 2 earth radii ( $L > 2$ ) from the center of earth at the magnetic equator (at  $L \gg 2$ , near the magnetic poles, lifetime is on the order of minutes). Thus, to avoid long-lived artificial belts, low-level nuclear-pulse operation should be restricted to regions of magnetic latitude  $\geq 40^\circ$  north or south, where  $L \geq 2$ .

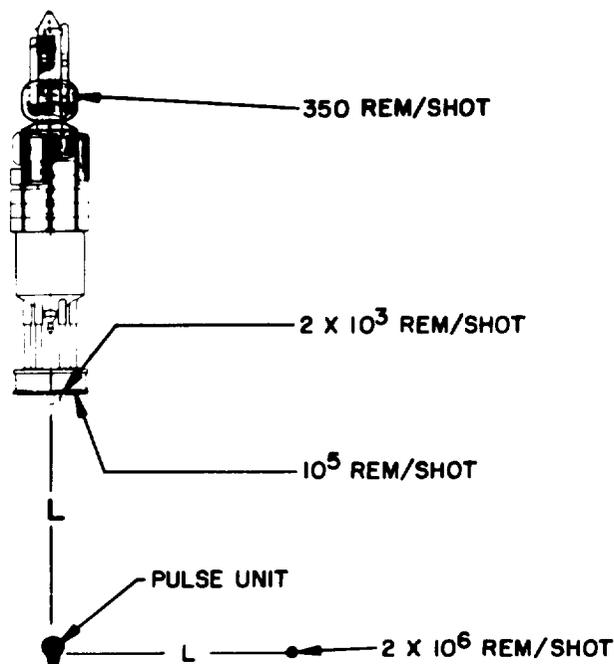


Fig. 5.1--Operational Nuclear Environment About the 10-M Propulsion Module

Eyeburn, due to casual observers seeing nuclear-pulse operations from on or near the surface of the earth, is still another radiation consideration (but now the phenomenon is thermal radiation, caused by re-radiation of air heated by the explosive energy). It has been determined conservatively, however, that nuclear-pulse operations above 90 km altitude will not be harmful to the unprotected eye. Other estimates place the damage threshold at approximately 30 km. The operational modes of this study all commence nuclear-pulse operations above 50 km and most occur above 90 km.

## 5.2. INTERNAL NOISE

A preliminary analysis to determine the existence (or absence) of an internal acoustic noise problem during pulse propulsion and the general magnitude of any such problem was conducted during the early study phase. It was concluded from the analysis that most of the acoustic energy in the shielded power-flight station would be at low frequencies with noise levels higher than the currently permissible maximum for military airborne vehicles, unless reduced.

The conceptual configurations designed in the later phase of the study, after the preliminary noise analysis, were reviewed for their acoustic qualities. The proposed configurations for the crew compartments were considered to provide excellent damping properties. Furthermore, the possible use of a foamed polyurethane core material for the

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payload spine walls (which support the crew compartment) would provide a means of high vibrational-energy absorption. It was therefore concluded that the configurations proposed provide sufficient stiffness and damping for the control of low-frequency noise and vibration. At the higher frequencies, the provision of conventional fiberglass sound-proofing should adequately control any excessive noise in this spectrum.

### 5. 3. GROUND FACILITIES AND OPERATIONS

The ground facilities, support equipment, and the operations required to prepare nuclear-pulse vehicles for space operations were investigated in a preliminary way. Primary consideration was given to the use of Saturn V in conjunction with the 10-m nuclear-pulse vehicles and to the required modifications to Saturn V facilities at Cape Kennedy.

For facility considerations, the stage preparation building(s), the vertical assembly building, a payload preparation area, the launch pad, and a nuclear-pulse-unit (propellant) storage and loading area were of principal concern. For operational Modes I or II, which use only the Saturn S-1C stage, the low-bay building normally housing S-II preparation activities could be used for the nuclear-pulse propulsion module. Minor modifications to platforms and dollies and different equipments would be required. For Mode III operations, which use the S-II stage, a separate propulsion module preparation building would be necessary. The high-bay area of the Saturn V vertical assembly building, with no major modifications foreseen, would be used to assemble the propulsion module to the S-1C or S-II stage. A new building to house and check out the spacecraft's operational payload section prior to its assembly with the launch vehicle would probably be required. The launch-pad structure, it appears, would require no additions or modifications. It is proposed that a conventional ordnance storage facility at a separate area in the vicinity of the launch pad be used to store the required nuclear-pulse units. Access to this area would be required only by the propellant delivery carrier (barges) and by the propellant loader (arming tower).

Major support equipment items, such as checkout, assembly, and launch equipment, used for conventional Saturn V operations in most cases appears applicable to nuclear-pulse spacecraft use. Some new items for ground transport of the vehicle components would be required and some Saturn transporters could be suitably modified. The mobile arming tower would require some major modifications. Transport barges would be used to deliver the propulsion module over significant distances; the large size could permit the use of intercoastal waterways.

Ground operations for Mode III, in which all spacecraft components are delivered to orbit by the two-stage Saturn V, would be similar to

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operations involving other Saturn payloads, except for the nuclear safety precautions when the payload is composed of loaded propellant magazines. For Mode II operations, in which the nuclear-pulse vehicle serves as the second-stage for the boost-to-orbit, ground operations would be different and a possible sequence of operations is proposed.

#### 5.4. GROUND-HAZARDS ASSESSMENT

Nuclear-pulse vehicles use a large number of plutonium and/or uranium-bearing nuclear explosive devices. The devices are similar to those of today's stock-piled weapons in that they have multiple fail-safe mechanisms plus inherent safety characteristics that preclude their accidental nuclear detonation. Each device also contains a quantity of high explosive (HE), however, which may or may not detonate under certain accidental conditions. Such detonation of the HE will rupture the case, fracture, disperse, perhaps vaporize the nuclear material, but will not create a nuclear event.

Nevertheless, because of the biological implications of plutonium dispersal and also the serious political consequence of any accidents involving nuclear materials, a preliminary assessment has been made of the maximum credible accidents which might occur and of the possible results. Three types of failures, classes I, II, and III, that might result in ground hazards have been investigated.

The Class I failure represents a booster or lofter-stage failure on or near the launch pad, which results in an explosion or fire. Assuming some 1,000 pulse units fall into the fire and all of the HE (~20,000 kg) detonates, there would be no nuclear event, but overpressures up to 1 psi would be expected at distances of approximately 300 m (1,000 ft) and a possible shrapnel hazard to distances of 2,000 m (6,500 ft) from the explosion. The first-stage chemical propellant would, in most instances, provide a more severe blast and shrapnel hazard. A more serious problem, however, would result from the possible burning or vaporization of plutonium, which could produce a downwind inhalation and ground-contamination hazard. This potential hazard could cause a requirement for a remote launch pad and a considerable over-water downwind exclusion area. An interesting possible countermeasure to such a Class I hazard is an off-shore launch pad, arranged for water quenching so that the HE in the pulse units would not detonate.

The Class II failure is one in which the nuclear-pulse vehicle or a load of nuclear-pulse propellant (pulse units) fail to reach orbital velocity and hence will return to earth. If the vehicle or payload carrying the pulse units impact the earth intact, the local area hazards as given for Class I again occur. In this event, however, the impact might be outside

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of a controlled area. A possible countermeasure for this class of accident is to intentionally detonate at the ballistic apogee all high explosives of one or more pulse units of such yield as to guarantee vaporization of the nuclear fuel at the maximum altitude possible. (It is not presently planned to carry personnel on boost-to-orbit launches of either the nuclear-pulse vehicle or pulse-unit payloads.) Dispersal of nuclear materials by this technique could then accrue over a sufficient area to attain dilution well below establish tolerance levels.

The Class III failure represents a single aborted pulse unit which has failed to destroy itself by either the automatic or command destruct system and thus is subject to reentering the earth's atmosphere undamaged and falling to earth. The HE in the pulse unit might detonate on impact, producing 1 psi overpressures up to 35 m (115 ft) and spreading shrapnel and plutonium contamination to 300 m (1,000 ft). Although no nuclear event would occur, the political implications of the HE event described, at some random point on earth, could be sizable. With proper redundancy in the design of the destruct system, however, the probability of such an event occurring can be made arbitrarily close to zero.

#### 5.5. FLIGHT-HAZARDS ANALYSIS

From a preliminary investigation of the flight hazards to mission personnel, to aid in the conceptual design of both the propulsion module and the personnel accommodations, five types of hazards were considered: nuclear radiation, on-board fire or explosion, propulsion failure, meteoroid damage, and boost-to-orbit abort.

Sources of radiation considered were the nuclear-pulse explosions, solar flares, and the earth's radiation belts. As discussed in Section 3 on vehicle designs, the nuclear-pulse propulsion requirements set the personnel shielding thicknesses, which are then found adequate for typical solar flare activity and more than adequate for traversing the radiation belts. The radiation accumulations during typical missions, on the other hand, are low enough to avoid severe material problems.

The possibility of uncontrolled fire or explosion, such as to call for escape from the basic vehicle, appears remote. An explosion on the propulsion module would, however remote, be most likely to occur during propulsion periods. In this event, all mission personnel are within the shielded powered-flight station, so this compartment was designed to also serve as an escape vehicle.

Failure of the nuclear-pulse propulsion module, due to its essentially mechanical nature, would probably result from failure of one of the mechanical components or of the control system that synchronizes their

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operations. It is likely that most failures of this nature could be repaired in coasting flight by the crew, using on-board equipment and spares. If an irreparable failure should occur, the mission personnel could elect to leave the basic vehicle via the more confining and less redundantly supported escape vehicle, choosing the most opportune time.

The nuclear-pulse vehicle is subject to meteoroid interactions as are other spacecraft. For several reasons, however, it appears to be less vulnerable than chemical or chemo-nuclear systems and thus provides a greater margin of safety from this hazard. A foremost advantage is the single, relatively dense, compact stage, which thus provides a small target. In addition, it carries the bulk of its propellant as discrete, dense pulse units, the remainder of which is dense and easily retained fluids like water and oil; no cryogenic materials. Finally, its most vulnerable component, the first-stage shock absorber, is readily retracted under a meteoroid bumper during coast phases.

Boost-to-orbit aborts, for the nuclear-pulse systems of this study, is not a personnel-safety concern, since the vehicles are either carried to orbit unmanned or boost themselves to orbit unmanned. Mission personnel are assumed carried to orbit by a chemical booster which is man-rated for the task.

## 5.6. MAINTENANCE AND REPAIR CONCEPTS

The nuclear-pulse vehicle has several inherent advantages over most other space propulsion systems with respect to in-flight maintenance and repair. It is noncryogenic, its propellant is carried as discrete, easily handled units, its working temperatures are moderate ( $\leq 600^{\circ}\text{F}$ ) and radiation activation is sufficiently low that it does not constrain even prolonged work with the module's internal mechanisms.

It is anticipated that a considerable amount of preoperation inspection and preventative maintenance of the propulsion system will be desirable, particularly after the long coast periods typical of exploration missions. Hence, access to most of the internal and external mechanism is planned for. Further, a relatively large repair-bay—spares storage volume, adjacent to the propulsion module body, is provided on exploration-vehicle designs.

In the current propulsion-module design concepts, reliability has been given considerable thought toward the goal of minimizing the need for maintenance and repair as well as minimizing the probability of failure. This has been reflected in the use of redundancy in some mechanisms and control subsystems, the employment of fail-safe concepts, and the application of conservative design allowances where redundancy is impractical.

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## 5. 7. FISSIONABLE-MATERIAL AVAILABILITY

Each nuclear-pulse unit contains an appreciable amount of fissionable material, and the larger missions of interest typically consume several thousand pulse units. This amount of consumption of a recently strategically-scarce material is naturally of some concern. It is therefore of interest to examine mission requirements in comparison to the availability of raw materials and recent, as well as projected, production rates.

In the recent past, the United States procurement of  $U_3O_8$  has been about  $3 \times 10^7$  kg annually, approximately one-fifth of which has been used for the production of  $Pu^{239}$ , a suitable fissionable material for nuclear-explosive devices. An estimated 30-yr supply of the  $U_3O_8$  is readily available at the  $3 \times 10^7$  kg annual rate, with a vastly larger quantity available by more difficult (and more costly) recovery methods.

The analysis of availability considered four production rates, or methods, all based on an assumed annual procurement of  $3 \times 10^7$  kg of  $U_3O_8$ . Method A (as has been done) converts one-fifth of the raw material at 55 percent efficiency. Method B converts all of the  $U_3O_8$  at the same 55 percent efficiency. Methods C and D assume large-scale breeding (with 40 percent breeding efficiency), which increases the  $Pu^{239}$  output by a factor of  $\sim 100$  and greatly reduces its cost at the same time. Method C also uses one-fifth of the  $U_3O_8$  supply; method D uses all of it.

Typical mission requirements for  $Pu^{239}$ , taken from missions of this study, compared to the annual production rates for the four assumed production methods are given in Table 5.1. This table also shows the plutonium cost per kilogram of payload carried on the mission (based, beyond the first column, on some rather gross assumptions as to cost-quantity effects and production method costs). The fraction of annually produced plutonium used per mission is not disturbing, even when present-day conversion methods are employed; it becomes nominal should large-scale breeding be undertaken.

Table 5.1

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

Configuration and Mission Description	Plutonium Requirements, in Per Cent 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

## 6. DEVELOPMENT PLANNING

A plan for the development of a nuclear-pulse propulsion module has been prepared and a cost estimate made. Certain characteristics of the propulsion system that appear significant to its developability are: (1) the propulsive energy source consists of repetitive nuclear explosions remote from the main propulsion-system hardware; (2) the nuclear and nonnuclear environments and effects appear to be almost completely separable; (3) neither high- nor low-temperature extremes exist in any structural material areas, with the exception of the high-temperature ablation problem constrained to a thin surface region of the module's pusher; and (4) nuclear radiation effects after periods of powered flight are quite low. These characteristics make practical the separating of development problems such that a larger percentage of the development can be carried out at the surface using high-explosive loading and underground nuclear testing, permit the handling of test specimens after exposure to nuclear-energy pulse units, and eliminate the need for extreme-temperature structures or subsystems.

### 6.1. OBJECTIVES AND DEVELOPMENT APPROACH

The aim of the development plan is to bring to a point of initial operational capability (IOC) a propulsion module for a nuclear-pulse vehicle that has the characteristics required to perform a variety of manned space missions. The 10-m propulsion module was chosen as the size to consider for purposes of establishing a tentative schedule and cost estimate.

A step-by-step development program, progressing from the development of components to the preliminary flight rating test (PERT) and qualification, can be carried out in fairly well defined predevelopment and development phases. Each phase will provide answers to definite operational problems. Figure 6.1 is a representation of the relation between the major development areas of effort 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. 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 with high explosives and then to complete propulsion modules with either high explosives or nuclear sources with meaningful results. Different specific 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/or prove each component is shown in Fig. 6.2. Though separable, development of the pulse unit and the propulsion module must be concurrent because the results of the development of one are the design parameters for the other. Each appears developable to a high degree of reliability prior to costly nuclear space tests.

DEVELOPMENT AREAS	EXPERIMENTAL TECHNIQUES									
	HE PLASMA	EM PLASMA	COMP. TEST (VACUUM)	HE PULSE	HE REP. PULSES	NUCLEAR UNDERGROUND			NUCLEAR REP. BALLISTIC	NUCLEAR REP. ORBIT
						PUL. UNIT	SCALES	MODULE		
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 4a. REP. S.A. & ATTACH. RESPONSE 4b. S.A. COOLING			S.A. ONLY S.A. ONLY	●	●		●	●	●	
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. 6.2--Development approach

### 6.2 SCHEDULE AND COST ESTIMATES

A schedule for the efficient execution of the development and qualification of the 10-m module is shown, in condensed form, in Fig. 6.3. The development schedule is divided into four phases: Predevelopment Phases I and II and Development Phases I and II. Predevelopment Phase I is considered accomplished. Each remaining phase is scheduled to require three years. Following these is a qualification program which is also for a three-year period; IOC is expected to be obtained at the end of this period, or about 12 years from the beginning of Predevelopment Phase II. Nine program milestones are scheduled throughout the program.

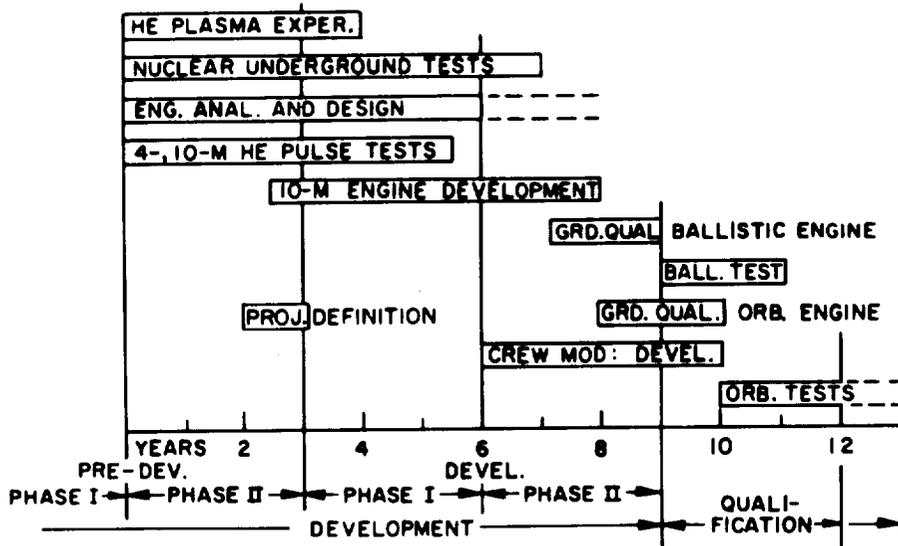
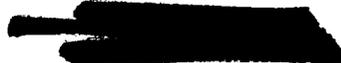


Fig. 6.3--Condensed development schedule

Development cost estimates for the 10-m propulsion module have been prepared, based on the phased schedule just discussed. Current estimated totals for the different phases of the program are as follows:

	<u>\$ Million</u>
Predevelopment Phase II . . . . .	63
Development Phase I . . . . .	152
Development Phase II . . . . .	239
Qualification program . . . . .	<u>1,037</u>
Total . . . . .	1,491

Development costs for the 10-m nuclear-pulse propulsion module are accordingly considered to be approximately \$2 billion or less (since one significant figure is about the limit of accuracy for early development-program estimates).



## 7. COMPARISONS WITH OTHER SYSTEMS\*

The propulsion module for nuclear-pulse systems is relatively small, dense, and compact in comparison to other space propulsion systems (due primarily to the high  $I_{sp}$  and to the dense nature of its propellant), yet it is relatively massive (due to the high-strength and mass-distribution requirements of its impulsive loading). Thus, it has a relatively high inert mass fraction and correspondingly low propellant mass fraction. In spite of this penalty, however, it excels in both payload fraction and payload delivery cost-effectiveness, when compared with other advanced space propulsion systems.

Figure 7.1 shows the variation of mass fraction with propellant weight for nuclear-powered hydrogen vehicles and for the 10-m and 20-m (diam) nuclear-pulse vehicles. The mass fractions for the hydrogen vehicles powered by solid-core-reactor (SCR) engines or gas-core-reactor (GCR) engines of given thrust ( $k = 10^3$  lb,  $M = 10^6$  lb) or  $I_{sp}$  values are based on detailed structural and weight analyses performed during General Dynamics/Astronautics studies of manned planetary missions and post-Saturn ELV's. The mass fractions for the nuclear-pulse vehicles were derived from General Atomic study results (as given elsewhere in this report). The propellant mass fractions for the nuclear-pulse vehicles are seen to be lower than for other systems, even when compared at comparable thrust-to-propellant-weight ratios (the effective thrust of the 10-m module "G" is 787,000 lb; the 20-m module "H" produces  $3.6 \times 10^6$  lb).

The specific impulse potential of the nuclear-pulse vehicle is seen to more than overcome its inert weight penalty, as shown in Fig. 7.2, a plot of payload fraction versus ideal velocity capability. Here nuclear-pulse vehicles somewhat larger than the 10-m module are judged to have a 3,000 to 5,000-sec  $I_{sp}$  potential as compared, for example, to 1,500 to 2,000 sec for the gas-core-reactor system.

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\*The system comparisons portion of this study was performed by Krafft Ehrlicke, as part of an interdivisional assistance effort by General Dynamics/Astronautics. Mr. Ehrlicke drew upon his considerable experience and from data resulting from many previous and concurrent studies of space systems. More complete comparison data may be found in Vol. IV of this report, Mission Velocity Requirements and System Comparisons (Unclassified), prepared by General Dynamics/Astronautics. This brief section is largely confined to a summary of Mr. Ehrlicke's work that includes classified performance data on the nuclear-pulse systems.

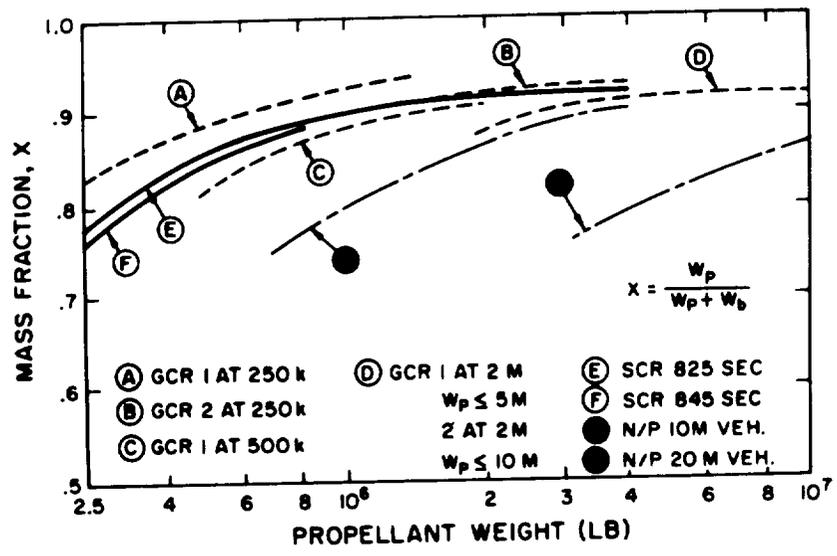


Fig. 7.1--Propellant mass fractions of nuclear propulsion systems vs propellant weight

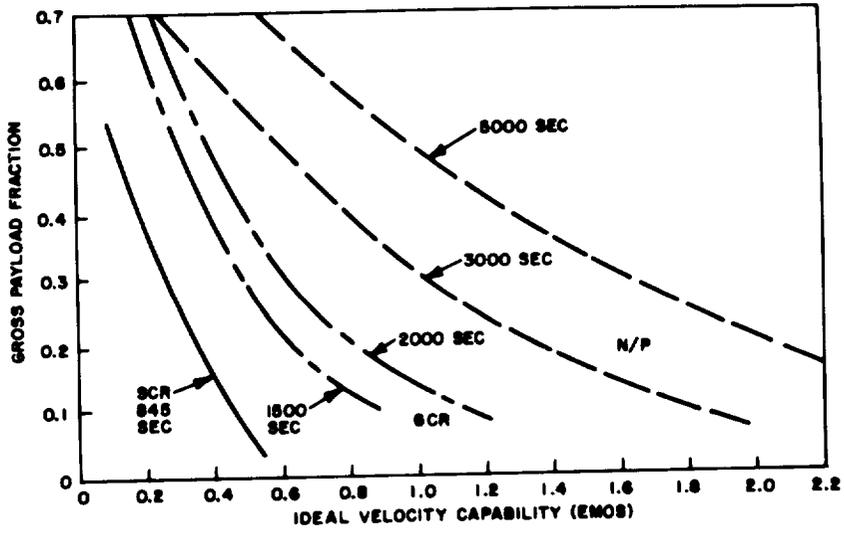


Fig. 7.2--Payload fractions on nuclear interorbital vehicles

Similarly, the cost-effectiveness of the nuclear-pulse systems is seen (Fig. 7.3) to be much better than that of the comparative systems except at low mission velocities. In this comparison, the 10-m nuclear-pulse vehicle category (1,900-sec  $I_{sp}$ ) is included as well as the larger or improved versions having  $I_{sp}$ 's of 3,000 and 5,000 sec.

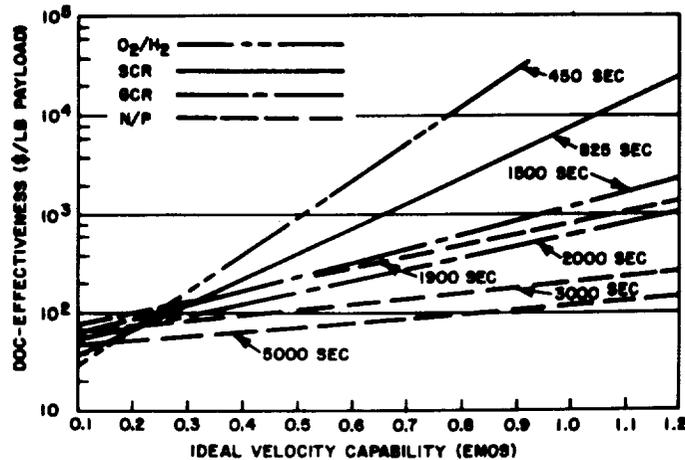


Fig. 7.3--Direct operating cost-effectiveness vs ideal velocities for various propulsion systems

Comparisons were also made of the applicability of the space propulsion system considered to the performance of various missions. The comparisons are on the basis of performance capability and operational characteristics. The results of one comparison are summarized in Table 7.1. The major column headings refer to initiation (suborbital or orbital) of the given type of propulsion, i. e., C = chemical, SCR = solid-core-reactor engine, CGR = gas-core-reactor engine, N/P = nuclear-pulse engine, and N/E = nuclear electric. The nonapplicability of nuclear-pulse propulsion in the suborbital-start category is based on the potential disturbances caused by the current-design nuclear pulse units if used on frequency missions in the vicinity of the earth.

Another comparison is summarized in Table 7.2; it presents an evaluation of the principal propulsion systems regarding their capability to operate in different regions of space. Here, "operation near Sun" means the capability to function at small fractions of the astronomical unit ( $0.4 \geq R \geq 0.2$  A. U.). The limited capability of the nuclear electric vehicle (N/E), and, to a lesser extent, of the controlled-thermonuclear-reactor (CTR) vehicle near the Sun is based on the intense heating of the radiators by the Sun, which greatly reduces and eventually eliminates their function of rejecting the vehicle's excess energy. The insensitivity of the nuclear-pulse vehicles (N/P) to a wide range of environmental conditions and energy requirements is apparent.

Table 7.1

MISSION VERSATILITY OF VARIOUS PROPULSION SYSTEMS

Mission	Suborbital				From Orbit				
	C	SCR	GCR	N/P	C	SCR	GCR	N/P	N/E
Earth orbital delivery	*	*	*						
Deep space injection parabolic	(*)	(*)	*		(*)	(*)	*	*	*
Deep space injection hyperbolic	(*)	(*)	*		(*)	(*)	*	*	*
Lunar orbit delivery	(*)	(*)	*		(*)	*	*	*	*
Lunar hovering delivery			(*)				*	*	
Hyperbolic rendezvous pickup			[(*)]	*			[*]	*	
Planet hyperbolic delivery		*	*		(*)	*	*	*	*
Planet capture delivery		*	*			*	*	*	*
Planet capture hovering delivery								*	

\* = Reusable; (\*) = one-way; [\*] or [(\*)] = Venus and Mars only.

Table 7.2

COMPARISON OF PRINCIPAL SPACE PROPULSION SYSTEMS

Propulsion System	Space Region or Operation				
	Ascent, Descent	H. R. † Pick-up Vehicle	Operation Near Sun	Solar System	
				Outer	Inner
Solid-core reactor	**	---	**	*	---
Gas-core reactor	**	*	**	***	*
Nuclear pulse	***	***	***	***	***
Nuclear electric; controlled ther- monuclear reactor	---	---	*	***	***

†H.R. = Hyperbolic rendezvous.    \*\*Possible, but not attractive for long range.  
 \*Barely possible.                    \*\*\*Excellent.

## 8. CONCLUSIONS AND IMPLICATIONS

A major and unexpected result of this study was the very significant mission potential of the 10-m, Saturn V compatible, nuclear-pulse vehicles, particularly when operated in the orbital start-up mode. The potential of such vehicles makes planetary explorations plausible with a small, single-stage (from orbit) vehicle, yet with larger personnel complements, more systems redundancy, more shielding, and a lower direct operating cost than heretofore seriously conceived. Large-capacity lunar transportation or logistic systems are also indicated, with their cost-effectiveness a factor of 2 to 4 better than otherwise foreseen. There are several implications to these results for the planning of advanced space missions and for near-term efforts to verify the indicated performance potential and the political acceptability of the system.

### 8.1. STUDY CONCLUSIONS

A very attractive inner-planet exploration mission capability is indicated for the 10-m class of vehicles compatible with Saturn V. The unusual performance reserve and mission versatility inherent in this single-size propulsion module (due largely to its dense, easily packaged, highly storable, high  $I_{sp}$ , propellant) permit its efficient use for a wide variety of payloads and mission velocities. This versatility permits, for example, sending scientists as well as astronauts on the mission, increasing the mass of the desired payloads, changing to lower-risk mission profiles, sustaining degradations in vehicle weight or performance, etc., with the change occurring after the propulsion system is well into development. With this high-performance system, the scheduling of interplanetary operations for specific "good" years becomes a secondary consideration. Large-capacity, cost-effective, lunar systems are shown feasible with the same 10-m propulsion system, and means were indicated that could further improve the system effectiveness, carrying smaller, near-term-compatible payloads. For missions to the outer planets, larger and/or higher  $I_{sp}$  versions show this capability.

An investigation of operational systems, operational problems, and hazards indicated no fundamental characteristic that would preclude the operational employment of nuclear-pulse vehicles. Operations from the Atlantic Missile Range, with some site modifications, appear feasible. Two potential hazards require further consideration and understanding to be satisfactorily handled: that of boosting aloft large quantities of high explosive packaged with plutonium (in nuclear pulse units) and the potential (though small) contamination of the earth's atmosphere (to varying degrees a characteristic of all nuclear propulsion space systems).

Development of the nuclear-pulse propulsion system, due partly to the smaller 10-m size and relaxed thrust-to-weight requirements, now appears to be a less-demanding program than previously assumed. Two new key program elements are responsible: (1) relatively inexpensive, contained, underground nuclear experiments to develop the pulse unit and to resolve propellant-pusher interactions with both scaled and full-size components; and (2) repetitively cycled high-explosive shock generators to attain on the ground by nonnuclear means repeated mechanical impulses on full-scale modules and at the proper firing rate, such as would be received by the propulsion module in powered flight. Development of the propulsion module through flight qualification is estimated to require 12 continuously active years and cost approximately \$2 billion. Major decision-point milestones occur at 2- to 3-year intervals.

## 8.2 IMPLICATIONS FOR FURTHER EFFORT

The mission capability and cost-effectiveness indicated for the 10-m propulsion module, plus the significant growth capability of the system, could, if confirmed have a strong influence on the scope and economy of our National Space Program. Furthermore, the past six years of analytical and experimental research on nuclear-pulse propulsion have done much to reasonably assure the technical feasibility of the system; but confirming nuclear tests have yet to be performed. A well-balanced program might well include the following elements.

Underground Nuclear Tests. Contained, small-yield-source, underground tests to correlate pulse-unit design parameters and to correlate ablation and other early-time interaction phenomena with calculations is now highly desirable. Such tests would do much to establish feasibility and to determine attainable specific impulses.

Nuclear-source Design. The design, fabrication, and testing of the nuclear-energy sources fall within the province of the Atomic Energy Commission. This includes consideration of such devices as the low-yield nuclear source recently proposed for underground testing. To date, all project technical studies have considered only nuclear devices such as are currently stockpiled or well understood. It is almost certain that nuclear devices designed specifically for propulsion purposes would provide improvements in the areas of economics, performance, and contamination.

Contamination-problem Investigation. During this study, the problem of atmospheric contamination was briefly investigated. Broad areas of uncertainty as to the trappage or contamination phenomena and hence the amount of contamination were found, but several potential ways of greatly reducing or possibly eliminating the problem were also found.

This situation, which also exists in different detail for other advanced nuclear propulsion systems, is believed to merit a more thorough investigation.

Engineering Studies. Engineering design studies, including studies of the propulsion modules, pulse units and over-all vehicle systems, should continue at a modest level. Such studies should be directed toward more integrated, more efficient, and more workable designs in sizes or capacities to best suit the current space program plans. Their further purpose is to give direction to associated research and test programs.

Applied Research Programs. Analytical and experimental programs now under way, in particular the HE-driven impulse testing of components and the HE-driven ablation tests, should continue. These applied research programs (1) investigate and establish criteria for impulsive-loaded components while further developing the HE testing and cycling techniques and (2) correlate ablation analytical and experimental data while developing instrumentation techniques for nuclear testing.

Advanced Mission Planning. Studies of space explorations and missions have retrenched in scope over the past few years as the mission problems and limitations of the planned propulsion systems became better known. This situation has resulted in plans for further subsystem optimization, miniaturization, and marginal weight-saving techniques and highly optimized, split-second, flight profiles--all of which are costly. If the mission potential of nuclear-pulse systems can be realized, it could stem or reverse these trends, making a profound difference in total system economics. Studies should be performed in sufficient depth to perform such economic trade-offs.

*(Unclassified title)*

**NUCLEAR PULSE VEHICLE STUDY**

**CONDENSED SUMMARY REPORT**

**TO**

**GEORGE C. MARSHALL SPACE FLIGHT CENTER  
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION  
HUNTSVILLE, ALABAMA**

**Contract NAS 8-11053**

*14 Jan, 1964.*

# UNCLASSIFIED

## FOREWORD

Technical studies on nuclear pulse propulsion were initiated in 1957 by the General Atomic Division of General Dynamics Corporation. Early governmental support was awarded in 1958, initially from the Advanced Research Projects Agency (ARPA). From 1960 to the present, the research support has been continued by the United States Air Force and by General Dynamics Corporation. In July, 1963, the Marshall Space Flight Center of the National Aeronautics and Space Administration initiated an applications study with General Atomic for determining possible NASA-oriented mission applications of nuclear pulse propulsion, for obtaining a better understanding of key operational questions and potential hazards, and for studying implications to other proposed future NASA programs.

Summarized here are the results of the mission-oriented applications study performed for NASA under Contract NAS 8-11053. The study is treated more fully in the expanded summary report and in the technical report on the *Nuclear Pulse Vehicle Study*, both of which are to be issued subsequently.

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## 1. INTRODUCTION

Over six years of continuous and intensive analytical and experimental research have been concentrated on nuclear pulse propulsion. This propulsion concept — a prime example of the peaceful application of nuclear explosions — offers performance potential for realizing economic manned space travel to any part of our solar system.

A nuclear pulse propelled vehicle is shown conceptually in the frontispiece. Briefly, the propulsion system operates as follows: Low-yield nuclear pulse units are detonated sequentially external to and below the vehicle. A substantial fraction of the mass of each pulse unit — the propellant — is directed toward the bottom of the vehicle as a high-velocity, high-density plasma which is intercepted by a large circular metallic plate — the pusher. The momentum of the propellant is transferred to the pusher and the resulting accelerations are smoothed out by shock-absorbing devices to levels of a few *g*'s in the upper vehicle — well within human tolerance. System performance is characterized by both high thrust-to-weight ratios and large specific impulses.

The research effort, about half of which has been experimental, was directed initially to demonstrating scientific feasibility. Now that the concept appears to be feasible without new "inventions," the current effort includes determining the engineering practicality of the concept through integrated propulsion-system design studies and applied research programs to provide technical information relating to pulse-unit design, pusher ablation, and structural integrity of the pusher attachments and shock-absorbing systems.\*

Earlier design studies concentrated on vehicles of large sizes (4,000 tons) and high specific impulse (4,000 to 6,000 sec) that would be capable of direct launch from the earth's surface or suborbital start-up and would have a vehicle thrust to weight  $\geq 1.25 g$ . Such vehicles would have propulsion-module inert-weight fractions of 0.3 to 0.4 and pulsing intervals of about 1 sec. During the current applications study, it became evident that NASA mission constraints on the propulsion system were far less demanding, which tended to relieve design and operational problems, improve reliability, and increase over-all system performance. For example, NASA mission requirements allow orbital nuclear start-up, with rendezvous, which in turn will permit:

- Selection of near-optimum vehicle thrust-to-weight ratios.
- Lower inert-weight fractions, higher payload fractions, higher mission velocities.
- Greater time spacing between nuclear pulses.
- Lower net vehicle accelerations, simpler and more reliable designs, improved specific impulse for a given size.
- Possible elimination of fission products dispersed in the atmosphere.
- Elimination of long-term artificial electron belts.

These new considerations, together with recent advances in pulse-unit design, greatly enhance the mission capabilities of small propulsion modules that are compatible with the Saturn V:

Propulsion module weight.....	$\leq 200,000$ lb
Inert weight fraction .....	10% to 20%
Specific impulse .....	1,800 to 2,500 sec
Propulsion module diameter.....	10 m

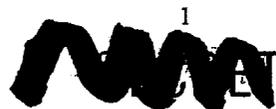
These smaller modules tend to be less efficient and may carry a larger nuclear-fuel inventory for a given mission, which increases the operating cost per unit payload over that of the larger, high  $I_{sp}$  vehicles. However, in comparison with other advanced propulsion systems for the 1970 era, the economics of the small vehicles are still very attractive. All of these factors, together with its desirable operational characteristics, make the Saturn V compatible module an interesting early development goal.

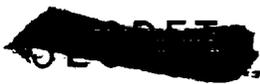
Further, this small propulsion module represents an early system capability that establishes a technology base from which gross performance improvements are indicated, leading to a true solar-system transportation capability. No major operational or fuel availability obstacles appear to preclude appropriate employment of nuclear pulse vehicles in the National Space Program.

Primarily because of the longer pulsing intervals allowable, a ground-oriented, phased, development plan appears practical, permitting not only prototype development but also prelaunch *qualification* of flight-rated operational propulsion modules at ground facilities.

A note of caution and of optimism must be injected here. Design data for the Saturn V compatible propulsion modules were not derived from specific point-design studies; they were obtained by scaling from much larger modules. Some performance inaccuracy may therefore exist. Design studies initiated too recently to be included in the present mission studies indicate that the scaled performance estimates used are probably conservative.

\*This work has been reported in *Technical Summary Report, Nuclear Pulse Propulsion Project (Project ORION)*, Air Force Systems Command, RTD TDR 63-3006, Vols. I-IV, 1963 (S-RD report) and in more than 300 technical reports that have been issued on the Project.





## 2. STUDY OBJECTIVES

This seven-month study was performed to explore the mission potential of the nuclear-pulse space vehicle concept. A variety of space missions were considered for both "favorable" and "unfavorable" departure years and for near-term and long range. The study, of broad scope and limited depth, encompassed three primary and three secondary objectives:

### *Primary Objectives*

1. Determining the mission potential of nuclear-pulse space vehicles for lunar and planetary missions in the time period between 1975 and 1995 for a variety of mission profiles and operational objectives.
2. Defining expected operational systems, operational problems, and possible approaches for solution to the problems.
3. Establishing typical development programs directed toward reaching major planetary mission capability in the 1975-1995 time period.

### *Secondary Objectives*

1. Developing parametric propulsion-vehicle system design data in a form useful for the various mission studies.
2. Defining possible areas of growth or improvement in technology which would have a substantial influence on future performance, operations, or development programs.
3. Providing nuclear-pulse vehicle performance techniques capable of simulating flight and indicating performance for any given vehicle.

## 3. RELATIONSHIP TO OTHER NASA EFFORTS

The nuclear pulse space vehicle, assuming that its operational potential approaches that indicated by this study, would appear capable of having a strong influence on future NASA programs during and beyond the 1970's. At least three areas of NASA effort could be affected: (1) studies and eventual execution of lunar and planetary exploration and transportation, (2) the earth-launch-vehicle programs, and (3) operational sites and ranges.

### LUNAR AND PLANETARY EXPLORATION AND TRANSPORTATION STUDIES

There are several significant operational and economic advantages for nuclear pulse propulsion in its initial realm of operation — earth orbit to the lunar or planetary orbit and return. A major operational advantage is that it is capa-

ble of performing *with a single stage* the complete round trip. The crew need master only *one* restartable propulsion system, which they can exercise prior to departing earth orbit on a difficult mission and, if necessary, maintain or service during the mission.

An economic advantage appears evident primarily for two reasons: (1) one stage, instead of several, for interorbit flight operations reduces the required over-all development task, interface and staging complexities, and series reliability requirements, all of which are very costly, and (2) relatively low earth-orbit departure weights keep down the boost-to-orbit costs, which typically predominate the economics of such missions.

### EARTH-LAUNCH-VEHICLE (ELV) PROGRAMS

The mode of operation most compatible with NASA's near-future requirements is an orbital start-up of the nuclear pulse vehicle, with conventional ELV's employed for the important boost to orbit. Saturn V appears to be appropriate for a considerable period of time for space exploration and transportation.

For the post-Saturn era, tentative ELV requirements for the nuclear pulse vehicle are low-aspect-ratio ELV configurations to match the relatively large diameters of larger nuclear pulse vehicles and ever-improving booster economics, both of which tend to favor recoverable boosters.

### OPERATIONAL SITES AND RANGES

The operations, sites, and facilities for nuclear pulse propulsion will introduce some new operational considerations, primarily in the handling, loading, and launching of nuclear fuel and high explosives. Here the utilization of AEC and military experience seems logical.

To eliminate fission-product fallout or the trappage of electrons in artificial radiation belts, some different constraints on earth-orbit departure trajectories may be required. These, in turn, may be reflected in modified boost-to-orbit trajectories, range-safety considerations, etc.

A preliminary investigation of basing requirements indicates that modest modifications to the now-programmed Cape Kennedy facilities will be required to support Saturn V-boosted nuclear pulse systems. No hazards are envisaged (for the orbital start-up mode) which would necessitate a launch site other than part of the Cape Kennedy complex.

## 4. METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

The study was divided into two phases: a parametric phase to explore a very broad range of sizes and mission capabilities and then a specific-conceptual-system phase to

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investigate in greater detail the mission capability of two selected sizes of nuclear pulse vehicles.

### PARAMETRIC STUDY PHASE

Four tasks were performed during this phase of the study.

1. Parametric characteristics defining the performance and operation of nuclear pulse propulsion modules, as functions of effective thrust, were derived from earlier propulsion-system design studies over a wide range of thrust. These parametric characteristics are the *principal assumptions* of the study.

2. Vehicle systems were defined and "exercised" by computing their performance for a range of mission velocities encompassing the simpler and more difficult Mars explorations, lunar missions, and selected Jovian explorations. Concurrently, mission payload requirements were compiled. Three modes of operation (Fig. 1) were considered: (a) A self-boost-to-orbit mode, called operational Mode I, which

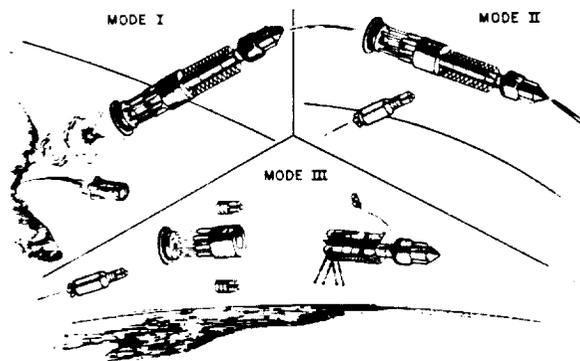


Fig. 1 — Operational modes

requires an effective thrust-to-weight ratio  $>1.0$  to escape the earth's gravity. (b) An orbital start-up mode, Mode III, in which the nuclear pulse vehicle is initially lofted to orbit by a chemical booster. The thrust-to-weight ratios for Mode III can be well under 1.0. (c) An intermediate Mode II in which the propulsion module is loaded in orbit after self-boosting.

3. Comparative direct operating costs and the major cost components were computed in a simplified cost analysis to derive the more economical operating modes and vehicle sizes over the broad range of nuclear pulse systems being explored.

4. Operational problems and hazards unique to nuclear pulse propulsion were explored so as to define and quantify,

to first order, the magnitude of the problem and to identify those problem areas that require further attention.

### SPECIFIC-CONCEPTUAL-SYSTEMS STUDY PHASE

During the second study phase, five tasks were performed; the major portion of the contractual effort, however, was devoted to the first three.

1. Two specific nuclear pulse propulsion modules were defined in conjunction with specified manned payloads for a variety of Mars and Jovian explorations, for Mars logistic delivery, and for lunar logistic and personnel transport. The propulsion modules were sized to be compatible with earth launch vehicles planned for the same time periods. The Saturn V compatible module, a 10-m-diameter (32.8-ft) configuration, received particular emphasis after it appeared capable of more than adequately performing most of the exploration and space logistic tasks of early interest. The second module, of 20-m-diameter (65.6-ft), is compatible with post-Saturn design concepts.

2. Performance and approximate direct costs were determined for the space missions mentioned above.

3. A sensitivity analysis was made by varying, one at a time, the more suspect vehicle-performance or unit-cost inputs and recomputing the total mission performance or costs.

4. A tentative development plan and schedule was generated for an orbital start-up 10-m propulsion module.

5. Advanced versions of nuclear pulse vehicles and their performance and economic potential were reviewed. These data are based on performance characteristics predicted by exploiting known fundamental properties of nuclear fission and fusion devices.

### 5. BASIC DATA GENERATED AND SIGNIFICANT RESULTS

Most of the significant results of this study concern a 10-m-diameter propulsion module, which is about half the size of the smallest module that has previously received serious design consideration, but which has a very impressive (scaled) performance capability in the orbital start-up operational mode. Much of the credit for appreciating such a vehicle's capability goes to NASA for recognizing the logic and value in this size vehicle in spite of its poor propellant economics and comparatively degraded specific impulse.

#### SYSTEMS INFLUENCE FROM THE PARAMETRIC STUDY PHASE

The performance and operating economics of nuclear pulse vehicles are strongly influenced by two rather unique characteristics. One is the variation of specific impulse with

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vehicle size, or with thrust if other operating conditions remain fixed. The nominal curve of specific impulse versus effective thrust used throughout most of this study is shown in Fig. 2. The curve indicates obvious performance advantage with increasing size or thrust.

The second characteristic is the variation of over-all propellant costs with size or thrust, as shown in Fig. 3. Propellant-unit cost decreases rapidly with increased size or thrust. Thus, larger vehicles are strongly favored from the viewpoint of direct operating cost per pound of payload delivered.

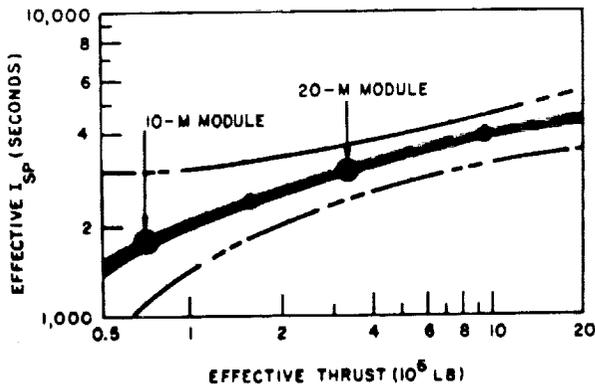


Fig. 2 — Propulsion-module specific impulse

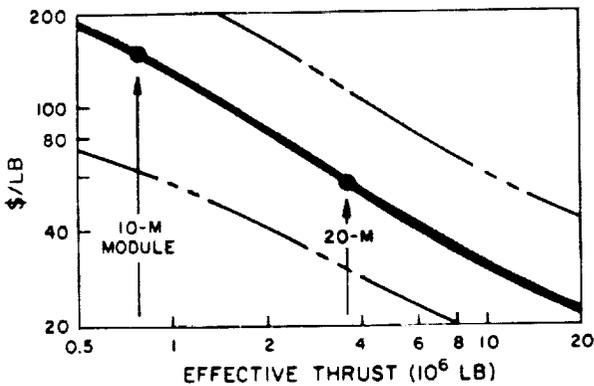


Fig. 3 — Over-all propellant costs

The most useful guidance from the parametric study phase comes from the performance benefits of an orbital start-up, reduced thrust-to-weight ratio, mode of operation. These benefits are seen in Fig. 4, where the single-stage payload delivery capability of typical nuclear pulse vehicles is shown for initial thrust-to-weight ratios of 1.25 and 0.25. At a thrust-to-weight ratio of 0.25, mission velocities of 100,000 fps are attainable with significant payloads at a thrust well under one million pounds.

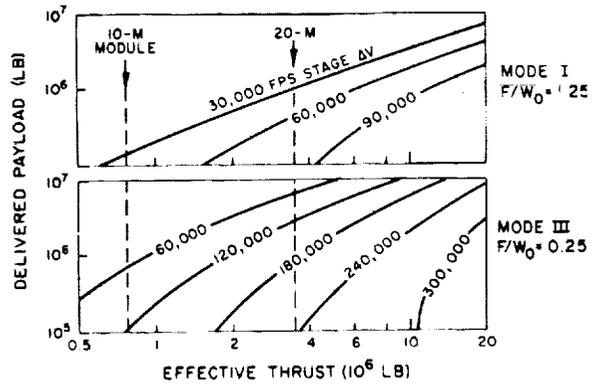


Fig. 4 — Typical payload capabilities

The definition of exploration mission payloads and mission requirements, the compatibility with Saturn V or other ELV's and their major facilities, and the evaluation of mission hazards associated with the various start-up modes of operation for early nuclear pulse systems were also derived from the parametric study phase.

#### THE 10-METER CONCEPTUAL VEHICLE SYSTEMS

The conceptual design of a 10-m propulsion module, as defined and "frozen" for this study, is shown in Fig. 5. The basic module can carry 900 pulse units internally in its "ready-to-fire" racks. Additional propellant may be carried external to the basic module in the expendable propellant magazines shown.

The payload and supplemental propellant region of manned vehicle configurations are atop and around a central payload spine, as shown. Support columns for the propellant magazines and for externally carried payload are located around the central spine.

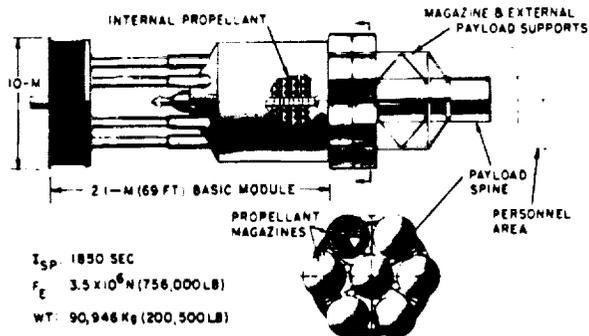
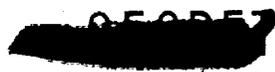


Fig. 5 — Ten-meter propulsion module

#### Compatibility with Saturn V

The 10-m-module configurations, as discussed previously, are compatible with the Saturn V earth launch vehicle. Typ-



ical earth launch configurations are shown in Fig. 6. For the boost-to-orbit vehicles, the Saturn's orbital payload is 100,000 kg (220,000 lb) or less. The center boost-to-orbit configuration can carry not only the entire operational payload structure, all hardware, and most of a Mars mission expendables, but also two manned (or mannable) orbital assembly modules.

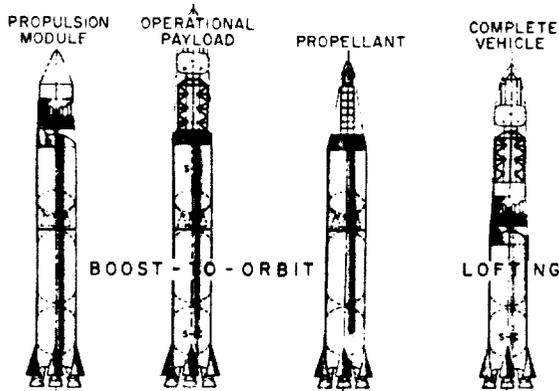


Fig. 6 — Module compatibility with Saturn V

For a suborbital start-up mode of operation, a complete, but ordinarily not fully loaded, 10-m nuclear pulse vehicle can be lofted to about 10,000 fps by a Saturn S-1C stage.

#### Personnel Shielding

During engine operation, all personnel are required to be within a shielded "powered flight station." Shielding requirements for this study have been based on an allowable propulsion dose of 50 rem per mission. These requirements result in a shaped shield of 25 g/cm<sup>2</sup> or more of hydrogenous material on all surfaces of the powered flight compartment. The compartment can therefore also serve as a solar "storm cellar."

#### Mars Exploration System Performance

Mars exploration vehicles using the 10-m propulsion module were conceptually designed and exercised on missions requiring velocities from ~50,000 to 100,000 fps. Departure weights and performance were computed for missions with and without destination payloads capable of providing manned landings on Mars, and for mission personnel complements of 8, nominally, or up to 20 persons.

An artist's conception of a two-vehicle convoy in earth orbit, being readied to depart for a manned surface exploration of Mars, is shown in Fig. 7. Each vehicle carries eight

persons plus 75,000 kg of destination payload for use while in Martian orbit. For personnel safety, each vehicle carries food and ecology expendables for a double personnel complement for 290 days, the trip duration remaining after the Mars approach maneuver. The total mission velocity requirement is 72,850 fps, based on a 1975 departure and a 450-day mission that terminates with a 50,000 fps earth approach velocity.

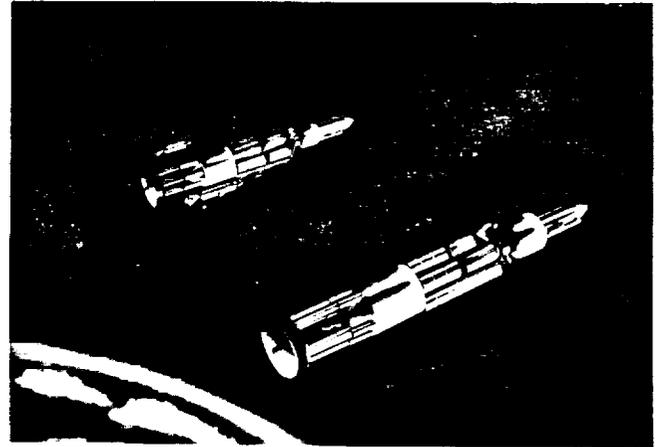


Fig. 7 — Two-vehicle convoy for Mars exploration

For the exploration mission depicted, each vehicle has an earth departure weight of 753,000 kg (1,660,000 lb). Eight successful rendezvous of Saturn V payloads are required to build up each vehicle of the two-vehicle convoy. For a single vehicle exploration (without the extra food and expendables) the departure weight is 741,000 kg (1,632,000 lb).

The effect of variations in the amount of destination payload and in number of exploration personnel is shown in Fig. 8. Each configuration employs the *same* basic 10-m propulsion module and operates through the same mission profile. The configuration shown in Fig. 7 is the center one in Fig. 8. The two 8-man configurations on the left carry as destination payload only 750 kg of mapping and data-storage equipment. Two 20-man configurations are shown on the right; one has a negligible destination payload, the other has 150,000 kg.

The effect of variations in mission velocity requirements is shown in Fig. 9. An 8-man, 450-day Mars operational payload and the small (750 kg) destination payload are carried in each vehicle. The configurations are identical except for variations in the reentry vehicle, when required, and in the quantity of propellant and propellant magazines. For the range of earth-orbit departure weights shown, four to nine successful Saturn V rendezvous are required.

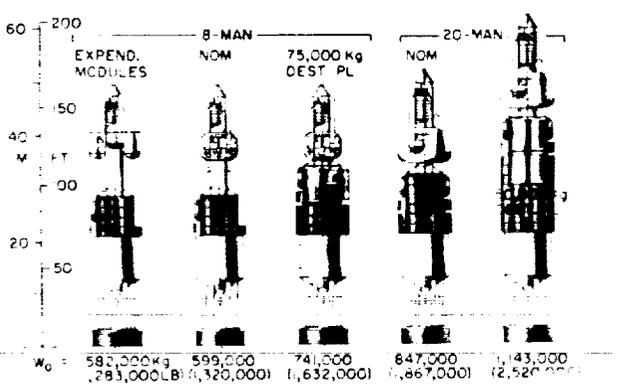


Fig. 8 - Variations in Mars-mission payload

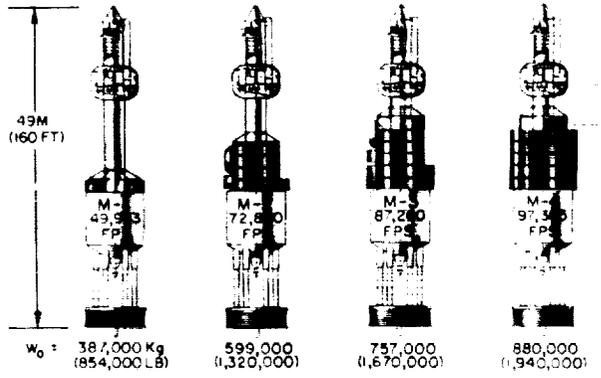


Fig. 9 - Variations in mission velocity

The range of mission velocities shown in Fig. 9 (based on the requirements of a 1975 departure) extends well beyond the range found in the cycle of "favorable" to "unfavorable" Mars departures. Departing on the most unfavorable year typically requires an 8% increase in departure weight, whereas the most favorable year permits a decrease of about 15%.

Direct-operating-cost comparisons and breakdowns for single-vehicle missions are shown in Fig. 10. The wider bars represent costs for the four mission velocities given in Fig. 9. The two narrower bars show cost differences for the M-2 case when carrying 75,000 kg of destination payload or when the personnel complement is 20 instead of 8. The latter comparisons indicate that useful payload could be more or less doubled at a direct mission cost increase of only one-third (presumably with little or no increase in the higher nonrecurring costs).

The earth-launch-vehicle (ELV) costs account for approximately 60% of total operations and the nuclear pulse pro-

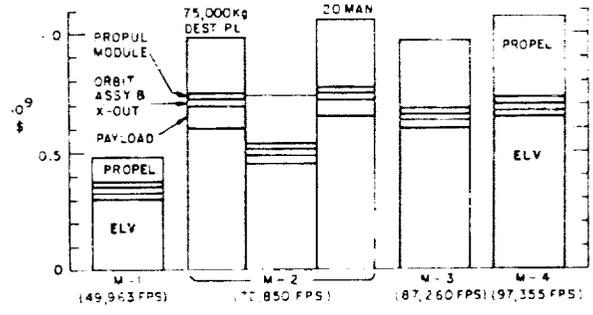


Fig. 10 - Mars-mission operating cost comparisons

pellant accounts for 20% to 30%. The comparatively small amount spent for the propulsion module indicates that it *should* be more expensive, if increasing its cost would improve the efficiency or weight so as to reduce propellant and/or boosting requirements.

The data of Fig. 10 include procurement of duplicate propulsion modules, payload sections, and redundant propellant and Saturn V operations to account for boosting, orbital mating, and loading probabilities. The probabilities used were 0.85 for successful orbital delivery, 0.97 for mating of propulsion module and payload section, 0.98 for loading propellant magazines, and 0.99 for loading destination payload. The over-all probability goal for orbital departure readiness was 0.75 or greater.

An abbreviated weight statement itemizing the make-up of the Mars exploration vehicle for both a Mars orbital and a Mars landing mission is given in Table 1.

Table 1  
MARS ROUND-TRIP WEIGHT STATEMENT

(8-man, 450-day, M-2 Mission)

	Orbital Mission (kg)	Landing Mission (kg)
Operational payload	80,000	80,000
Intransit payload	250	1,000
Destination payload	750	75,000
Total payload weight	81,000	156,000
Propulsion module	92,430	93,800
Propellant magazines	13,735	16,785
Guidance and start-up fluids	6,570	7,235
Propellant	405,265	467,180
Earth-orbit departure weight	599,000 (1,320,000 lb)	741,000 (1,632,000 lb)

In reviewing the Mars exploration performance capability of the 10-m nuclear pulse configuration, no attempt has been made here to make comparisons with other space propulsion systems. It is believed that in numerous respects—e.g., single-



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stage round-trip capability, modest orbital departure weights, payload versatility, relative insensitivity to mission year or mission profile, etc.—the merits of the system for Mars, Venus, and other early explorations will be recognized.

### Lunar Transport Systems Performance

Applications of the 10-m propulsion module to both a reusable lunar ferry and one-way expendable logistics vehicles were investigated. Large transport capabilities, in comparison with today's lunar base planning, are indicated. Here, again, the operating costs are dominated by the earth-launch-vehicle contribution.

*The Lunar Ferry.* An earth-orbit-to-lunar-orbit reusable ferry appears to be the better utilization of the nuclear pulse system for lunar base support. Figure 11 is an artist's conception of such a lunar ferry in operation, where passengers are transferring to lunar shuttles for descent to the lunar surface. The configuration shown carries 20 passengers per

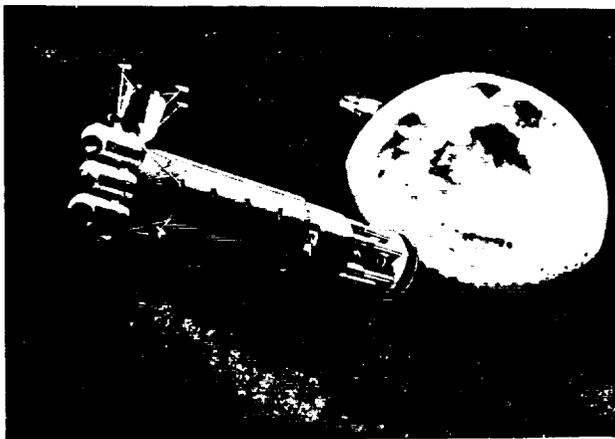


Fig. 11 — Nuclear pulse lunar ferry system

trip plus 200,000 kg (440,000 lb) of cargo for the lunar surface plus propellant for the lunar shuttle vehicles.

For most of the trip, the passengers occupy the passenger modules attached to the central powered-flight-station—command module. During powered flight, radiation-belt penetration, or solar storms, they would remain within the more compact, shielded, powered flight station.

The lunar-surface logistic capability of the above system is shown in Fig. 12. The delivered payload is seen to be relatively linear with departure weight, which indicates a wide range of payload capability per trip without greatly changing the delivery cost effectiveness (since delivery costs are largely a function of the mass required in orbit).

Preliminary estimates of direct operating costs for all elements of the system have been compiled and are summarized in Fig. 13. The cost estimate assumes a maximum stay time of 6 months for lunar-base personnel and a support requirement of 4,000 lb per man-year. The costs level off at about \$7.5 million per man-year as the base size reaches several hundred men.

Figure 13 also shows a percentage breakdown of the operating costs, indicating that the two components that need minimizing are the earth-launch-vehicle operations and the

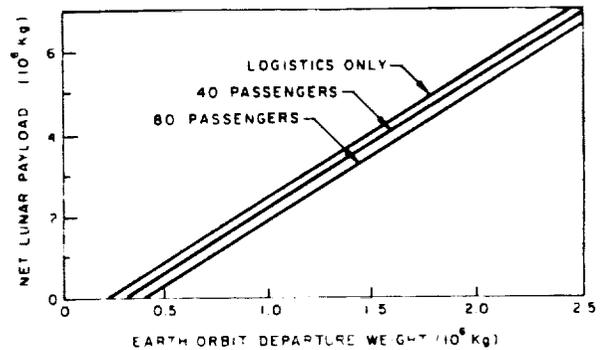


Fig. 12 — Lunar-base-support capability

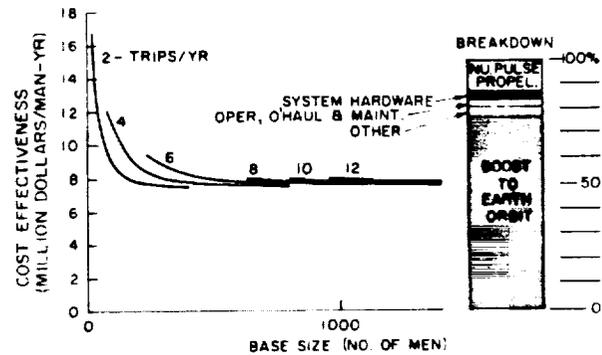


Fig. 13 — Lunar-base-support operating cost per man year

nuclear pulse propellant. A reoptimization of the nuclear pulse propulsion parameters, for the lower velocities characteristics of lunar missions, may do much to reduce the latter. This reoptimization has yet to be performed.

*One-way Lunar Logistics Vehicles.* Two methods of employing nuclear pulse vehicles expendably offer system gains in one-way logistic delivery to the lunar surface. One employs nuclear pulse propulsion from earth orbit to near the lunar surface and the other from near the earth (after lofting to about 10,000 fps by a Saturn S-1C stage) to near the

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lunar surface. Both use chemical-rocket deceleration for the last 2,000 fps of lunar let-down.

A typical configuration for the earth orbit start-up mode of operation is shown in Fig. 14. With an earth-orbit departure weight of 1,800,000 kg (3,960,000 lb), this vehicle could deliver a payload of about 850,000 kg (1,870,000 lb) to the lunar surface. A similar but smaller vehicle, using an S-1C first stage, in a direct flight from the earth's surface could deliver approximately 67,500 kg (143,000 lb) to the lunar surface. A preliminary cost analysis indicates delivery costs of about \$1500/kg for the earth orbit start-up mode

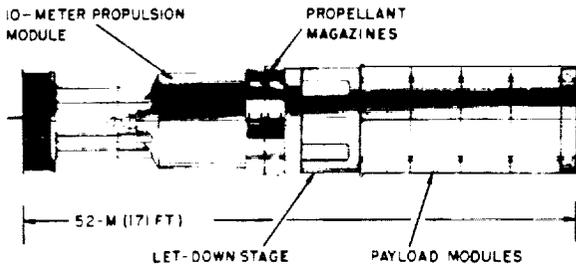


Fig. 14 — Orbital start-up lunar logistic vehicle

and about \$1600/kg for the direct-flight mode. These are about one-third the delivery costs when using an all-chemical Saturn V.

#### LARGER VEHICLE CAPABILITY AND GROWTH POTENTIAL

Two approaches can provide marked increases in system effectiveness beyond that shown for the 10-m configurations. The first, with the same state of the art as the 10-m designs, will capitalize on the increased specific impulse and reduced propellant cost per kilogram for larger vehicles. This approach is pictured in Fig. 15, where a 20-m diameter exploration vehicle is shown starting a maneuver to orbit Callisto, a moon of Jupiter. The 3,150-sec  $I_{sp}$  of this vehicle will permit, for example, a single-stage round trip to Jupiter, with a 20-man complement and 100,000 kg of destination payload. Such a mission will require a total mission velocity of 192,000 fps and a duration of 900 days. In Fig. 16, the conceptual design of a 20-m vehicle is shown in comparison with a 10-m Mars configuration.

The second approach to improve system effectiveness exploits more fully the potential of nuclear pulse propulsion. Pulse units having nuclear explosion devices designed specifically for propulsion purposes will be utilized, thus significantly improving propellant economics even for small vehicles. Advanced materials for propulsion module fabri-

cation will also be utilized. For the post-1980 era, a specific impulse between 10,000 and 20,000 sec is predicted for vehicles somewhat larger than the 20-m vehicle depicted here.

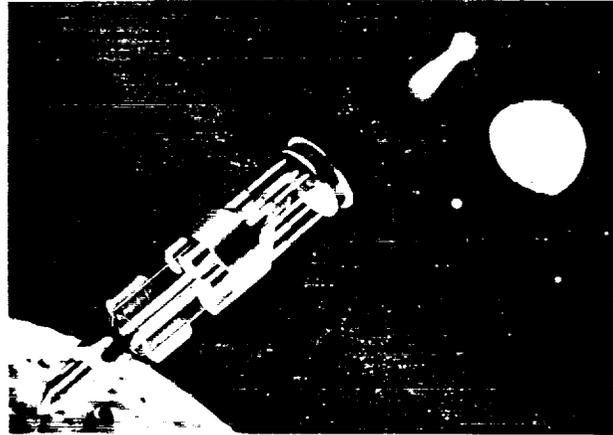


Fig. 15 — Jovian exploration vehicle

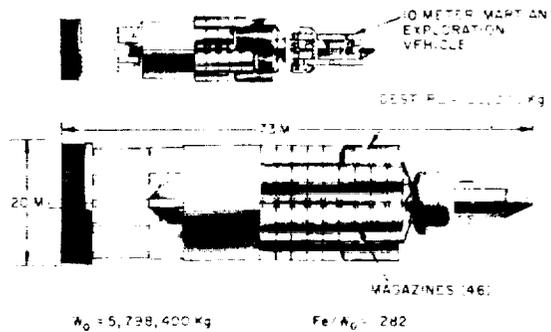


Fig. 16 — A 20-meter exploration configuration

#### OPERATIONAL PROBLEM CONSIDERATIONS

The more critical operational and hazards problems that are unique to nuclear pulse propulsion have been reviewed in relation to the launch complex, the earth's environment, the flight crew, and mission objectives to determine whether any fundamental characteristics might preclude or seriously constrain a major operational employment of nuclear pulse systems. These studies included various credible systems failures and resulting hazards, eyeburn to population, fission-product dispersal in the atmosphere, nuclear radiation levels during operations, creation of artificial electron belts, vehicle acoustical levels, and nuclear materials availability. Under the operational modes, constraints, and safeguards suggested, no issue under investigation to date appears to preclude





large-scale employment of nuclear pulse systems for space exploration and transportation. Potential hazards to the earth's environments and population may not, in fact, be significantly different from those incurred by use of all-chemical systems.

#### DEVELOPMENT PLAN

Two new key research and development program elements are envisioned: (1) relatively inexpensive, contained, nuclear underground experiments to permit recovery of scaled to full-scale engine prototypes, and (2) repetitively placed high-explosive shock generators to attain time and spatial reproduction of the mechanical impulse received by the propulsion module at *full scale* and at the proper *firing rate*. These, plus other supporting techniques, permit a phased ground-based development program leading in well-defined stages to a Preliminary Flight Rating Test (PFRT) propulsion module for ballistic and finally orbital mission qualification. Preliminary studies indicate that the development program through orbital qualification should cost approximately \$2 billion and require 12 years. Major program landmarks accrue at about 3-year intervals.

The fact that the same developmental techniques may later be used to prelaunch-qualify operational propulsion modules prior to commitment to manned space missions is of considerable operational importance.

#### 6. STUDY LIMITATIONS

A major limitation of the study has been the lack of design data on small propulsion modules or on modules designed for orbital start-up. The findings of this study on the mission applications and capability of the smaller (10-m) modules has accentuated this limitation.

A secondary limitation, after determining the desirability of lower thrust-to-weight ratio start-up, has been the lack of time and data to "optimize" the orbital start-up operations. Several factors need to be considered simultaneously: gravity losses caused by reduced thrust-to-weight ratio, electron trapping (latitude) limitations, "multiburn" earth escape, launch-to-orbit ground-safety limitations, and coolant savings ( $I_{sp}$  gain) due to reduced thrust-to-weight ratios. In the current study, such optimization was of necessity neglected, again making the study results conservative.

A third limitation has been the limited depth of coverage of systems economics. There are many economic trade-offs that need exploration to understand and optimize the system parameters. This study has pointed out the importance of propellant economics and the relative unimportance of propulsion module hardware cost in the smaller thrust designs.

(Thus, more should be spent on hardware if a reduction of mass or in propellant consumption, etc., can be realized.)

#### 7. IMPLICATIONS FOR RESEARCH

Technical feasibility of nuclear pulse propulsion now seems reasonably assured; the major elements of a ground-oriented development plan have been identified and appear practical; current application studies confirm a revolutionary improvement in space operations if pulse propulsion can be developed and utilized. However, as in any new technology, a significant amount of basic knowledge and understanding should be generated, as well as a compelling reason for being, before any large-scale development effort is undertaken.

The immediate research objective, therefore, is to augment present technical knowledge in the more sensitive areas so that continued successful progress may lead rationally to a stepwise development program, provided, of course, that the payoff remains as relatively attractive as it does now. A well-balanced research program to attain this objective should contain, at least, the following major elements:

*Design studies*, including pulse-unit designs and ablation calculations, to evolve optimum propulsion module designs compatible with vehicle sizes and operational modes now of interest to the National Space Program.

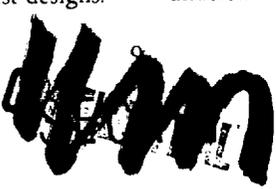
*Underground nuclear tests and continued HE ablation experiments* to correlate ablation and other early-time interaction phenomena with calculations, under conditions of propellant velocities, temperatures, pressures, densities, and scales relevant to operationally sized vehicles. Additional underground nuclear tests to establish pulse-unit specific impulse.

*High-explosive dynamic tests* (nonnuclear) of critical structural elements under conditions which simulate mechanically the impulse delivered to the vehicle by the pulse unit, again at scales relevant to usefully sized vehicles.

Completion of research programs in these areas, possibly within three years, would confirm in considerable detail the performance, structural integrity, and operational reliability of early nuclear pulse vehicles. A strong indication of growth potential would also be ascertained. These data, acquired at modest cost, would provide a sound technical basis for any further course of action.

#### 8. SUGGESTED ADDITIONAL EFFORT

The current technical and programmatic status now supports consideration of additional effort directed toward better understanding both the opportunities and special implica-





tions of nuclear pulse propulsion in a context responsive to an ever-changing National Space Program. The more pertinent or limiting technical considerations must certainly receive additional attention. In addition, formal effort, primarily by appropriate government agencies, should be concerned with certain policy issues confronting national use of nuclear pulse systems. Specifically, the following areas are suggested as now justifying increased attention.

- *Propulsion Module Design Studies*

Integrated design studies should be undertaken to provide additional technical support or modifications to the conclusions reached herein. Particular emphasis should be given the Saturn V compatible propulsion modules.

- *Applied Research Programs*

Analytical and experimental effort in the key technical problem areas should be expanded. Specifically, a series of small, underground, nuclear experiments and HE-driven dynamic response experiments should be initiated.

- *Mission Applications Studies*

Increased effort should be undertaken to place nuclear pulse propulsion capability more clearly in a mission-oriented framework, to establish operational criteria, to evaluate hazards, to understand better the over-all economic advantages of such space propulsion capability, and to indicate how and when such a capability might best be integrated into the space program.

- *Development Planning*

A comprehensive development planning effort, with joint participation by government agencies and contractors, should assess in detail the elements of the development plan proposed in this report. Proposed facilities scheduling and time-integration into other elements of the space program should be established.

Required resource availabilities, such as boosters, launch facilities, and special nuclear materials, should be specified.

- *Nuclear-source Design*

The design, fabrication, and testing of the nuclear energy sources in the pulse units fall within the province of the AEC. To date, all project technical studies have considered only nuclear devices currently stockpiled or those with principles well established. However, there is a certainty of achieving substantial improvements in performance and mission economics if the nuclear devices were specifically designed to satisfy nuclear pulse vehicle requirements. It is therefore suggested that the AEC, working closely with the propulsion-system specialists, consider initiation of design studies for this purpose.

- *Nuclear Test Ban*

Current wording of the Nuclear Test Ban Treaty precludes the final development and/or operation of nuclear pulse vehicles. However, the professed *intent* of the agreement is basically to limit the arms race and to prevent further major contamination of the earth's environment. Nuclear pulse propulsion systems violate neither of these criteria, but their use may be prevented by the wording of the treaty language. Provided a strong mission requirement can be shown for their development, a substantial effort will be needed to advance such arguments clearly so as to obtain the necessary adjustments in the treaty language. It is therefore suggested that, at the appropriate time, a joint study be undertaken to attempt resolution of this issue. Although nuclear testing in space would not in any event be required for some six to eight years, some indication that development could be accommodated is certainly desirable prior to large-scale developmental funding.

# TECHNICAL MEMORANDUM X-53200

## ADVANCED POST-SATURN EARTH LAUNCH VEHICLE STUDY EXECUTIVE SUMMARY REPORT

### SUMMARY

The purpose of this report is to summarize the results of a study of advanced (chemo-nuclear) Post-Saturn launch vehicles. By investigating vehicles beyond the purely chemical systems, a safeguard is provided against the danger of being caught short relative to future extraterrestrial operational requirements.

A number of concepts were investigated from three points of view: design, vehicle-engine integration, and performance. After an extensive process of evaluation and selection, the most promising concepts were studied in greater detail.

The results indicate that gas core reactor and nuclear pulse engines are both attractive for the advanced Post-Saturn vehicle, and both should be investigated further. If emphasis is on Earth orbit and lunar delivery missions, the gas core reactor shows a slight advantage. The nuclear pulse concept is clearly preferable if emphasis is on lunar and planetary deliveries.

### SECTION I. INTRODUCTION

In order to make adequate long range plans, it is necessary to study at this time various missions and their requirements because of the long leadtime associated with the development of advanced space transportation systems. The study summarized in this report is one of a series to assist in the selection and definition of the next large launch vehicle after Saturn V.

The key to extraterrestrial operations is the development of an adequate Earth to orbit logistics system, of which the Earth launch vehicle is the primary element. The largest of the present Earth launch vehicles under development, i. e. , Saturn V, opens Earth orbit to manned operations on a large scale, but is not economically satisfactory from a long term standpoint. It barely extends manned flight to the Moon for small crew excursions. With further improvements

in payload capability, the Saturn V could be used to build-up and maintain a small (10 people) lunar base, but the cost would be high. It could also be used for minimum level manned interplanetary expeditions, such as Mars and Venus fly-bys, but would require extensive orbital operations.

If the national space program is going to continue at the present funding level for the next two decades, it is likely that new launch vehicles superior to Saturn V, in terms of payload capability and cost effectiveness, will be developed. This Post-Saturn Earth launch vehicle is the key to extended economical manned exploration and exploitation of the inner solar system. This study investigated the very advanced concepts of Post-Saturn launch vehicles and in particular their compatibility with advanced nuclear upper stages.

The study reported here was conducted on a parallel basis by the Douglas Aircraft Company (Contract NAS8-5021) and General Dynamics/Astronautics (NAS8-5022). The funding for the two contracts amounted to \$ 449,000 for Douglas and \$ 382,824 for General Dynamics/Astronautics.

The purpose of this report is to provide executive personnel with a condensed description of the study. The report describes the study objectives, discusses the approach, provides some results and conclusions, and recommends future work. If more details are desired, refer to References 1 and 2 listed at the end of this report. These documents can be obtained from the Scientific and Technical Information Division, Code ATSS-A, National Aeronautics and Space Administration, Washington, D. C. 20546.

## SECTION II. SCOPE AND OBJECTIVES

The subject of this study is Advanced Post-Saturn (Class III and IV) Earth launch vehicles. Since the date of development initiation for the Post-Saturn is not known, it is necessary to study several classes of vehicles. In order to put this study (Class III and IV vehicles) in the proper perspective, the four classes of Post-Saturn launch vehicles are defined below:

1. Class I represents current technology, i. e. , expendable stages using propulsion systems that are currently available or under development.
2. Class II represents advanced technology, i. e. , a reusable first stage with advanced propulsion.

3. Class III represents very advanced technology, i. e. , reusable single stage to orbit with advanced propulsion.

4. Class IV is a chemo-nuclear launch vehicle with very advanced technology in both the chemical and nuclear stages.

Thus, within the broad spectrum of Class III and IV vehicles, the objectives for this study were performance of the following tasks:

1. Identification of areas of greatest potential improvement through basic analysis and reviews of previous nuclear reactor technology and launch vehicle study programs.

2. Develop criteria for realistic comparison of concepts.

3. Develop summary descriptions of vehicle concepts considered along with estimated capabilities and characteristics peculiar to the concept.

4. Make comparisons leading to selection of concepts recommended for more detailed investigation.

5. Perform a conceptual design and analysis of an attractive vehicle concept selected with MSFC approval.

6. Perform compatibility studies of recommended concepts with the Post-Saturn Class III booster, emphasizing lunar and planetary missions.

7. Analyze procedures, cost, and performance parameters influenced by the desire to recover and reuse the chemical boost stage of the chemo-nuclear launch vehicle under consideration.

This study was limited by the relatively small amount of factual information available on nuclear engines, particularly the gaseous core concept. Performance and weight information are only reasonable estimates. Also, a limitation in evaluating the vehicles is the present lack of plans regarding future extraterrestrial operations. Thus, the lack of factual information and planning were limitations in performing conclusive schedule and cost analyses.

## SECTION III. METHOD OF APPROACH AND ASSUMPTIONS

### A. GENERAL APPROACH

This study concentrated on chemo-nuclear Earth launch vehicle concepts as outlined below:

1. Two-stage system with recoverable chemical first stage and solid core reactor nuclear second stage.
2. Two-stage system with recoverable chemical first stage and gaseous core nuclear second stage.
3. Two-stage system with recoverable chemical first stage and nuclear pulse second stage.
4. Modular chemical and nuclear propulsion.
5. Single-stage vehicle with separate chemical and nuclear engines structurally integrated.
6. Vehicle powered by nuclear engine with chemical afterburning for thrust augmentation and radiation level reduction in lower atmosphere.

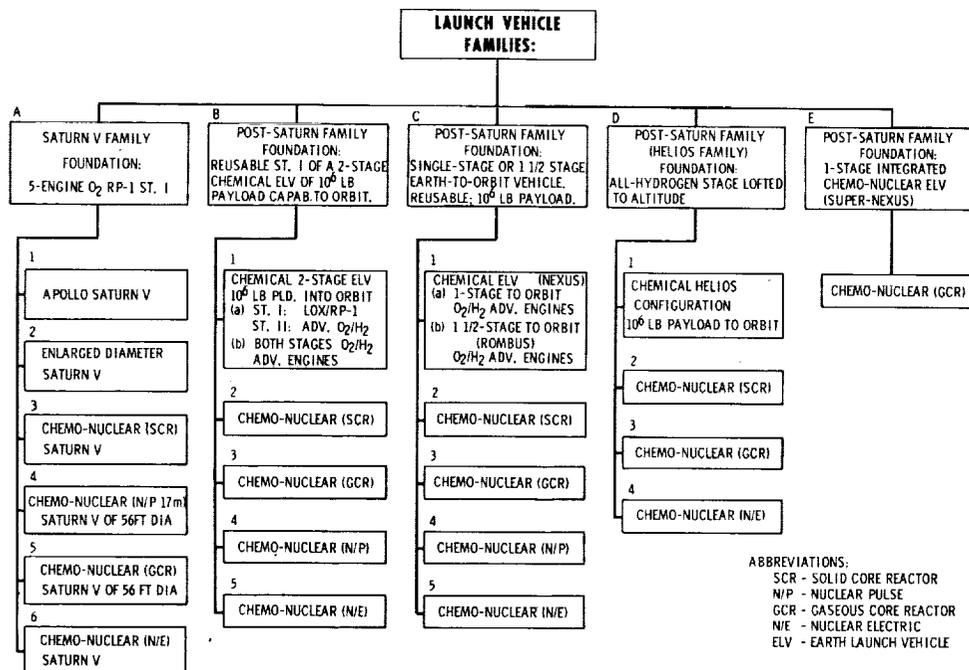


FIGURE 1. SURVEY OF VEHICLE TYPES CONSIDERED IN THIS STUDY

Initially, a number of concepts (Fig. 1) were investigated from three points of view: design, vehicle-engine intergration, and performance. An extensive system of evaluation criteria was developed, and vehicle configurations were analyzed as design points to provide a basis for parametric variation. The following parameters were varied on each vehicle type considered: velocity, thrust, propellant weight, ignition weight, structural configuration, and number and type of engines. Every attempt was made to keep the weight analysis realistic.

To avoid investigation of unfeasible configurations, particular emphasis was placed on constraints nuclear engines might pose because of clustering and operational characteristics. For sea retrieval operations, the system was defined as consisting of recoverable booster, ground support equipment, facilities, supporting personnel, and documentation.

A launch complex study was carried out for the most important configuration. Also, the schedules and cost aspects of development and operation were established and compared.

After an extensive process of evaluation and selection, the most promising vehicle types were selected and studied in greater detail.

## B. EVALUATION CRITERIA

A system of evaluation criteria (five groups) was established and is outlined below (see Reference 2 for evaluation results):

1. Mission Worth. This area integrates the effect of all important mission-related considerations such as: applications, performance sensitivity, mission versatility, operational cost effectiveness, reusability, growth potential, etc. A high mission worth rating signifies:

a. Satisfactory probability that original mission objectives can be carried out.

b. Satisfactory adaptability to limited modifications or extensions of original mission objectives.

c. Satisfactory operational life of the individual vehicle and economy.

d. Satisfactory operational life span of the "species."

2. Service Reliability. This area integrates all important indices that affect the reliability of the Earth launch vehicle system such as: number of events required for mission success, vehicle mission reliability, sensitivity to launch and flight malfunction, flexibility of launch rate, etc. A high service reliability rating indicates:

- a. Satisfactory vehicle reliability.
- b. Satisfactory confidence level that a projected critical launch rate can be maintained if malfunctions occur.
- c. Satisfactory confidence level that the launch rate can, readily and economically, be varied within given limits derived from uncertainties in present estimates of extraterrestrial supply and supply rate demands; and that the launch rate can be varied as projected from extraterrestrial activities and launch vehicle capability.

3. Operational Availability. This criterion integrates the effects of development aspects that can be programmed, and those which are difficult or impossible as yet to program because the component or process is still in the applied research phase. These aspects include: principal development milestones, schedule uncertainties, and growth potential availability. A high rating in this area signifies:

- a. Vehicle operational capability matches desired time schedule of progress in those activities which have to rely upon the supply capability of the respective logistic system.
- b. Schedule uncertainties are likely to be within the tolerance of the evolution of extraterrestrial supply requirements.
- c. Satisfactory confidence level that both possibility and probability of growth potential state of the art matches the projected increase in space logistic requirements.

4. Operational Characteristics. This criterion integrates all significant indices of operational aspects which contribute to the evaluation of particular vehicles and logistic concepts. These are areas such as: noise; critical propellant characteristics; radiation; thrust vector control; engine-out sensitivity; launch facilities and operations; operations during ascent, descent and orbital coast; recovery; and refurbishing. A high rating in this category indicates:

- a. No significant local nuisance aspects.
- b. Satisfactory local safety aspects for surrounding areas.
- c. Satisfactory global safety aspects.
- d. Satisfactory vehicle control aspects.
- e. Satisfactory routine operational aspects.
- f. No excessive requirements on construction or location of launch sites or facilities.

5. Development Characteristics. This criterion integrates all significant aspects associated with development of the launch vehicle and associated logistics system. This includes: critical development items; state of the art continuity; value analysis; facility and GSE requirements; special test problems; development funding; and growth development problems. A high rating in this area signifies:

- a. Satisfactorily low development risk.
- b. Objective evaluation from the overall program standpoint of local improvements or marginal applications of the state of the art which may be attractive in a special area, but whose benefit to the overall program is not obvious.
- c. Satisfactorily low development cost and absence of extreme funding rate requirements.

#### SECTION IV. BASIC DATA GENERATED AND CONCLUSIONS

The following paragraphs will present only some of the highlights of the data generated in this study. If more details are desired, refer to the references at the end of this report.

##### A. MISSION ANALYSIS

The various mission modes involved in space transportation were defined because they represent one important input into the orbital, lunar, and

interplanetary mission model analysis. These mission modes and the analysis are discussed in the subsequent paragraphs.

Delivery of payload to an extraterrestrial destination can be accomplished either by direct delivery to the departure orbit or a two-step mode via a waiting orbit. For the latter, the Earth launch vehicle may carry the entire inter-orbital space vehicle (and possibly a separate orbit launch vehicle as the upper stage to serve as orbital booster) into parking orbit. This is referred to as the direct flight mode (DFM). Alternately, the interplanetary space vehicle may have to be assembled in orbit, which is the orbital vehicle assembly mode (OVAM).

Post-Saturn Class IV type vehicles do not necessarily have to be restricted to Earth orbital delivery. First, there is the deep space injection mode, ranging from parabolic to hyperbolic injection of lunar and interplanetary interorbital space vehicles. With enough energy, recovery of the Earth launch vehicle for reuse from orbit or the surface is feasible but expensive. Hyperbolic rendezvous with a returning manned interplanetary vehicle is a risky but possible mission of this type. Secondly, the operational range of some Earth launch vehicles can be extended to the Moon and nearer planets.

For individual missions and for initiation of shuttle service, the direct flight mode is the most economical method in comparison to the orbital vehicle assembly mode. The direct flight mode displays the greatest mission sensitivity, because its requirements are such that the Earth launch vehicle is neither too small or too large for the majority of the interorbital space vehicle stages, which as part of the Earth launch vehicles are launched into parking orbit. Of course, the orbital vehicle assembly mode displays the least sensitivity in that respect.

Mission analysis led to the definition of energy plateaus, which can serve as guides for standardized Earth launch vehicle performance requirements. Orbital and planetary mission models were developed, from which lunar mission models were derived. A summary chart, which correlates ideal velocity, payload fraction, mass ratio, and energy plateaus, is presented in Volume I of Reference 2.

Because of the importance of the direct flight mode, a series of departure weights of the interplanetary vehicles was determined (Fig. 2). These vehicles use various advanced propulsion systems considered available during the operational life of the chemo-nuclear Post-Saturn launch vehicle. A total of 21 cases were considered, of which 19 are manned space vehicles and 3 are instrumented probes. It was found that with an orbital payload of  $10^6$  lb, 2 of the instrumental

probes and 4 of the 19 manned vehicles considered (20 percent) could be delivered into orbit as complete, fueled vehicles. At  $1.5 \times 10^6$  lb orbital payload capability, the third instrumented probe and 13 out of 19 manned vehicles (68 percent) could be delivered in operational condition; at  $2 \times 10^6$  lb the number increased to 15 (79 percent). Although these numbers are not conclusive, they do indicate that an orbital payload capability of  $1.5$  to  $2.0 \times 10^6$  lb may reduce the obsolescence rate of the chemo-nuclear Post-Saturn vehicle as far as long range planetary mission capabilities are concerned.

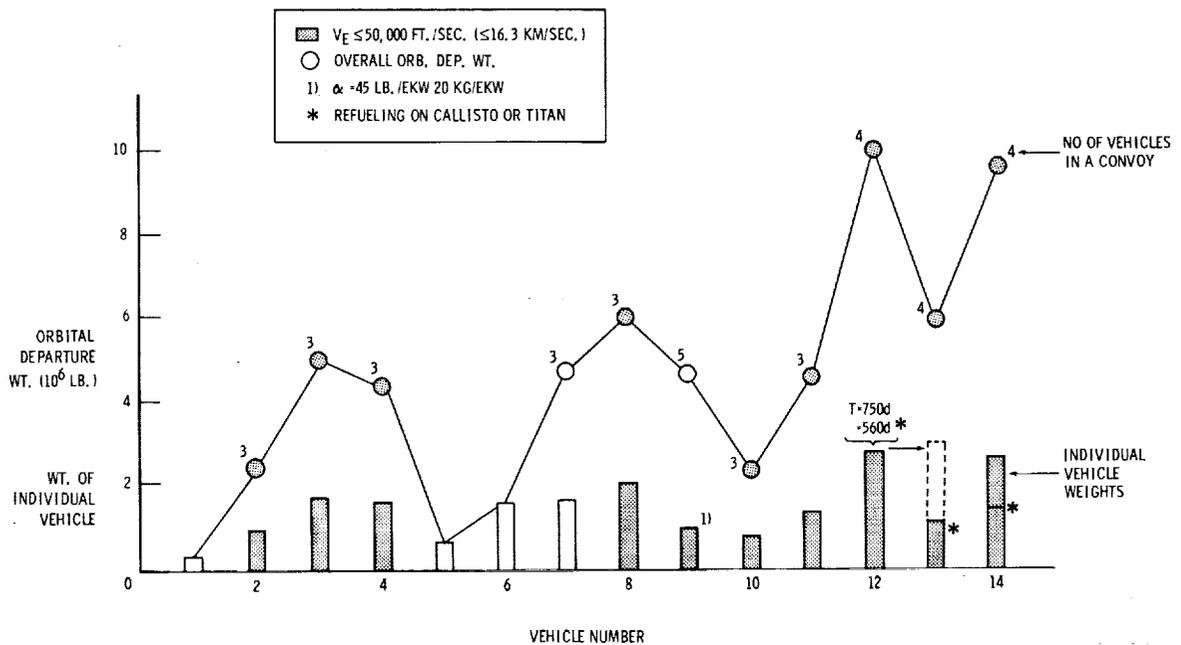


FIGURE 2. WEIGHT OF INTERPLANETARY VEHICLES AND OVERALL ORBITAL DEPARTURE WEIGHT

Three manned planetary mission models were established and it was found that for the period 1981 to 2000, the minimum number of successful direct flight mode deliveries may range from 18 (conservative), to 26 (intermediate), to almost 76 (optimistic at present rate of progress). If direct flight mode is replaced by orbital vehicle assembly with two launches per orbital vehicle, the number would double. These figures refer to successful launches. Reliability analysis shows that if the probability of successful orbit delivery is 0.75, it takes procurement of 3 sets (Earth launch vehicle and interorbital space vehicle) to accomplish 2 deliveries with 75 percent success, and of 4 sets (i. e., potential

launches) to accomplish 3 deliveries with 75 percent probability, provided the interorbital space vehicles are identical. If they are not identical, then 4 and 5 sets are required to deliver 2 and 3 vehicles with 75 percent probability. If the success probability of individual delivery is 85 percent and if the overall probability of success remains 75 percent, then 2 deliveries require 2 procurements, 3 deliveries require 4 sets procured if the interorbital space vehicles are identical; or 3 and 5 sets, respectively, if they are different. This shows that even for the direct flight mode throughout the entire period, at least 25 percent more than the minimum number of launches is needed, raising the total to at least 23, 33, and 94 launches in 20 years.

A graphic representation was developed of the relation between minimum number of launches and actual number (at least the number of procurements) as a function of probabilities for the cases of identical and different interorbital space vehicles. On the basis of these data, it was found that a suitable parametric range of planetary launches between 1981 and 2000 is 60 to 180, or, on the average, between 2 and 9 per year at an average delivery of  $1.5 \times 10^6$  lb into orbit. Although the average number of flights does not appear to be high, the number is initially lower and, in the 1990's, considerably higher.

With respect to payload capabilities of the vehicle concepts investigated, the following figures are considered representative:

The payload soft-landed on the Moon is 30 and 50 percent of the orbital payload in the case of gas core reactor propulsion with 1500 sec specific impulse. The payload fraction is higher still with nuclear pulse stages (specific data are classified). Compared to this, the payload soft-landed by a 3-stage,  $O_2/H_2$  Earth launch vehicle is 16 percent of orbital payload. For chemo-nuclear vehicles, therefore, the same number of launches delivers between 30 - 45 and 90 - 145 million pounds to the lunar surface. For lunar bases employing 10 to 100 persons, supply requirements of 100,000 to 250,000 lb per annum was derived for the base equipment, and 33,000 lb per man-year for the crew subsistence. Thus, even a 100-man base requires a supply of about  $3.5 \times 10^6$  lb per annum. For a chemo-nuclear vehicle of  $1.5 \times 10^6$  lb orbital payload and 600,000 lb lunar soft-landed payload per flight, this means 6 successful flights or 9 potential launchings per annum, if the delivery probability to the lunar surface is 85 percent, and the probability that 6 vehicles reach the Moon's surface is to be 90 percent. Therefore, by preparing the same number of launchings for the lunar base as for the planetary missions, we could establish and maintain a lunar base growing from 30 to 100 people in the 1981 to 2000 period if a chemo-nuclear (gas core reactor) Earth launch vehicle is used (1500 sec specific impulse). It is concluded that larger lunar bases can be established and maintained with the nuclear pulse vehicle efficiently.

## B. VEHICLE DESIGN AND WEIGHT ANALYSIS

Over 20 vehicles of the Post-Saturn family, with emphasis on nuclear upper stages, were investigated (Fig. 1). The more important configurations consist of the following vehicle groups:

1. Chemo-nuclear Earth launch vehicle using nuclear pulse engine in the upper stage.
2. Chemo-nuclear Earth launch vehicle using gas core reactor propulsion in the upper stage.

Solid core reactor systems show considerably less promise because of high sensitivity of payload weight to changes in mass fraction, ideal velocity, and specific impulse, and because of operational problems and the long cool-off periods required in orbit. Thus, solid core reactor vehicles are only discussed for comparison purposes. Also, nuclear pulse vehicles cannot be discussed in this summary because of security considerations.

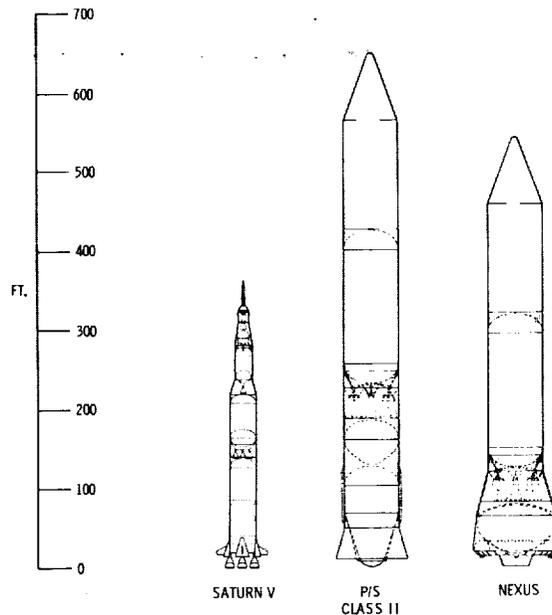


FIGURE 3. POST SATURN CLASS IV W/M NUCLEAR STAGE ON BASELINE AND NEXUS BOOSTERS (Eight 750 K Water Moderated Metal Core Engines)

Figure 3 shows a solid core reactor (water moderated) nuclear stage mounted on a Post-Saturn Class II baseline vehicle, and on a tailored Nexus first stage. The vehicles are sized for orbital delivery, and the booster stages are off-loaded to about 75 percent capacity. This places staging velocity at approximately 10,000 ft/sec. The vehicles are all 70 ft (21.3 m) in diameter. The Nexus has a length/diameter ratio of about 7.8, whereas the Post-Saturn Class II vehicle length/diameter ratio is 9.3. The Post-Saturn Class II is 650 ft tall whereas the other two measure about 550 ft each. The nuclear stages of 5.45 million lb each place nearly 1.5 million lb into Earth orbit. The launch weights are approximately 14.4 million lb for all three versions. Payload fractions are 0.102.

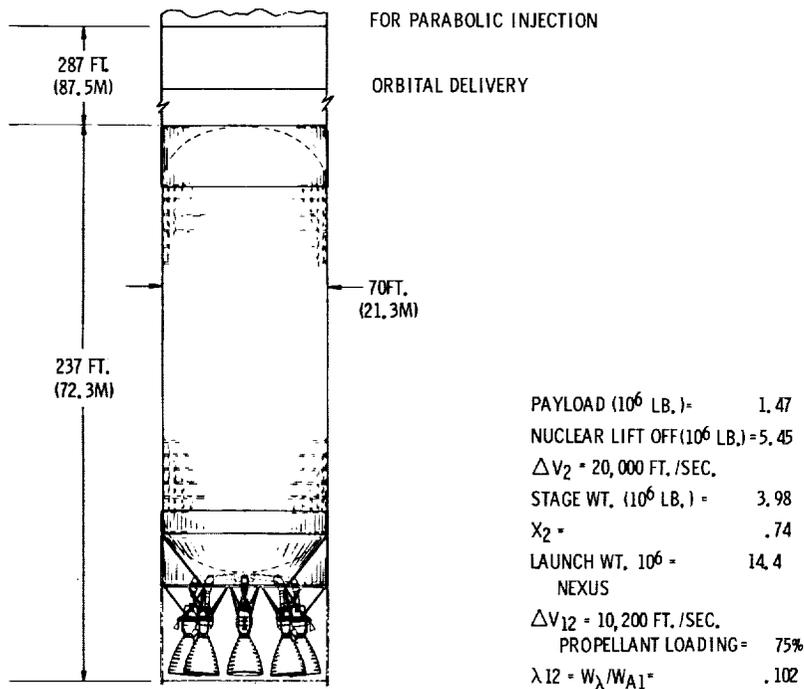


FIGURE 4. POST-SATURN CLASS IV NUCLEAR STAGE (Eight 750 K Water Moderated Metal Core Engines)

Some details of the solid core nuclear stage are shown in Figure 4. This type of stage has a great number of technical and operational problems, which are discussed in detail in the references.

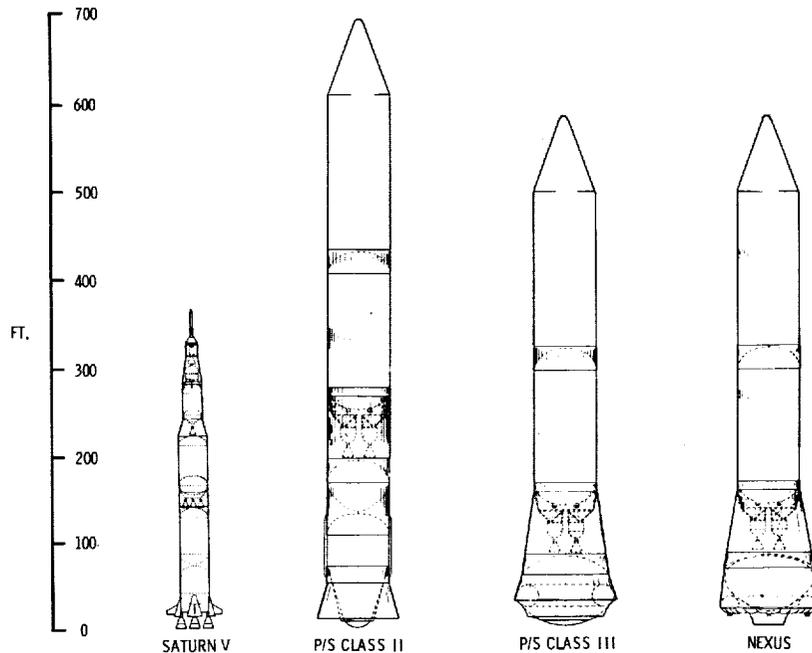


FIGURE 5. POST-SATURN CLASS IV GCR NUCLEAR STAGE ON NEXUS BOOSTER

Figure 5 shows a gas core reactor installation on the Post-Saturn baseline vehicles. Represented is a nuclear stage with two 3000K gas core reactors. The baseline vehicles are staged at approximately 10,000 ft/sec and delivers 1.96 million lb to parabolic injection. Some details of gas core reactor stages are shown in Figure 6. The two 3000K engines are mounted with their lines of thrust passing through the vehicle center of gravity. This scheme allows variations in thrust between the two engines without having a penalty in the vector control system. This system operates from bleed from the main engine through small jets.

Figures 7 and 8 show Post-Saturn family vehicles derived from an initial chemical one-stage to orbit version. In these two cases, Nexus was used as the model because it offers a broader base for mounting the heavy upper stages.

Considerably emphasis was placed on the use of gas core reactor and nuclear pulse powered upper stages. This was done after an extensive study of solid core reactor engines of the graphite as well as the metal core type leading to a deemphasis of this particular approach.

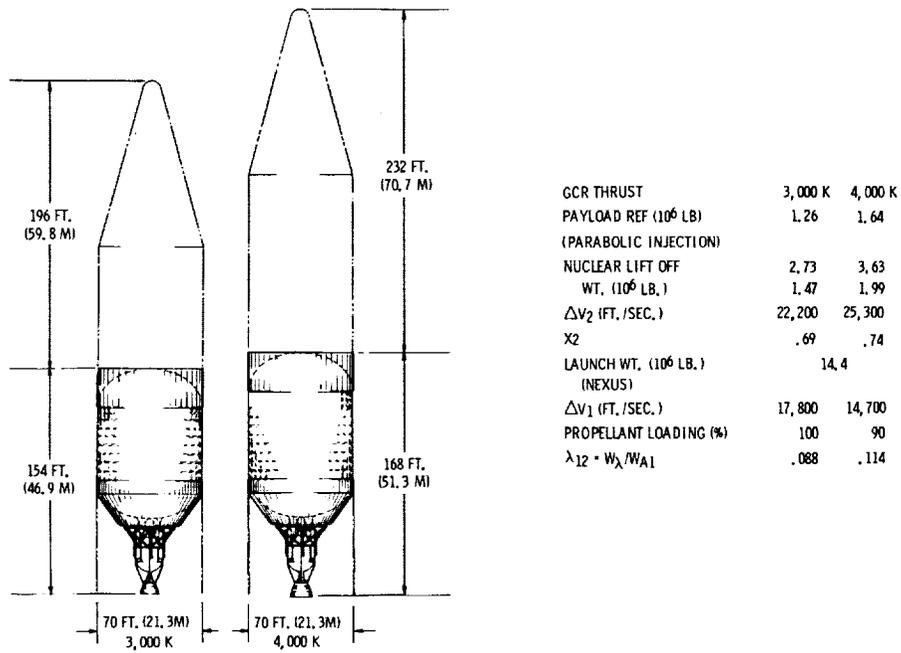


FIGURE 6. POST-SATURN CLASS IV NUCLEAR STAGES  
(3000 K and 4000 K GCR Engines)

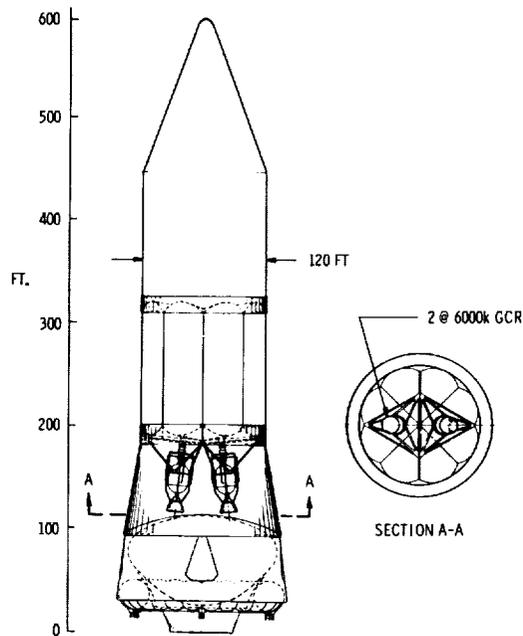


FIGURE 7. GCR (2 at 6000 K) NUCLEAR STAGE ON UP-RATED NEXUS

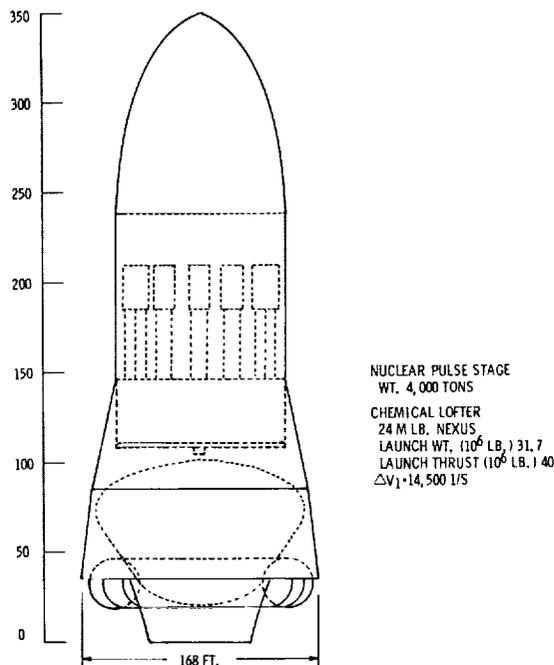


FIGURE 8. NUCLEAR PULSE VEHICLE WITH NEXUS LOFTER

Briefly, the conclusions from the comparison of the propulsion systems are:

1. Solid core reactor systems (graphite) in the thrust range of 250 to 900K impose severe limitations on the Earth launch vehicle design if applied to vehicles with  $10^6$  lb or more orbital payload capability.
2. Solid core reactor (metal) engines (namely, the water-moderated slow to medium fast neutron spectrum engine) is distinctly superior to its graphite counterpart in the same thrust range.
3. All solid core reactor engines compare poorly with gas core reactor engines, even at the comparatively moderate specific impulse of 1500 sec, and even more so with the nuclear pulse engines, when applied to the very large payload Post-Saturn vehicles. Thus, all solid core reactor engines proposed so far are unsatisfactory for use as second stage propulsion systems for this size of launch vehicle.

4. In comparing nuclear pulse engines, one must remember that the mission spectrum of the Post-Saturn (Class IV) Earth launch vehicles combines orbital as well as lunar and planetary deliveries. This is a disadvantage to the nuclear pulse engine because its use for Earth orbit delivery may be unacceptable because of non-technical reasons (nuclear test limitations); particularly, since its superiority is not exercised at this limited energy level.

5. The principal disadvantage of the gas core reactor engine appears to be that its specific impulse growth potential appears to be limited to 2000 to 3000 sec, for thrust/weight ratio of one, required for sub-orbital start. Also, development problems appear to be considerably more severe than those associated with a similar increase in nuclear pulse specific impulse.

6. Slight preferences toward the gas core reactor or nuclear pulse may be expressed depending on the mission emphasis. Ratings of the two concepts for different mission emphasis are shown below:

a. Primary emphasis on Earth orbit delivery:

(1) First stage, chemical; second stage, gas core reactor (Saturn V Family or Helios).

(2) Main reasons - economy, reusability, mission versatility, highest launch facility rating.

b. Primary emphasis on lunar or planetary delivery:

(1) First stage, chemical; second stage, nuclear pulse (Saturn V Family or Post-Saturn Baseline).

(2) Main reasons - economy, reusability, mission versatility, performance growth potential.

7. Item 6 above is based on the assumption that either nuclear stage could and would eventually be used for direct delivery, i. e., sub-orbital start. If this should become unacceptable for the nuclear pulse, then the ideal chemo-nuclear Earth launch vehicle would have the following characteristics:

a. Stage 1 would be chemical, reusable.

b. Stage 2 would be gas core reactor powered into Earth orbit; have alternate mission capability involving direct delivery to lunar destinations; stage 2 would be reusable from orbit and surface.

c. Stage 3 would have nuclear pulse power for delivery via direct flight mode; low thrust/weight orbital start-up for very heavy payload mission to the Moon, round trip missions to the planets (reusable from orbit), and very high injection velocities for trans-Saturn and trans-Pluto probes (expendable).

### C. LAUNCH SYSTEMS

Based on the mobile assembly method, a launch support facility can be used to support either the off-shore launch system or the Saturn type support system. This area would house all of the support operations needed to prepare a vehicle for launch. Individual stage receiving and checkout buildings would permit minimum requirements for the vertical assembly building as major rework, maintenance, and checkout operations are performed at the stage buildings. Assembly of the stages and payload and final vehicle checkout is performed in the vertical assembly building. Installation of retro-rockets, touchdown engines, and ordnance items would be made after leaving the vertical assembly building. After recovery, the stage would be purged, placed aboard a transporter, and transferred to its refurbishing bay.

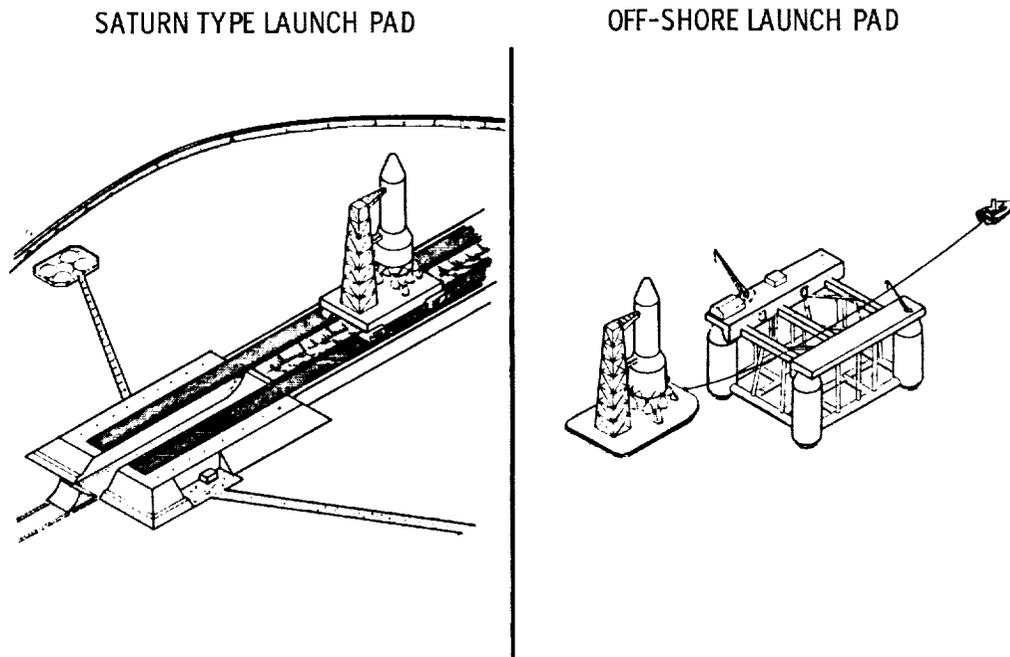


FIGURE 9. LAUNCH PAD COMPARISON

After completion of vehicle preparation operations at the assembly area, the vehicle would be transferred to the launch pad (Fig. 9). Depending on the system selected as most compatible with the vehicle, the pad could be either the off-shore floating launch pad or the land-based pad as used for Saturn V. The floating pad is similar to platforms now used in off-shore oil-drilling operations. The buoyant caissons at each corner of the platform and the low center of gravity provide stability. A desired elevation would be attained by means of ballast. Such a mobile launch pad could be used when the need for a remote site existed, as for single-stage nuclear, or when real estate needs became prohibitive as for the single stage chemical. For vehicles using gas core reactor engines with a launch weight of  $7 \times 10^6$  lb, the Saturn type launch system appears most attractive.

#### D. CONCLUSIONS: SUMMARY

1. By investigating Earth launch vehicles beyond the purely chemical systems (chemo-nuclear), a safeguard is provided against the danger of being caught short relative to future extraterrestrial operational requirements.
2. A 48 percent increase in gross payload for Earth orbit delivery can be obtained by replacing the second stage of a two-stage chemical ( $O_2/H_2$ ) vehicle of  $10^6$  lb payload to orbit with a nuclear stage using solid core reactor engines of 825 sec specific impulse. However, this does not look attractive from the overall operational and program viewpoint.
3. Replacement of the second stage with a gas core reactor engine powered stage with a specific impulse of 1500 sec, increases the gross payload into Earth orbit by 140 to 200 percent. The reason for this spread is that the design performance of this vehicle is a parabolic injection type mission. The lower percentage represents orbit delivery at partially de-tanked condition.
4. Two basic mission concepts can be defined for the Post-Saturn Class IV vehicle (chemo-nuclear) for escape type mission. These concepts are: (a) A two-step delivery with the subgroups to the departure Earth orbit involving orbital vehicle assembly; or (b) The entire interorbital (lunar or planetary) vehicle may be carried into orbit in operational condition, which would permit direct flight out of the parking orbit. Comparison of orbital vehicle assembly mode, direct flight mode, and direct ascent delivery shows that the direct flight mode is preferable, assuming everything else is equal. This approach, however, favors larger than  $10^6$  lb payloads into orbit, i. e., 1.5 to 2.0 times  $10^6$  lb.

5. Solid core reactor engines (graphite or metal base) compare poorly with gas core reactor and nuclear pulse on large vehicles considered in this study. They also do not show significant advantages with respect to cost effectiveness over advanced chemical ( $O_2/H_2$ ) vehicles. In addition to this, operational disadvantages penalize solid core propulsion systems.

6. Gas core reactor engines and nuclear pulse engines are both attractive propulsion systems for the Post-Saturn Class IV vehicle, and both should be considered in future studies. Mission emphasis may affect their selection, that is, if emphasis is on Earth orbit and lunar delivery, the gas core reactor shows a slight advantage. If primary emphasis is on lunar and planetary deliveries, the nuclear pulse concept is clearly preferable.

## SECTION V. RECOMMENDED FUTURE ACTIVITIES

Listed below are some key problem areas and recommended actions:

### A. PROPULSION

1. Accomplish test program for  $O_2/H_2$  high pressure engines and feed system.
2. Perform test program on transpiration cooling of high temperature nozzles.
3. Strengthen R&D program to establish a basis for choice between graphite and water-moderated engines, and perform studies and test program on:
  - a. Neutron interaction between clustered nuclear engines.
  - b. Reactor and engine control aspects in nuclear engine clusters.
  - c. Thrust structure and propellant tanks for nuclear engine clusters.
  - d. Perform studies and test program regarding cool-down of nuclear engine clusters of stages to be recovered from orbit.
4. Initiate R&D programs to provide basic design and performance data for gas core reactor engines and establish a basis for choice between various schemes.

# External Pulsed Plasma Propulsion And its Potential for the Near Future

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**Abstract.** This paper examines External Pulsed Plasma Propulsion (EPPP), a propulsion concept that derives its thrust from plasma waves generated from a series of small, supercritical fission/fusion pulses behind an object in space. For spacecraft applications, a momentum transfer mechanism translates the intense plasma wave energy into a vehicle acceleration that is tolerable to the rest of the spacecraft and its crew. This propulsion concept offers extremely high performance in terms of both specific impulse (Isp) and thrust-to-weight ratio, something that other concepts based on available technology cannot do. The political concerns that suspended work on this type of system (i.e. termination of Project ORION) may now not be as insurmountable as they were in 1965. The appeal of EPPP stems from its relatively low cost and reusability, fast interplanetary transit times, safety and reliability, and independence from major technological breakthroughs. In fact, a first generation EPPP system based on modern-day technology (i.e., GABRIEL - an evolutionary framework of EPPP concepts) may very well be the only form of propulsion that could realistically be developed to perform ambitious human exploration beyond Mars in the 21st century. It could also provide the most effective approach for deterrence against collision between earth and small planetary objects - a growing concern over recent years.

## INTRODUCTION

NASA is currently conducting research on advanced propulsion technologies capable of supporting ambitious human exploration of the solar system in the early part of the next century. Most research to date has been geared towards concepts that offer tremendous performance improvements over current systems. The only problem is that virtually all of these technologies, such as fusion, antimatter and beamed-energy sails, have fundamental scientific issues and practical weaknesses that must be resolved before they can be seriously considered for actual applications. For instance, fusion is limited by the fact that we are still far away from demonstrating a device having energy gains sufficient for commercial power, let alone space applications. Antimatter has much appeal because of its high energy density, but it is severely hampered by extremely low propulsion efficiencies and high costs of current production methods. Beamed energy offers great potential too, but requires materials far beyond current state-of-the-art and tremendous investment in ground/space-based power beaming infrastructure.

Although we are optimistic that some of these issues will eventually be overcome, there is no guarantee that any of these technologies will be available by the first half of the next century. This state-of-affairs points to the disappointing fact that none of the advanced, high-power density propulsion concepts being considered by NASA could, with any degree of certainty, meet the goals and timetables of NASA's own Strategic Plan. This is especially true in light of the conservative fiscal environment of the post-Apollo era, which could limit the sizable investment needed to resolve the fundamental issues associated with these concepts. Moreover, the cost for developing actual vehicles based on these technologies and their required infrastructure could realistically be on the order of hundreds of billions of dollars.

To obtain a quantum jump in propulsive capability by the early part of the next century, we must have safe, affordable systems with very high-power densities. Precedents suggest that any device engineered within the next 30 to 50 years should be based on the well-understood physics of today. The need for high power densities eliminates

all but nuclear energy sources. The emphasis on known physics and affordability limits the scope still further to fission processes. Of the fission-based concepts that have been considered in the past (e.g., solid-core nuclear thermal, gas-core, internal and external nuclear pulse), only external nuclear pulse circumvents the  $I_{sp}$  constraints imposed by containment of a heated gas, and provides the very high power densities needed for ambitious space transportation.

In the past, both internal and external pulse-engine concepts have been considered. Comparisons between these two approaches pointed to external pulse as the best candidate mainly because of its higher temperature limits and lower inert mass (Martin and Bond, 1979, Nance 1965). In addition, several researchers have investigated various forms of external momentum coupling. The most prominent examples are the standard pusher plate (Reynolds, 1972), the large lightweight sail/spinnaker (Solem, 1993), the rotating cable pusher (Cotter, 1971), and the combined pusher plate/magnetic field (Martin and Bond, 1979).

The most familiar effort in the area of external pulse-engines was Project ORION, which took place between 1958 and 1965. The Air Force spent approximately 8 million dollars on the program over its first 6 years (Prater, 1963). ORION, which was classified throughout most of its brief lifetime, engaged an impressive group of physicists and engineers who carried out numerous studies and tests on most aspects of the vehicle. The basic ORION design is shown in Figure 1. The proposed ships were large (from 10 to 30 meters in diameter) since performance tended to increase with diameter of the ship's pusher plate. This was due to the higher specific yields (i.e., burn up fractions) of larger pulse units, and the wider propellant interception angles at the minimum standoff distances allowed by material strength considerations. NASA funded several additional studies until 1965 when the entire effort was terminated - primarily for political reasons. The extensive analyses and experiments performed for ORION and subsequent studies indicate that spacecraft with high thrusts (~1 to 10 g accelerations) and high  $I_{sp}$ 's (~10,000 sec) could be built, even with 1960's materials technology.

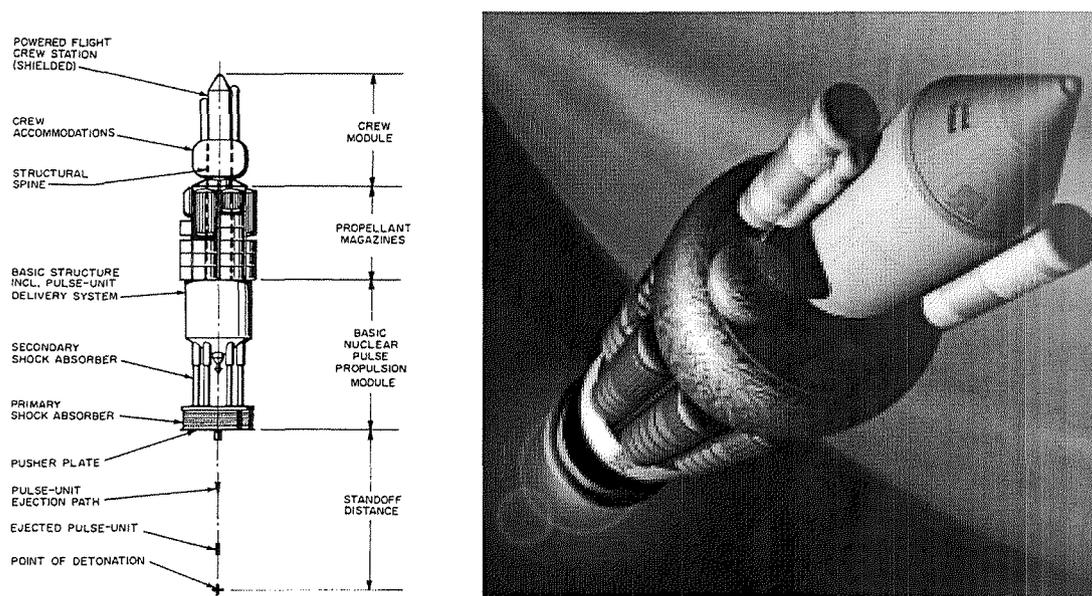


FIGURE 1. 1960 ORION concept.

## CONCEPT OVERVIEW

At first glance, a nuclear pulse rocket appears to be quite radical, although it is conceptually very simple. Thrust is produced by ejecting and detonating small, fission-driven, pulse units at the aft end of the vehicle. This "external" engine operation, where the fission process is unconfined by material walls, is relatively independent of the reaction rate, temperature, pressure and other characteristics of the fuel. In practice, the system must be operated in a pulsed mode to allow the transfer of energy into a practical acceleration of the ship, which is limited by human and

equipment tolerances. The physics behind creating a highly efficient fission burst is well understood, and in a vacuum, it produces a shell of ionized particles with an extremely high radial velocity. Thus, this concept of “riding on a plasma wave” is appropriately termed External Pulsed Plasma Propulsion or EPPP.

Key to EPPP’s extraordinary performance are the facts that: (1) common materials can withstand an intense nuclear environment for very brief periods of time (i.e., nanoseconds), and (2) nuclear detonations are not only well understood, but also come much closer to achieving the maximum power density available from the fission process. Also, high thrust over a relatively short time imparts nearly optimum impulse to the vehicle for fast, efficient trajectories. In sharp contrast to the original ORION approach, recent analyses based on present-day considerations and technologies (e.g., dedicated in-space operation, low-energy pulse unit yields, low-ablation pusher plate materials) indicate that the performance advantages of EPPP could be applied to relatively small vehicles. If this is the case, then it is possible to develop small spacecraft that could carry human crews between Earth and Mars in just 1 to 3 months, as opposed to 6 to 12 months with chemical or nuclear thermal propulsion technology. In addition, EPPP would permit much more flexible return windows and eliminate the need for long stay times in the vicinity of Mars. Most importantly, EPPP provides a technology path leading to much higher Isp’s (~100,000 sec) using larger vehicles and more energetic detonations (e.g., fission/fusion and fusion) which could ultimately be used to open up the entire solar system to human exploration.

The main objection to EPPP has been the concern over nuclear contamination. Since modern-day practices would assuredly limit this concept strictly to space, radioactive contamination may not be as serious of issue as with ORION. Furthermore, the harsh environment of space has far more background radiation (particularly in the form of harmful gamma rays) than that produced by very small pulse units. Within 24 hours, the pulse unit’s ionized mass dissipates completely into the background of the nominal space plasma density. Depending on the pulse unit efficiency, the exhaust velocities of the radioactive particles could exceed solar escape velocity (certainly beyond that of earth escape). Thus, there is no residue or permanent contamination to the environment beyond the natural sun’s radiation.

### **Application #1: Human Interplanetary Exploration**

There are two reasons for seriously considering EPPP as an option for future development. The first is its potential for human exploration. Since the early years of the space program, most human exploration studies have concentrated on either the Moon or Mars. Although it is recognized in NASA’s Strategic Vision that the ultimate goal is to extend human presence throughout the solar system and eventually the stars, only a negligible amount of effort has been devoted to these type of missions. EPPP provides a technology that would allow us to seriously consider missions to the outer planets. It would also enable dramatically shorter trip times to Mars and other nearer-term destinations.

The propulsion concepts that have been traditionally considered for Mars missions are chemical propulsion based on O<sub>2</sub>/H<sub>2</sub> combustion and solid-core nuclear thermal propulsion. Although the Isp of nuclear thermal (~900 sec) is approximately twice that of chemical (~450 sec), both systems suffer from the same limitations with regards to trip time and mission planning. The main advantage of nuclear thermal is its potential to reduce vehicle mass in low-earth orbit, thus reducing the number of heavy-lift vehicle launches.

The performance that characterizes these two concepts favors Hohmann-type transfers into very slow heliocentric orbital trajectories. This narrows the available trajectories for return and necessitates long stays on the Mars surface while awaiting favorable return windows. This leaves the crew and equipment exposed to an extremely hostile environment for long periods of time - nominally 560 days surface stays with 170 to 200 day transit times (Kos, 1998). Cost is also significant, since earth launches are about half the mission budget in most conventional scenarios. Longer missions translate to larger payloads and more expendables, both of which increase launch requirements.

EPPP can solve this problem with its much higher Isp (5,000 to 10,000 seconds), while still providing the high-thrust needed for fast orbit transfers. The result is higher energy transfer orbits, which could greatly reduce not only transit time, but permits broader return windows. This provides much more flexibility in mission planning and would not constrain the crew to long stay times on the Martian surface. It would also reduce the crew’s exposure to the highly radioactive space environment and long periods of weightlessness.

## Application #2: Comet/Asteroid Deflection

The other and perhaps most compelling application for EPPP is its use in asteroid or comet defense. Collisions between the Earth and small planetary objects occur frequently, with the typical result being that the objects burn up in the atmosphere. However, there is a low, but not negligible, probability of a collision with objects of sufficient size to cause catastrophic damage or an extinction-scale event. Good risk management would dictate that some effort be placed on devising countermeasures, if possible. Past studies identified a number of possibilities, almost all of which entailed ground and space-based infrastructure more extensive than that envisioned for ballistic missile defense. Because of the limitations of current propulsion technology, these systems would require permanent deployment of interceptors in deep space in order to allow engagement at a sufficient distance from Earth. In addition, the low-impulse methods of altering the object's trajectory, such as sails or electric thrusters, would probably not provide enough time for adequate trajectory alteration between detection and impact - especially in the case of a comet.

EPPP could be applied to the development of a much less expensive, purely ground-based deterrence system. If a likely catastrophic collision were identified, an EPPP-propelled interceptor could be launched into space using a conventional chemical launcher. It would have the power density necessary to rapidly travel to the target in time to force the threatening object from its collision course. The object's course change might be performed using sails or electric thrusters. However, these schemes are very risky since their effectiveness depends on the body's size, shape, speed, trajectory and many other properties. There is little room for error once the target is engaged, and the propulsion systems must operate reliably for very long durations to effect the change.

Alternatively, the same EPPP system that propelled the interceptor could be used to move the target. Single or successive pulse detonations at a predetermined distance from the asteroid's surface could be used to easily "nudge" the planetesimal and alter its course. The first wave of X-rays from the pulse would illuminate the planetesimal's surface causing ablation and thrust parallel to the object's projected area. The second wave of pulse fission products would produce another impulse in the same direction.

This approach has important advantages. It does not require asteroid capture or attachment of a propulsion unit to a highly variable surface. Since the "thrust" is parallel to the object's projected area, this approach is independent of the object's relatively indeterminate mass distribution and angular momentum. Also, the amount of impulse delivered can be easily tailored to any asteroid by the number of pulses, detonation standoff distance, and type of pulse unit.

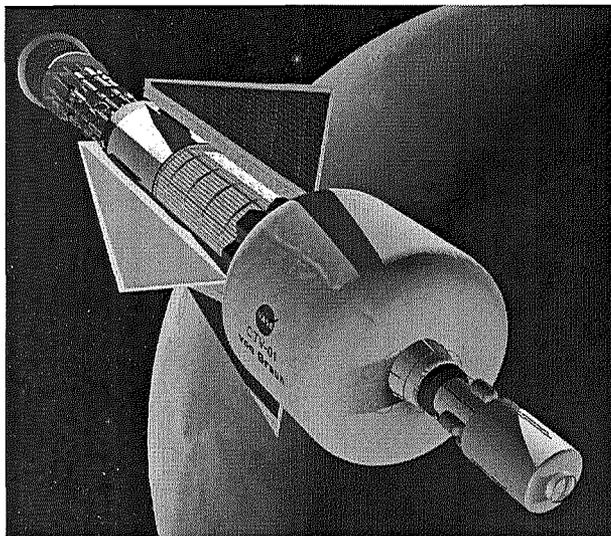
## DESIGN CONCEPTS UNDER STUDY

The realistic maximum Isp obtainable with fission-based EPPP is ~100,000 seconds. However, this type of performance would only be possible with very large spacecraft. Such vehicles would be impractical until the cost of access to space dropped substantially or in-space manufacturing became available. Therefore, a more conservative approach has been taken by considering smaller vehicles with lower performance ( $I_{sp} \leq 10,000$  seconds) using technology available in the near-term. This concept has been informally termed "GABRIEL." The GABRIEL series includes an evolutionary progression of vehicle concepts that build upon the nearest-term implementation of EPPP. This concept roadmap eventually culminates in larger systems that employ more sophisticated methods for pulse initiation and momentum transfer. GABRIEL is characterized by the following four levels:

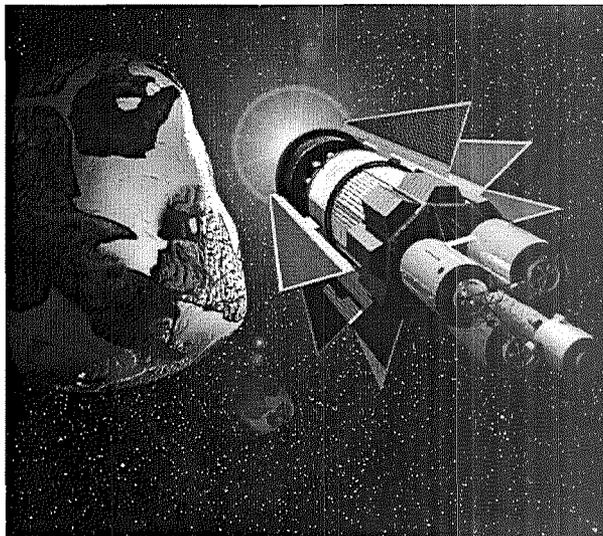
1. Mark I: Solid pusher plate and conventional shock absorbers (small size)
2. Mark II: Electromagnetic coupling incorporated into the plate and shocks (medium size)
3. Mark III: Pusher plate extensions such as canopy, segments, cables (large size)
4. Mark IV: External pulse unit driver such as laser, antimatter, etc. (large size)

All of these levels, besides the GABRIEL Mark I, require technology that is not currently available, but may be attainable for a second-generation vehicle. The Mark I (Fig. 2) is also the smallest and least expensive version, but suffers from the poorest performance (nominally 5,000 seconds and 4 million newtons of thrust). Nonetheless, the Mark I has better Isp and thrust than any other known rocket system that could be reasonably developed within the next 20 years. Its heavy payload capacity and short trip times would significantly reduce the development challenges associated with manned spacecraft, as well as add extra safety margins through redundant systems, large reserve supplies and increased robustness. Interestingly, the same shielding used to protect the astronauts from solar

flares could be used during engine operation (usually only a few hours at most), and the resulting radiation dose received would be much less than conventional multi-year missions. It is even conceivable that a vehicle with a performance as high as 4,000 seconds and 2 million newtons of thrust could be deployed and assembled in orbit using several Titan IV launch vehicles.



**FIGURE 2.** GABRIEL Mark I vehicle.



**FIGURE 3.** Asteroid deflection maneuver.

Several technical issues and trades must be addressed in order to define even a Mark I vehicle. These are the type of pulse unit, its degree of collimation, detonation position and fissile burn-up fraction. These issues dictate propulsion efficiency and drive design of the vehicle's mechanical elements. Another issue is the pusher plate-plasma interaction. The amount of ablation experienced during each pulse could be significant and would dramatically affect Isp and thrust levels. Other issues include shock absorber efficiency, timing and dynamic response. Reusability will be important, so component wear must be kept to a minimum. In-space assembly, earth-to-orbit launch packaging and pulse unit safety and loading also must be addressed. Most of these issues have been investigated in the past and, although engineering challenges still remain, there are no formidable technical problems to overcome.

The ultimate hurdle in developing EPPP would be political in nature. Although GABRIEL does not face any insurmountable technical or financial obstacles, it does face one of perception. Use of nuclear material is almost always met by vehement opposition. However, there have been some important changes in the political landscape that may afford EPPP a chance where ORION failed. The Cold War is over and the fears of a large-scale nuclear conflict have abated somewhat. The existing ban on nuclear weapons in space actually has provisions that may allow peaceful uses of EPPP-type techniques below certain energies.

Even if EPPP is still viewed as too controversial for development in the near future, it would be worthwhile to begin reexamining it within the context of modern technologies and capabilities. Unlike physics, the sociopolitical environment does change, and a propulsion system with this tremendous capability may be needed - possibly on rather short notice (Fig. 3). The fact that many of the advanced propulsion concepts being researched now may never move beyond the "proof-of-principle" phase suggests that EPPP may be the only option we have for very ambitious human exploration of space in the foreseeable future.

## SUMMARY

The case for reexamining nuclear pulsed propulsion and more modern embodiments of the EPPP concept has been made. The modern version of this propulsion concept, GABRIEL, is distinguished by its superior performance (i.e., both high Isp and high thrust-to-weight), its practicality (borrowing from only existing technologies), benign

environmental impact (i.e., dedicated in space operation and reduced crew radiation exposure) and its economics (i.e., small size and reusability). More advanced systems with much better performance could be developed as technology in key areas mature. Improved performance can be achieved through advanced materials, magnetic fields (both on the pusher plate and along the shocks), novel momentum transfer schemes, and pulse unit drivers.

However, it is the rationale for considering EPPP that is most important. EPPP offers a highly effective method for deflecting comets or asteroids. Trips to and from Mars may be significantly shorter and safer than with conventional propulsion concepts. The flexibility of missions employing EPPP is enormous, allowing massive payloads, emergency return capability and routine transit from a reusable vehicle. Beyond Mars, missions to the asteroid belt, Jupiter and other planets are possible with the same basic system.

Timing for development of EPPP may also be better than during the days of ORION. In many ways, international cooperation is more prevalent, and could conceivably be extended to the peaceful application of unused nuclear material. Stockpiles of fissionable material can be permanently disposed of and environmental contamination is negligible if used outside the earth's magnetosphere. Finally, the human race is at the threshold of truly exploring, developing resources and permanently inhabiting space. GABRIEL may provide the best means of accomplishing this in the near future.

## ACKNOWLEDGMENTS

The authors would like to thank George Dyson for providing much of the background documents on the ORION program, and Peggy for providing the many final proof readings and grammatical corrections.

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## 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

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.

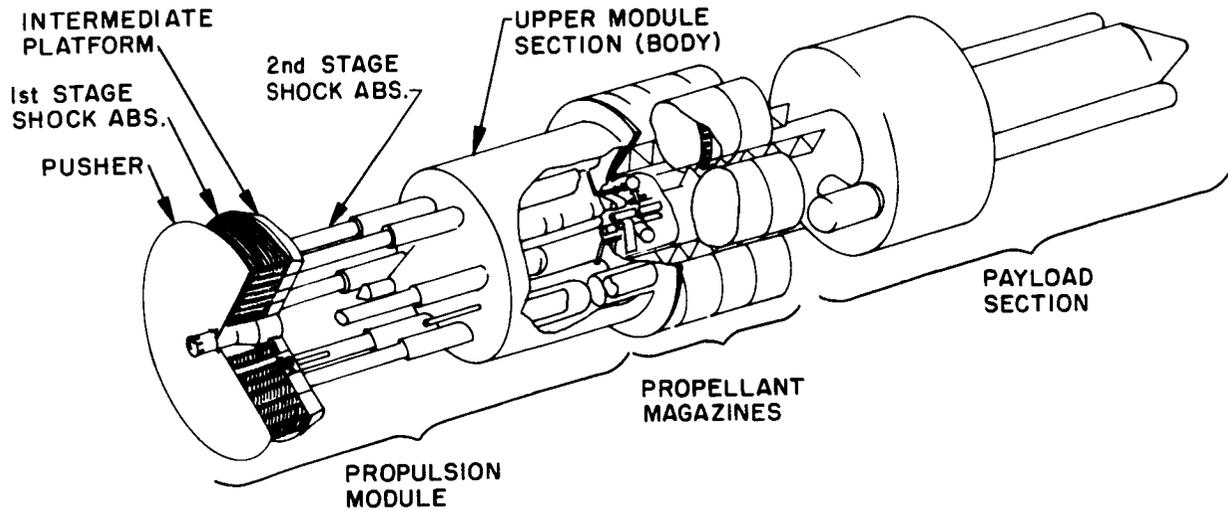


Fig. 2.1--Typical nuclear-pulse vehicle

### 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

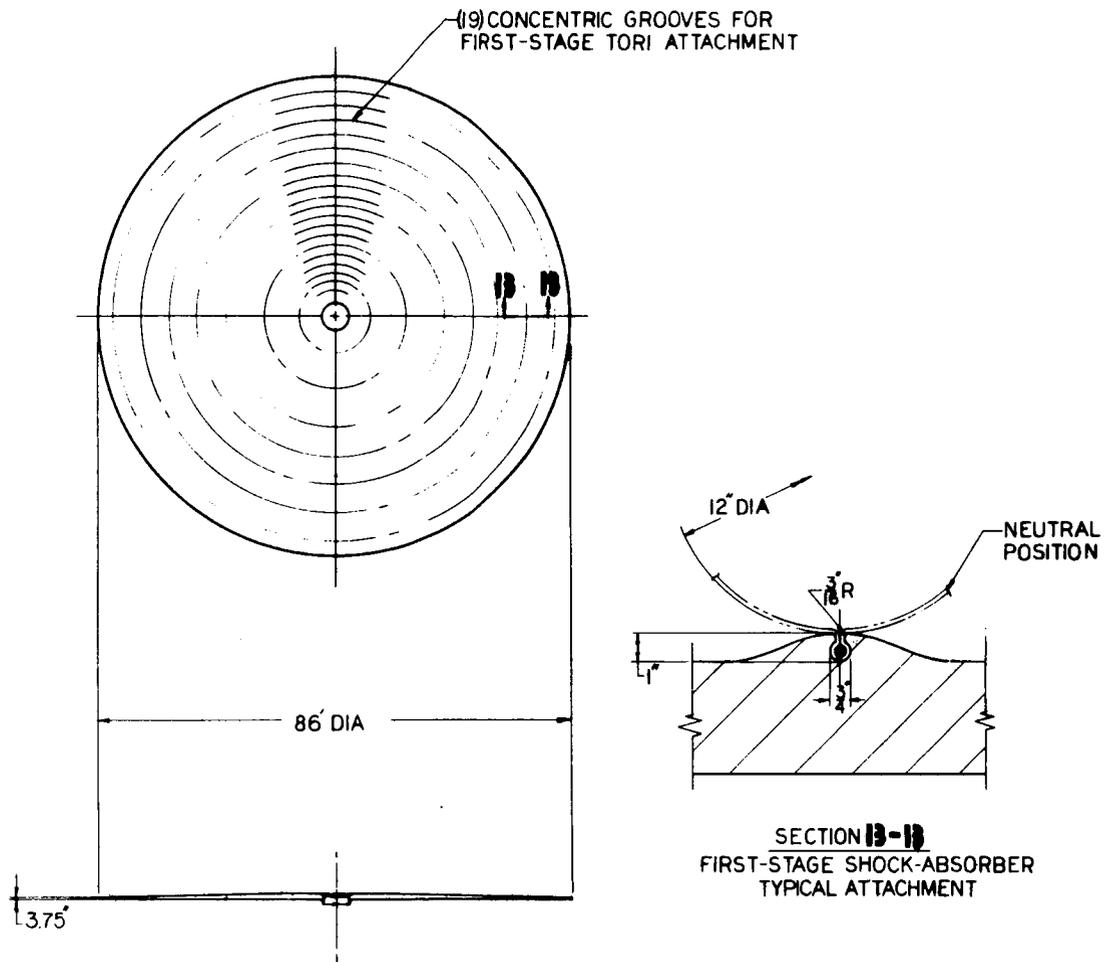


Fig. 2.2--Design of a large pusher plate

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

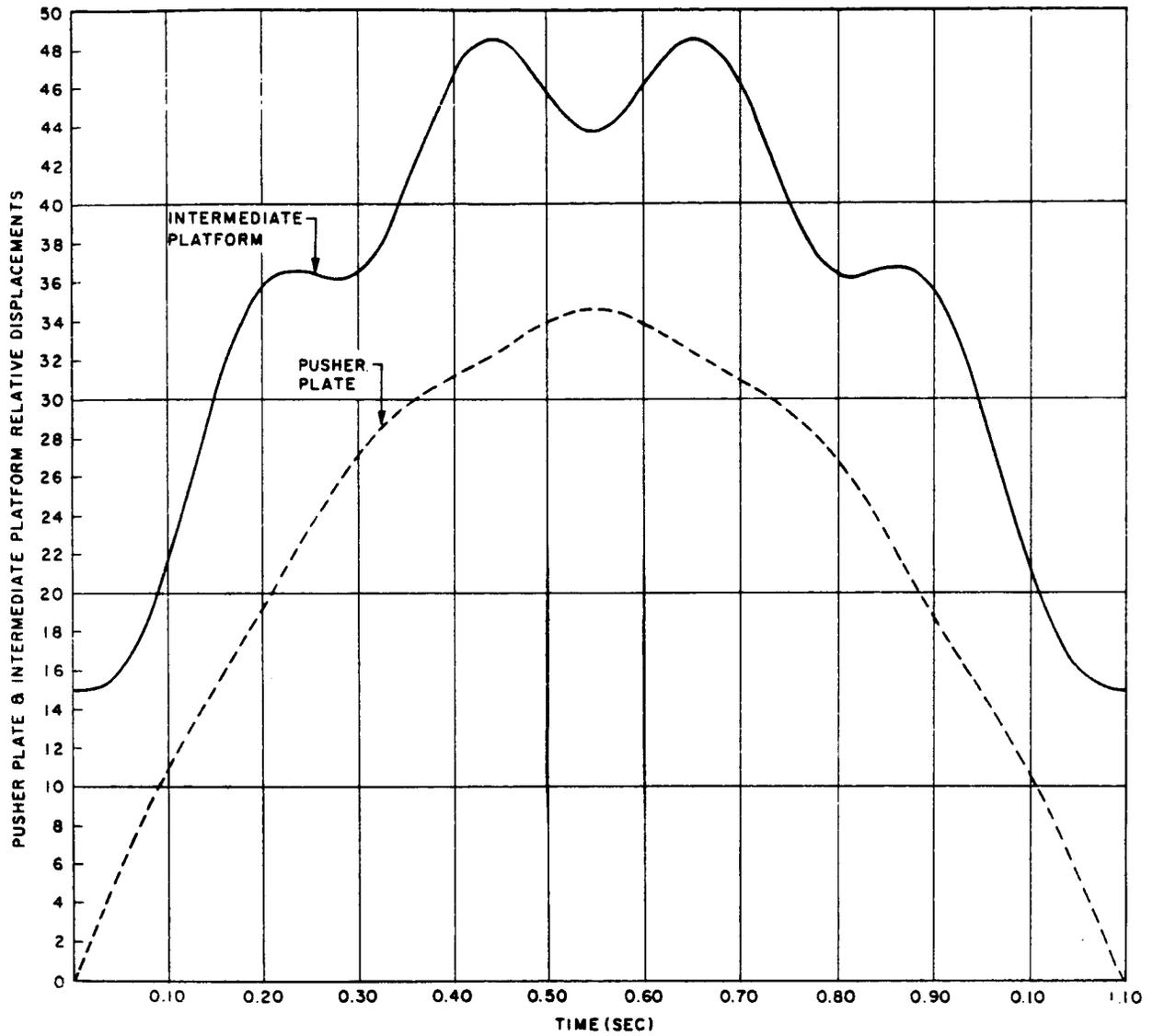


Fig. 2.3--Pusher and intermediate platform displacement with respect to the vehicle

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.

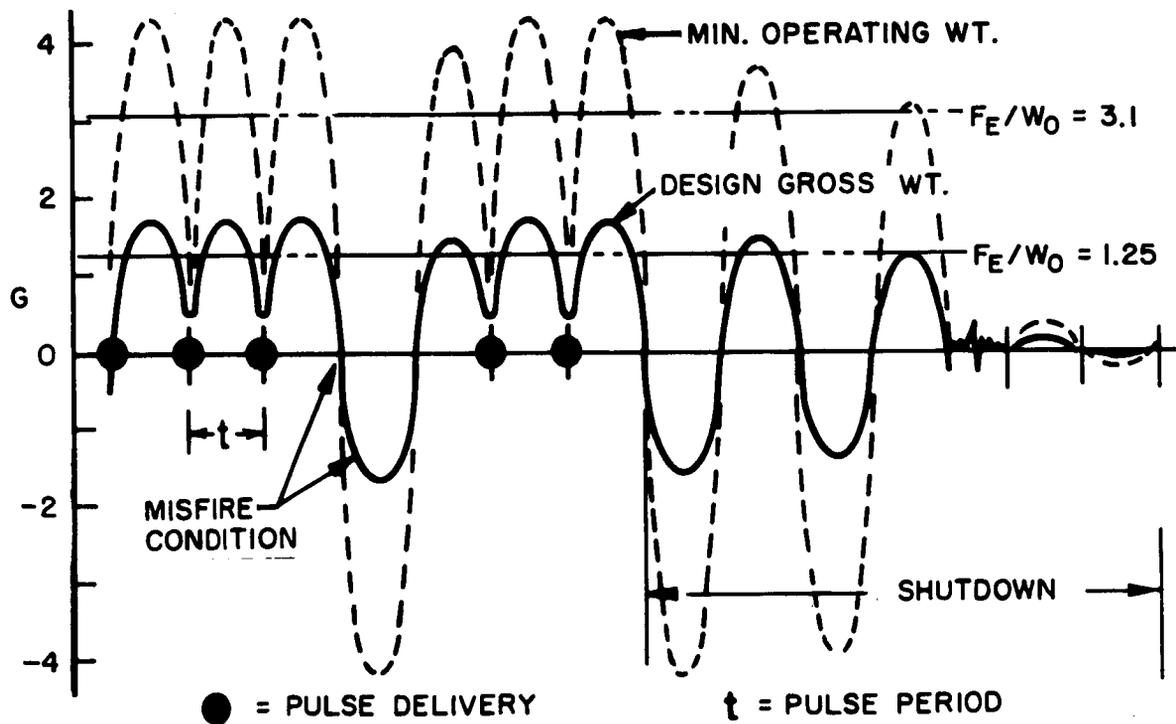


Fig. 2.4-- Typical nuclear-pulse-vehicle acceleration profile

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

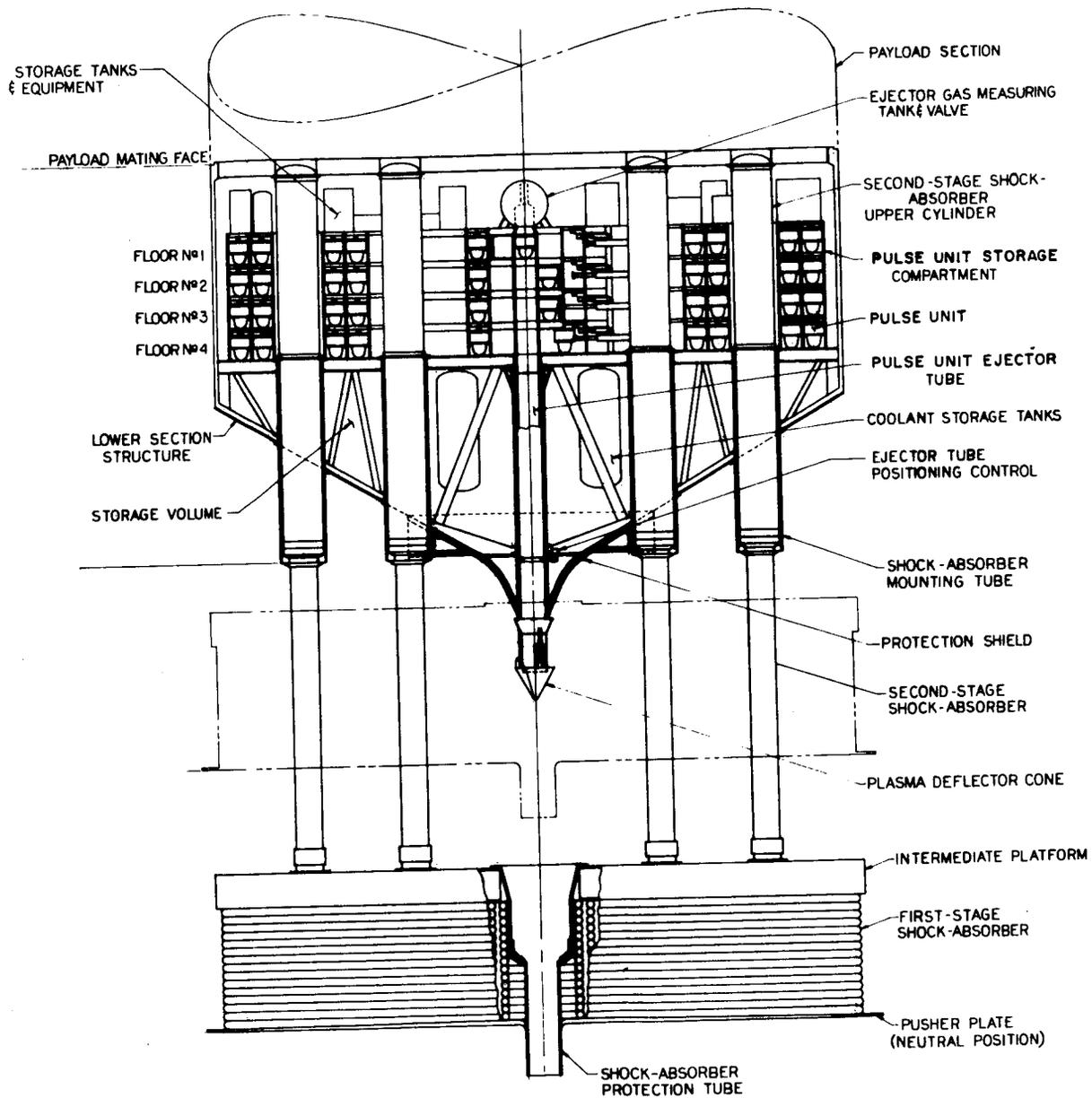


Fig. 2.5--Propulsion-module section layout

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 propellant 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.

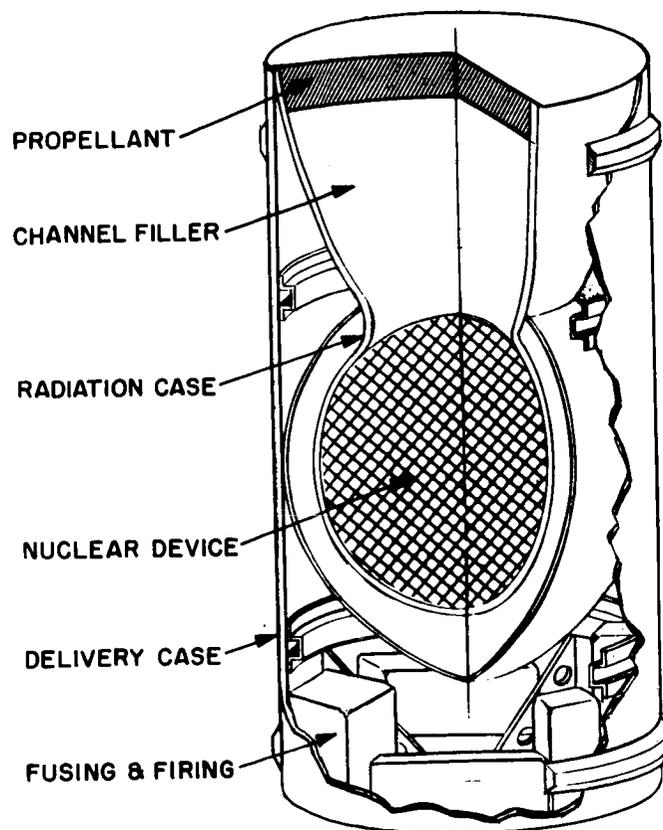


Fig. 2.6--Pulse-unit design for a 10-m propulsion module

The propellant and channel filler also serve as a shield to protect 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 breach 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.

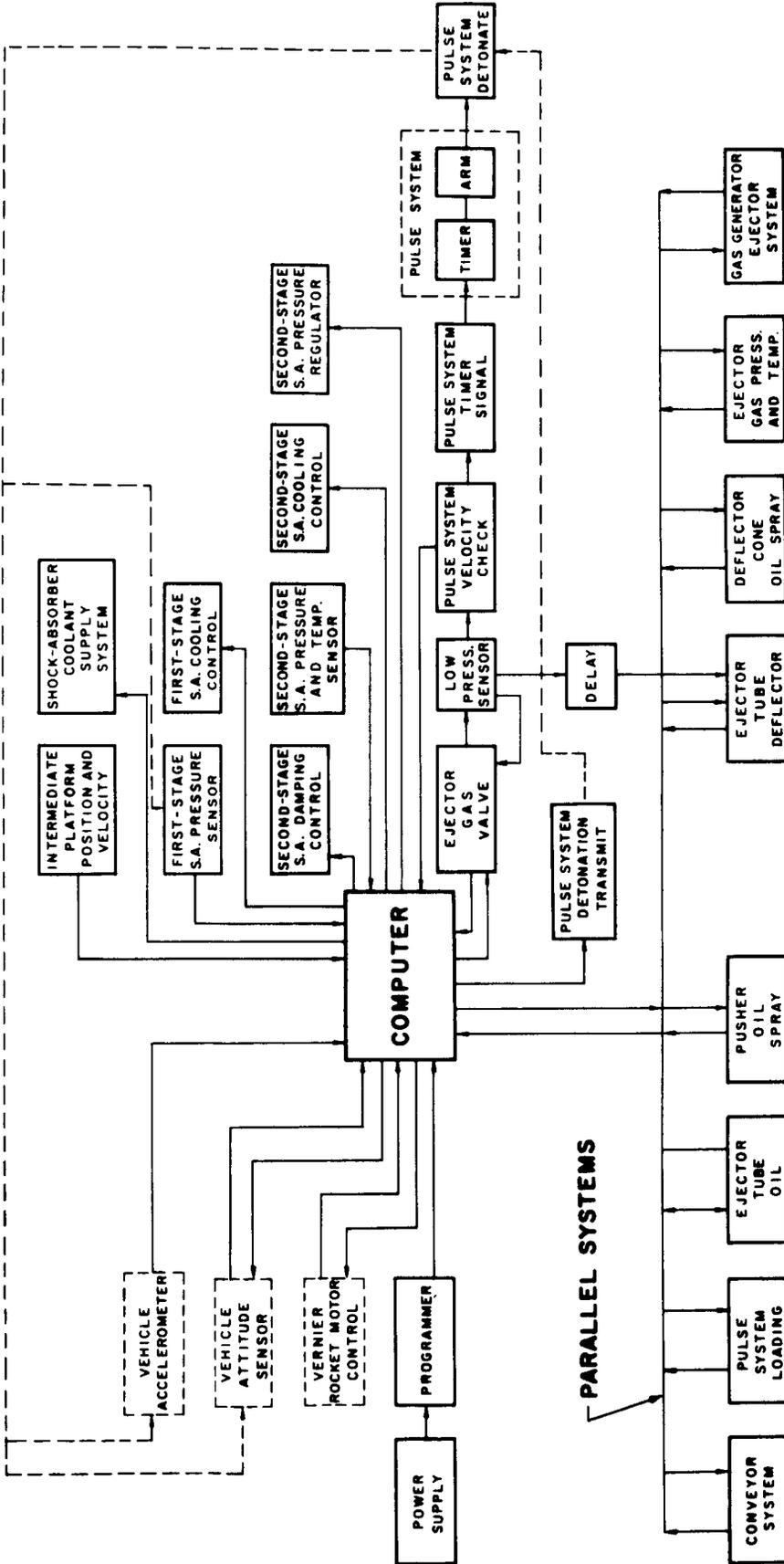


Fig. 2.7--Propulsion-system-control block diagram

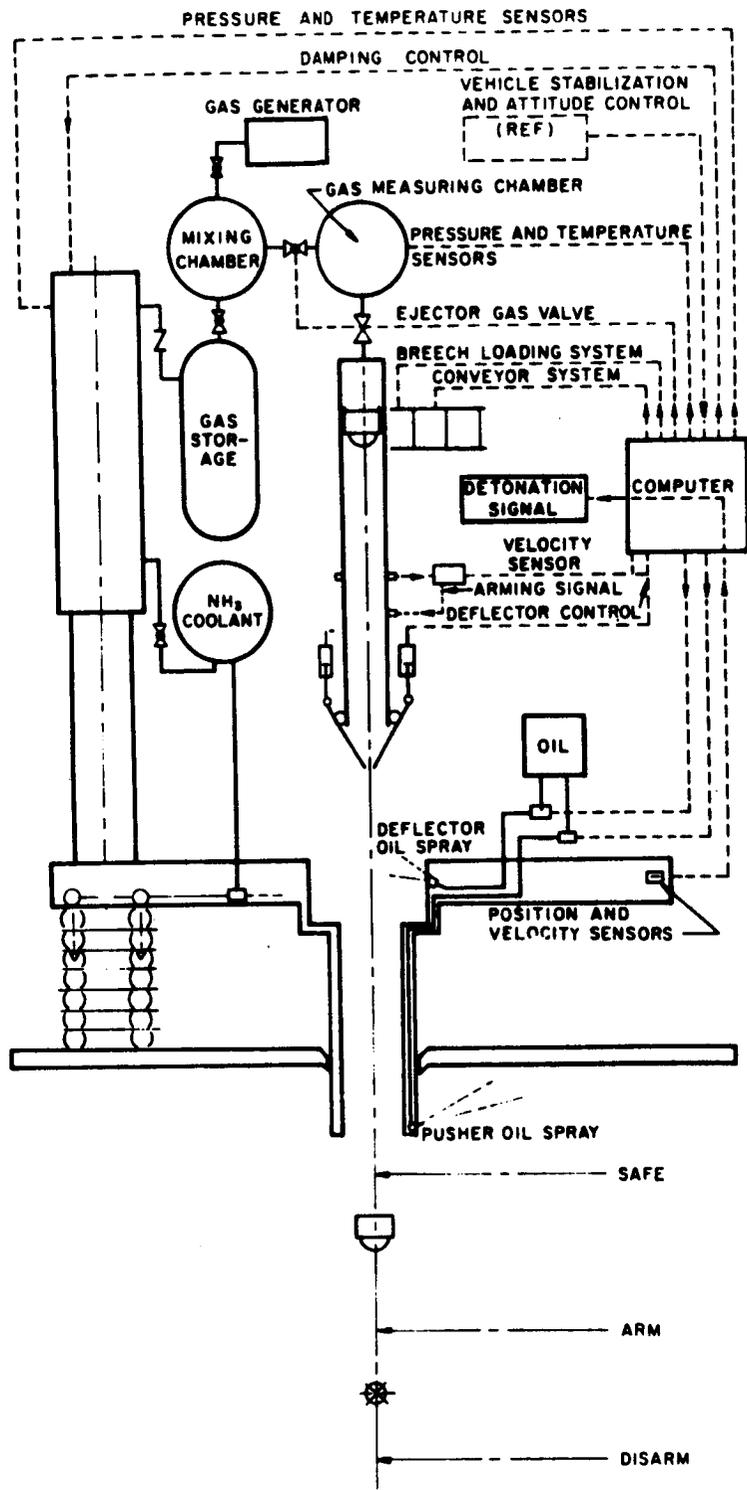


Fig. 2.8--Principal elements of propulsion system control

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  $44 \times 10^6$  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  $44 \times 10^6$  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.

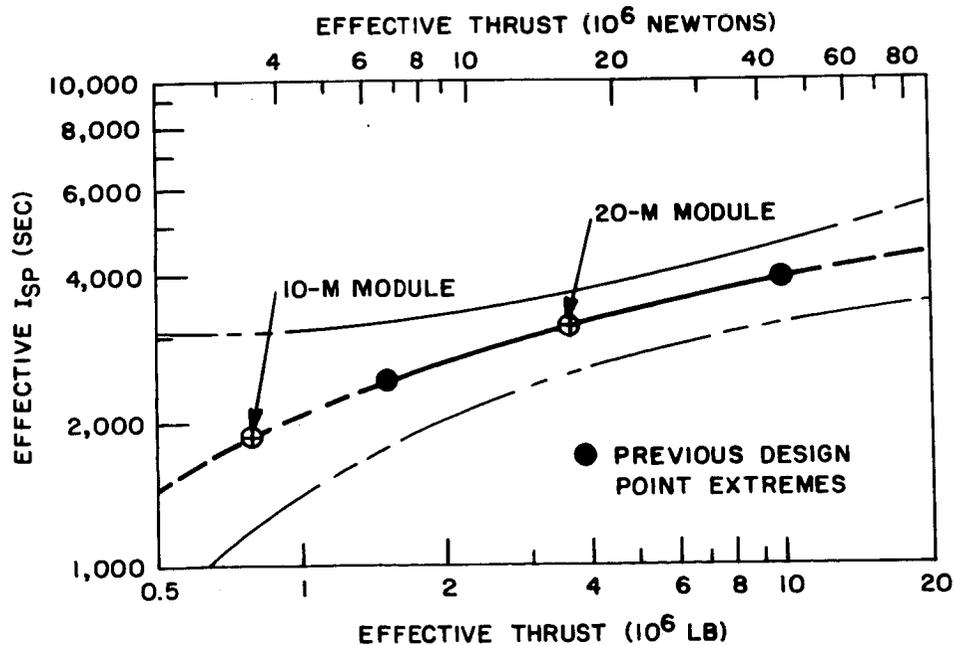


Fig. 2.9--Propulsion-module specific impulse

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:

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

where  $M(\alpha_e)$  = mass contained within the angle defined by  $\alpha_e$ ,  
 $V(\alpha_e)$  = average velocity directed perpendicular to the pusher within the angle defined by  $\alpha_e$ ,  
 $M_{ps}$  = 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  $\epsilon$ , where

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

Thus, the effective specific impulse for the propulsion system is

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

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  $\epsilon$  is approximately 0.2 for most systems which have been studied.

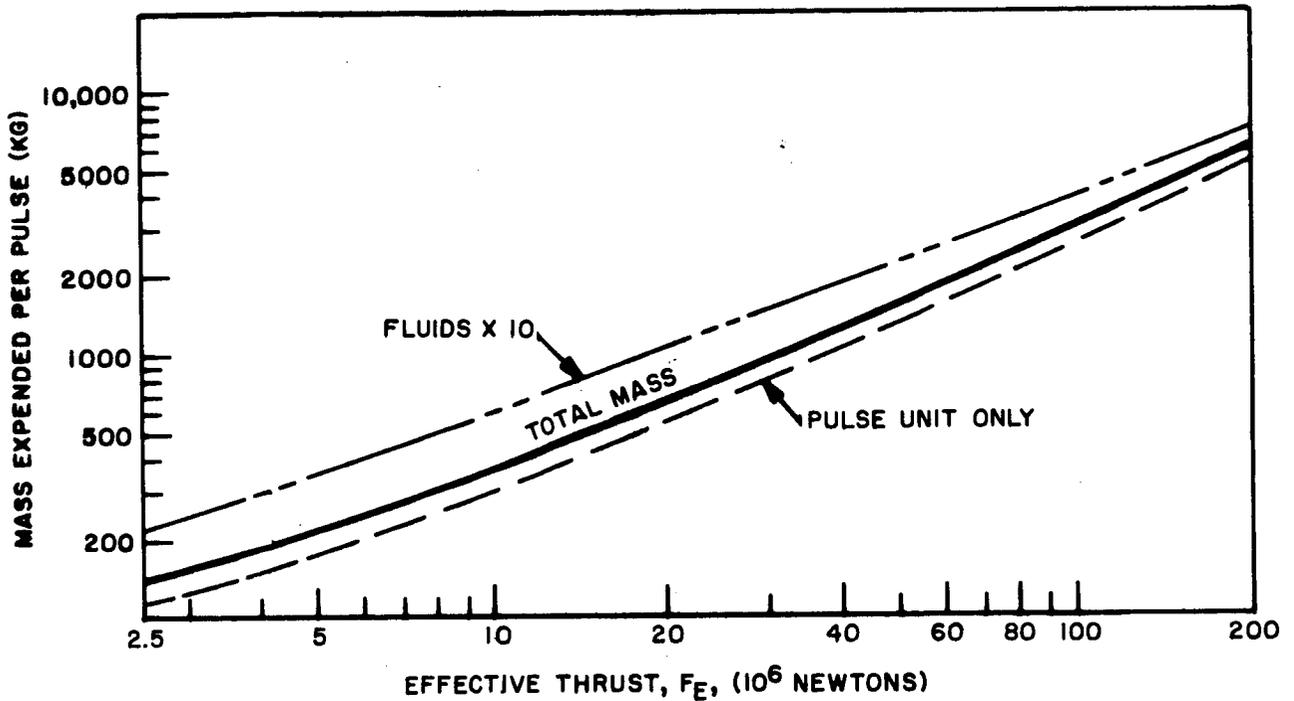


Fig. 2.10--Over-all propellant composition

### 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.

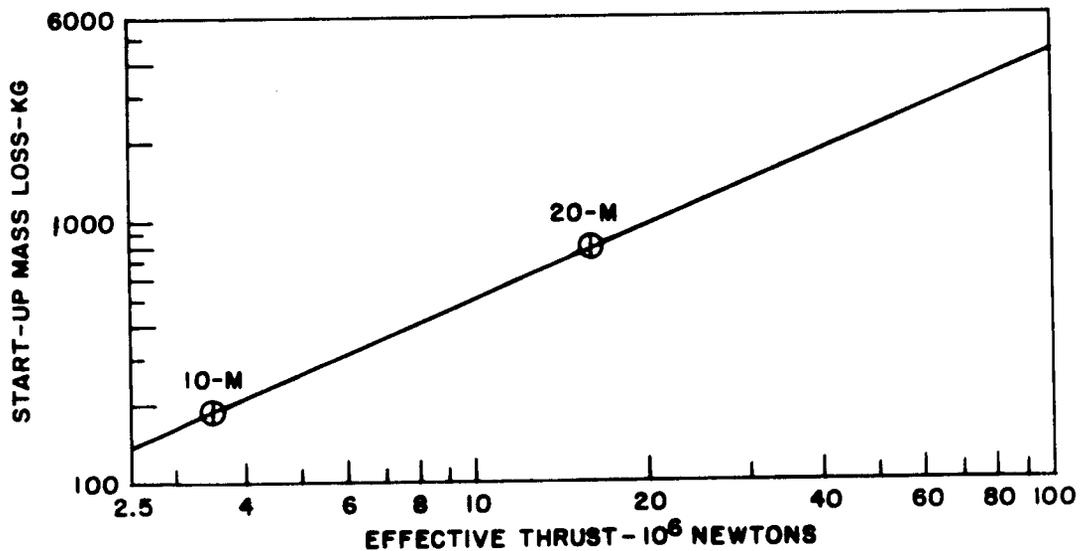


Fig. 2.11--Estimated mass loss per propulsion start-up cycle

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  $6 \times 10^6$  to  $50 \times 10^6$  newtons. At  $44 \times 10^6$  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.

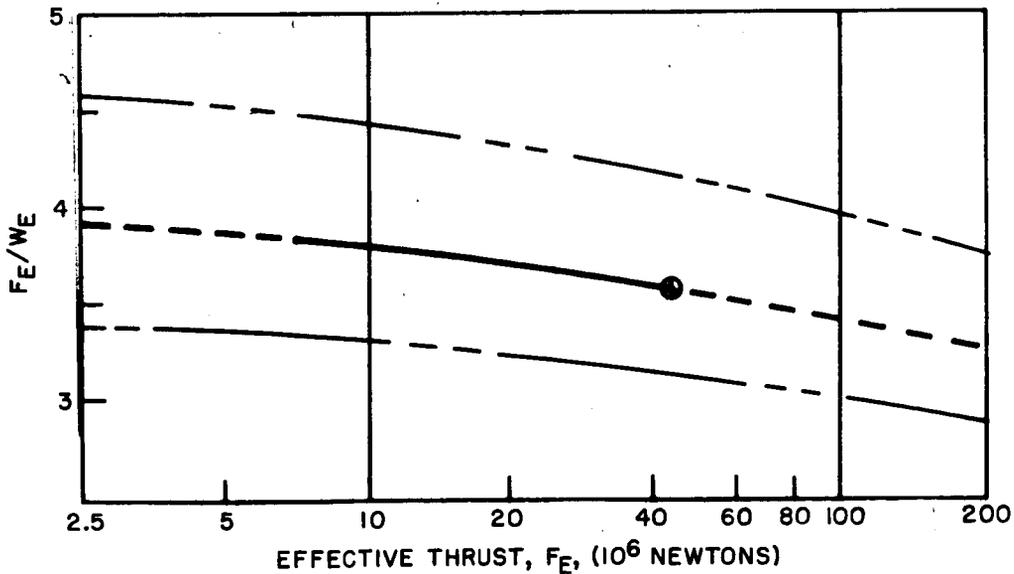


Fig. 2.12--Propulsion-module effective thrust-to-weight ratio ( $F_E/W_E$ )

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

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  $\Delta L$  of  $0.03 L_E$  is added to the basic length for each additional 500 pulse units desired.

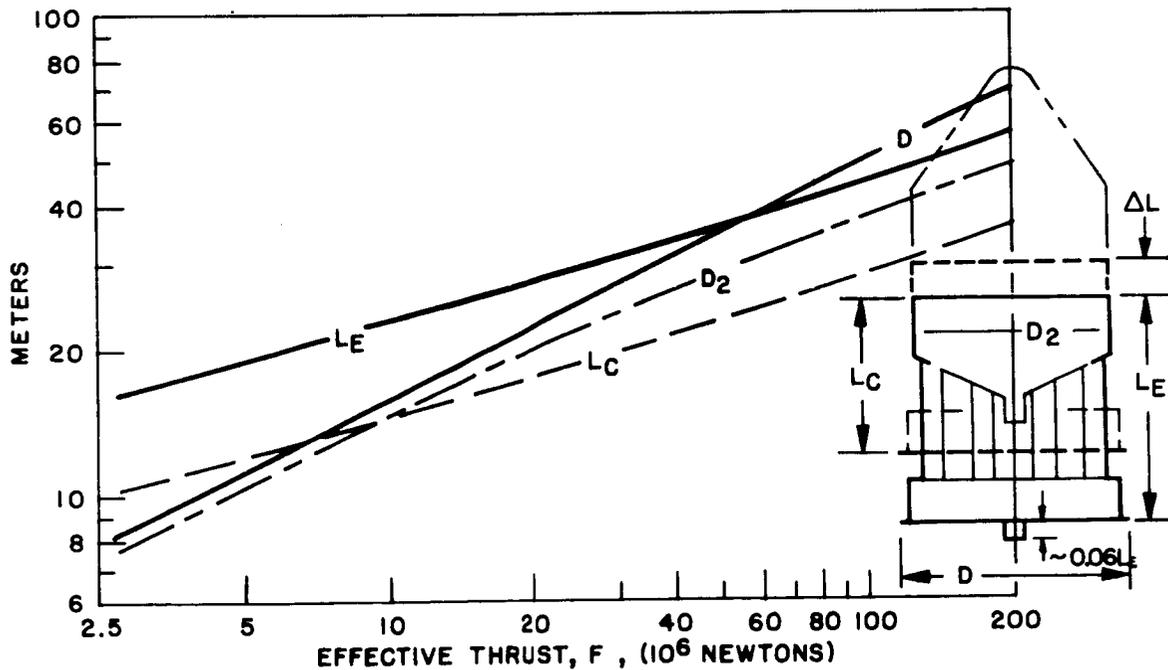


Fig. 2.13--Propulsion-module over-all dimensions

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

parameters in touch with reality, can be used to relate the pulse-unit design to propulsion-module design.

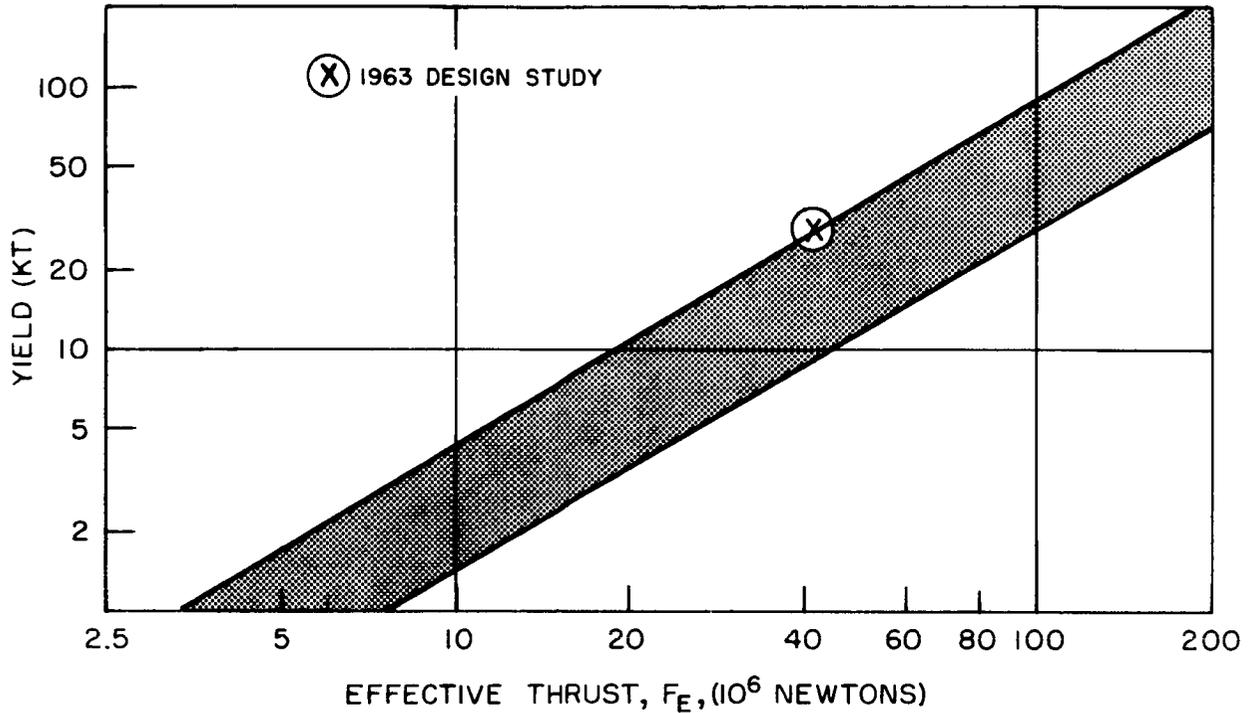


Fig. 2.14--Nuclear pulse-unit yield range

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,  $F_E$ :

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

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

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

where  $I_{sp}$  = effective specific impulse, in seconds,

$F_E$  = effective thrust, in newtons,

$W_E$  = 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.:

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

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

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

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.

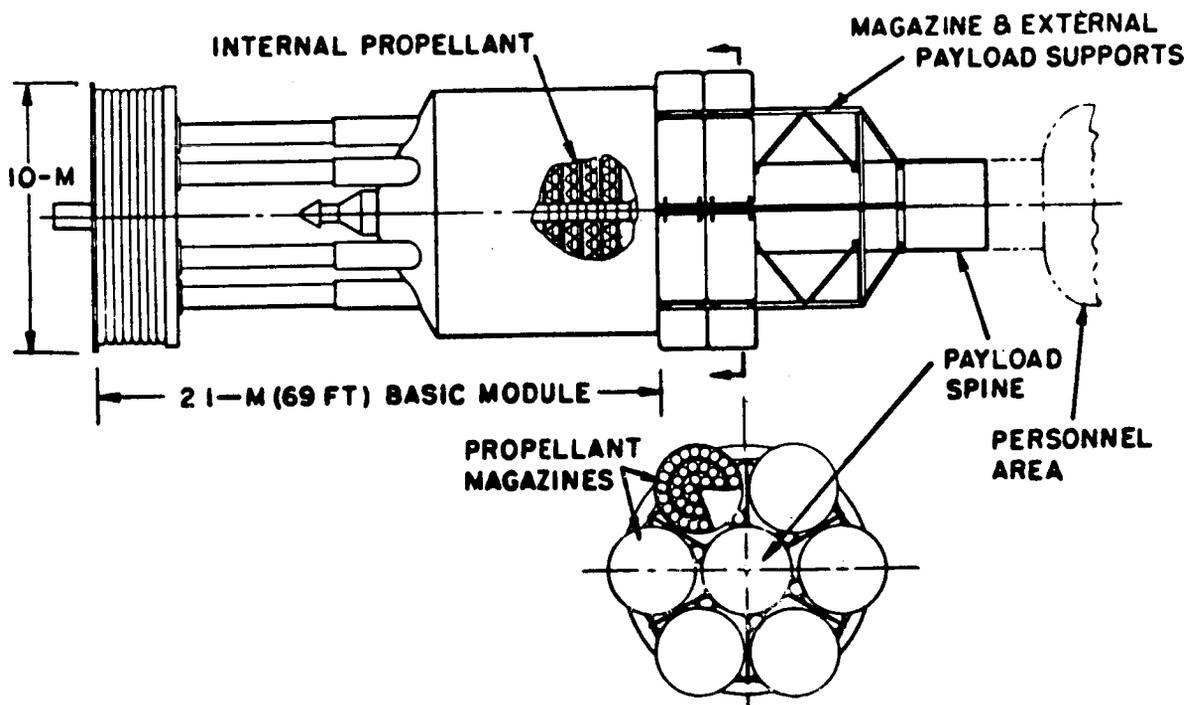


Fig. 2.15--Study configuration of the 10-m nuclear-pulse-propulsion module

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  $3.5 \times 10^6$  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:

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

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  $2 \times 10^7$  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  $4.5 \times 10^7$  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  $4 \times 10^7$  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.

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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.

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### 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 must 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<sup>2</sup> 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<sup>2</sup> 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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\*A Study of Manned Interplanetary Missions. Vol. II-Weight and Payload Report, General Dynamics/Astronautics, Report AOK 64-006-5, 1964.

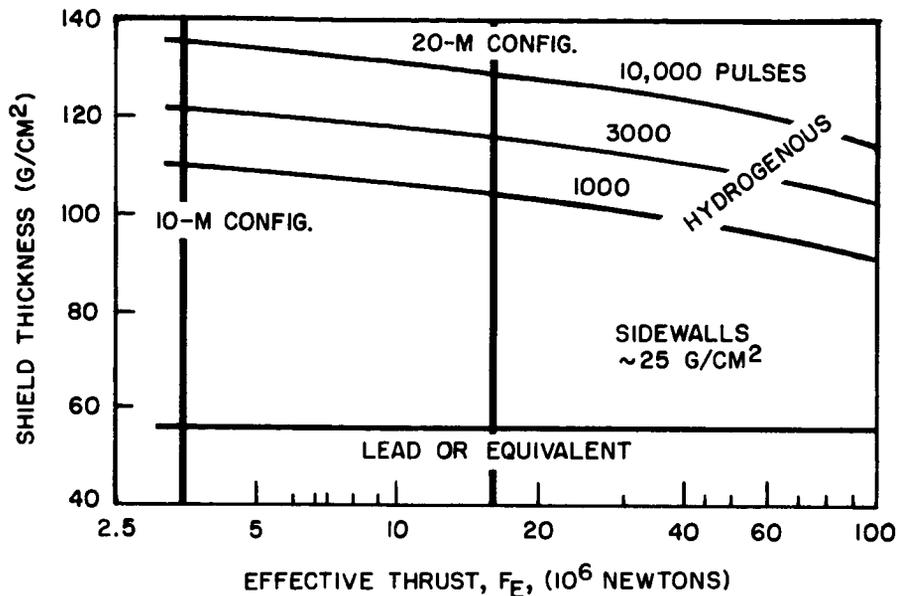


Fig. 3.1--Shielding requirements for bottom of powered flight station

Most missions for the 10-m configuration require less than 3,000 pulses, so a bottom shielding requirement of 120 g/cm<sup>2</sup> of hydrogenous material was used for most conceptual designs.

Shielding of 25 g/cm<sup>2</sup> 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

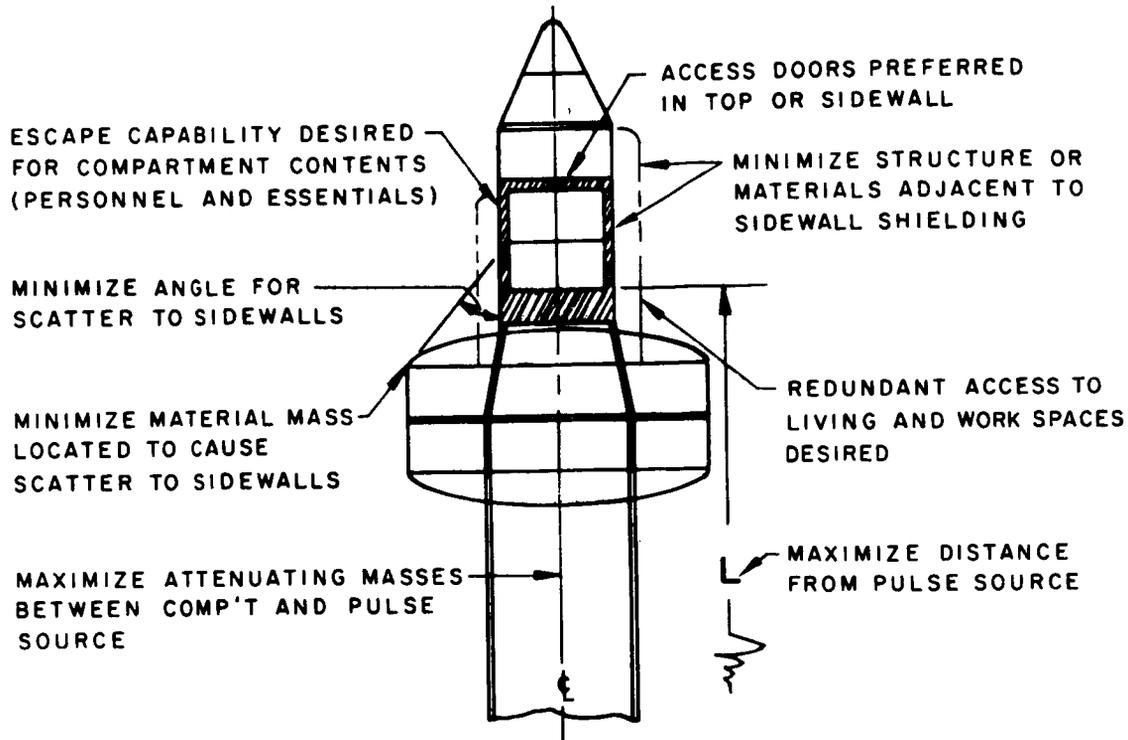
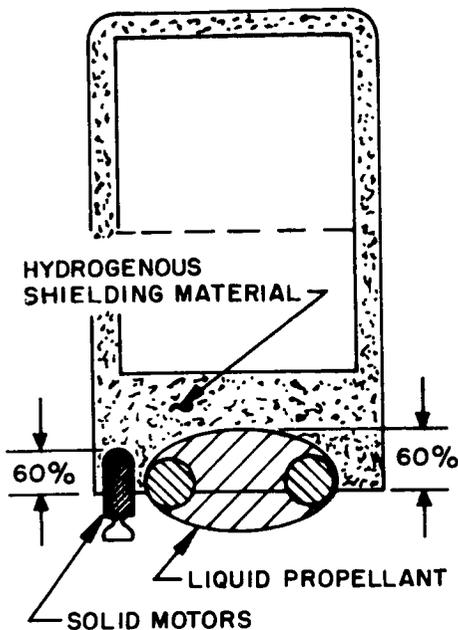


Fig. 3.2--Factors that influence the location of the shielded powered flight station

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 considerations 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 considered 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 diagrammatically the extent that these propellants were integrated into the shielding for the escape vehicles or exploration missions. The quantity of propellant provided for post-escape maneuvers gave the shielded escape vehicle a  $\Delta V$  capability of  $\sim 2,000$  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 catastrophic failure of the basic vehicle be imminent (an uncontrolled fire, explosion, etc.) or should propulsion fail completely while on a catastrophic 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 continuous 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

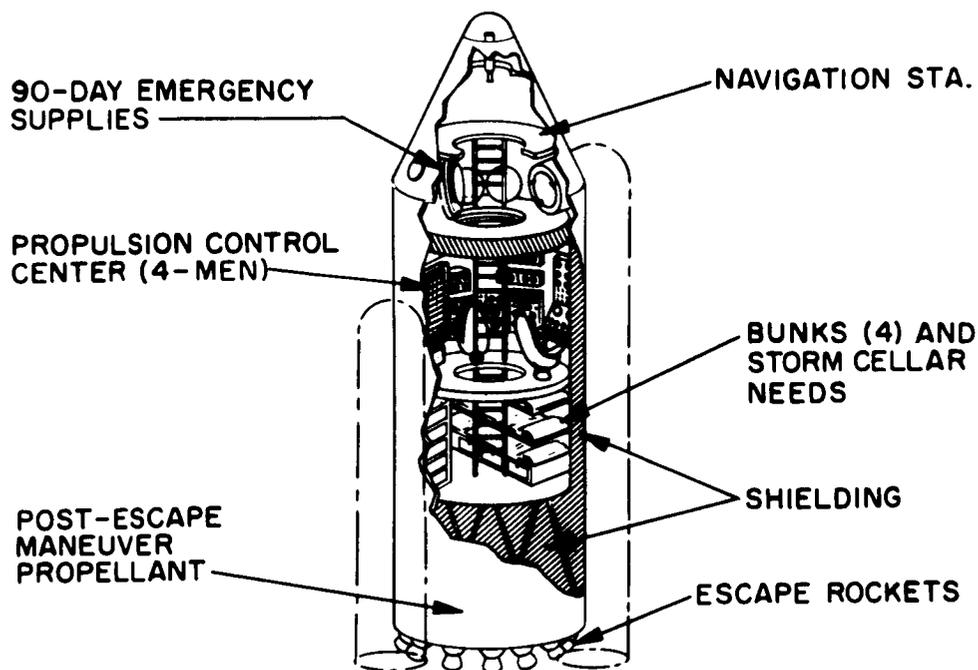


Fig. 3.4--Powered flight station-escape vehicle for 8-man exploration missions with 10-m configurations

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 compartment 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 propulsion 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

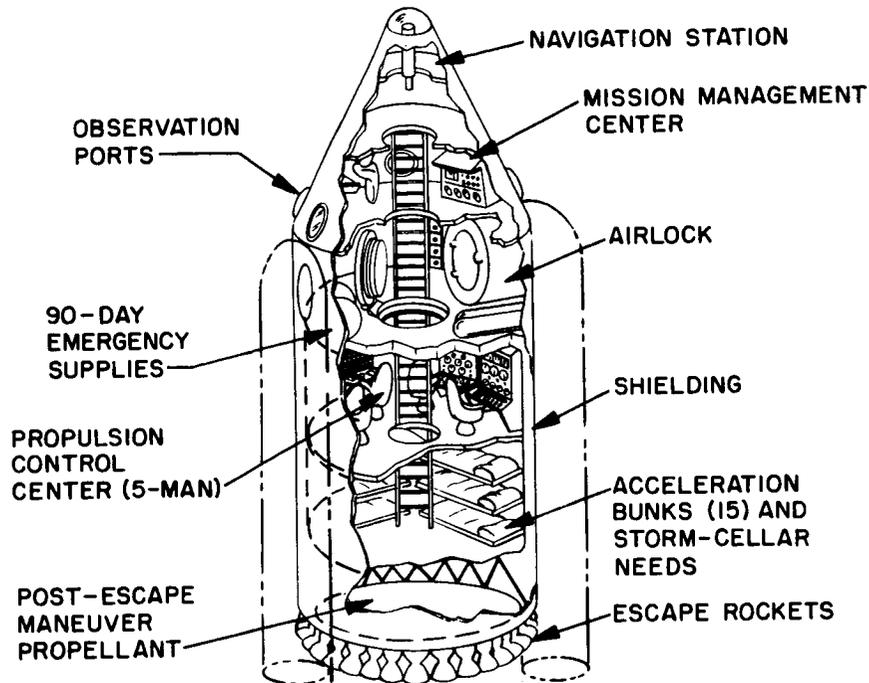


Fig. 3.5--Powered flight station-escape vehicle for 20-man exploration missions with 20-m configurations

(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 personnel in the affected compartment would be lost.

Table 3. 1

**WEIGHT SUMMARY FOR TYPICAL EXPLORATION  
MISSION ESCAPE VEHICLES**

(In kg)

	<u>8-man Vehicle</u>	<u>20-man Vehicle</u>
<b>Vehicle and propellant:</b>		
Shielding . . . . .	18, 170	38, 380
Structure . . . . .	1, 730	5, 520
Motors, tankage and cases . . . . .	600	1, 600
Propellant (escape and maneuvers) . . . . .	<u>4, 500</u>	<u>12, 000</u>
Totals . . . . .	<u>25, 000</u>	<u>55, 500</u>
<b>Contents at separation:</b>		
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 . . . . .	<u>5, 000</u>	<u>10, 500</u>
Total escape-vehicle weights . . . . .	<u>30, 000</u>	<u>66, 000</u>

Previous technical reports and papers on personnel requirements 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 propulsion 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.

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

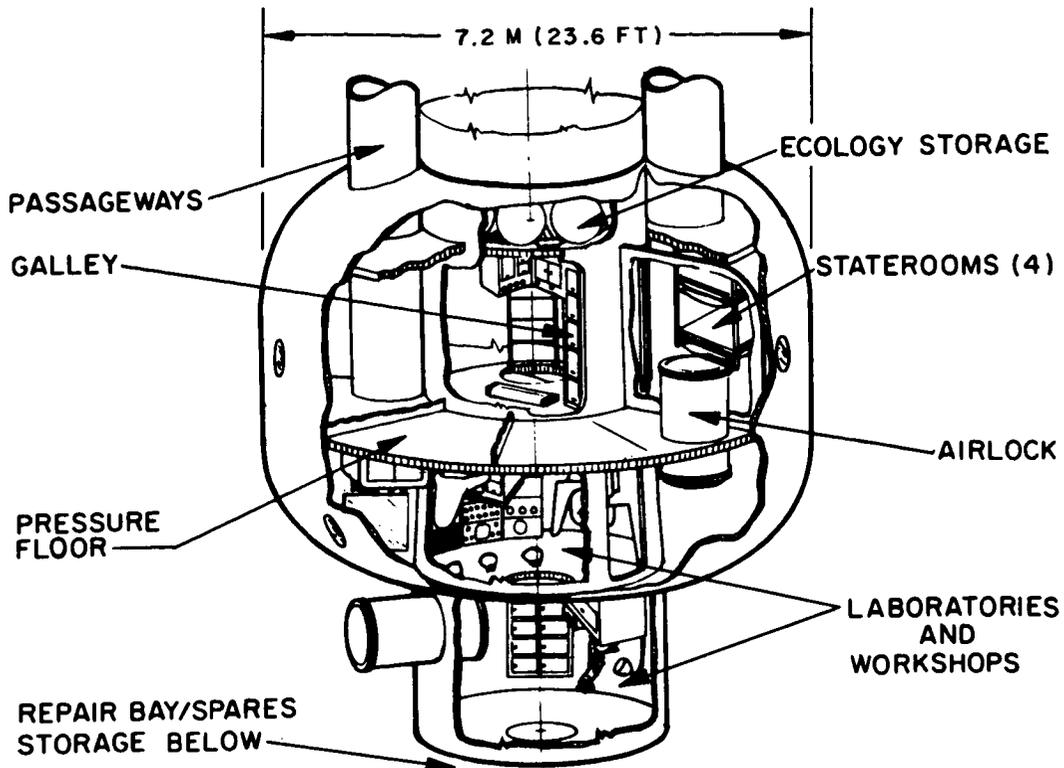


Fig. 3.6--Exploration-mission personnel accommodations for an 8-man complement

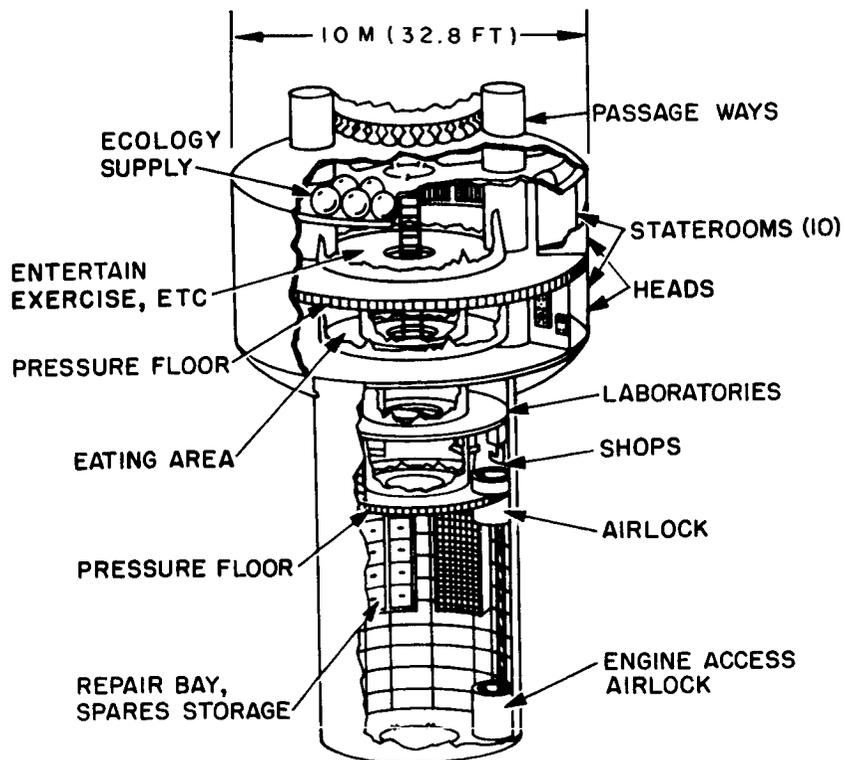


Fig. 3.7--Exploration-mission personnel accommodations for a 20-man complement

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<sup>3</sup> (7,100 ft<sup>3</sup>) or approximately 25 m<sup>3</sup> (880 ft<sup>3</sup>) 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<sup>3</sup> 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<sup>3</sup> (20,800 ft<sup>3</sup>) or about 29.5 m<sup>3</sup> (1,040 ft<sup>3</sup>) 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<sup>3</sup> (820 ft<sup>3</sup>) 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)
<b>Floors</b>	
3.1-m ID pressure floors (3) . . . . .	414
Central large pressure floor (1) . . . . .	706
Upper large nonpressure floor (1) . . . . .	306
Total . . . . .	<u>1,426</u>
<b>Exterior surfaces</b>	
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 . . . . .	<u>3,070</u>
<b>Internal vertical surfaces</b>	
Internal spine extension . . . . .	500
Floor-supporting walls . . . . .	470
Nonstructure partition walls . . . . .	120
Total . . . . .	<u>1,090</u>
<b>Miscellaneous internal structures</b>	
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 . . . . .	<u>1,287</u>
Contingency (~6 percent) . . . . .	<u>727</u>
Total structure weight . . . . .	<u><u>7,600</u></u>

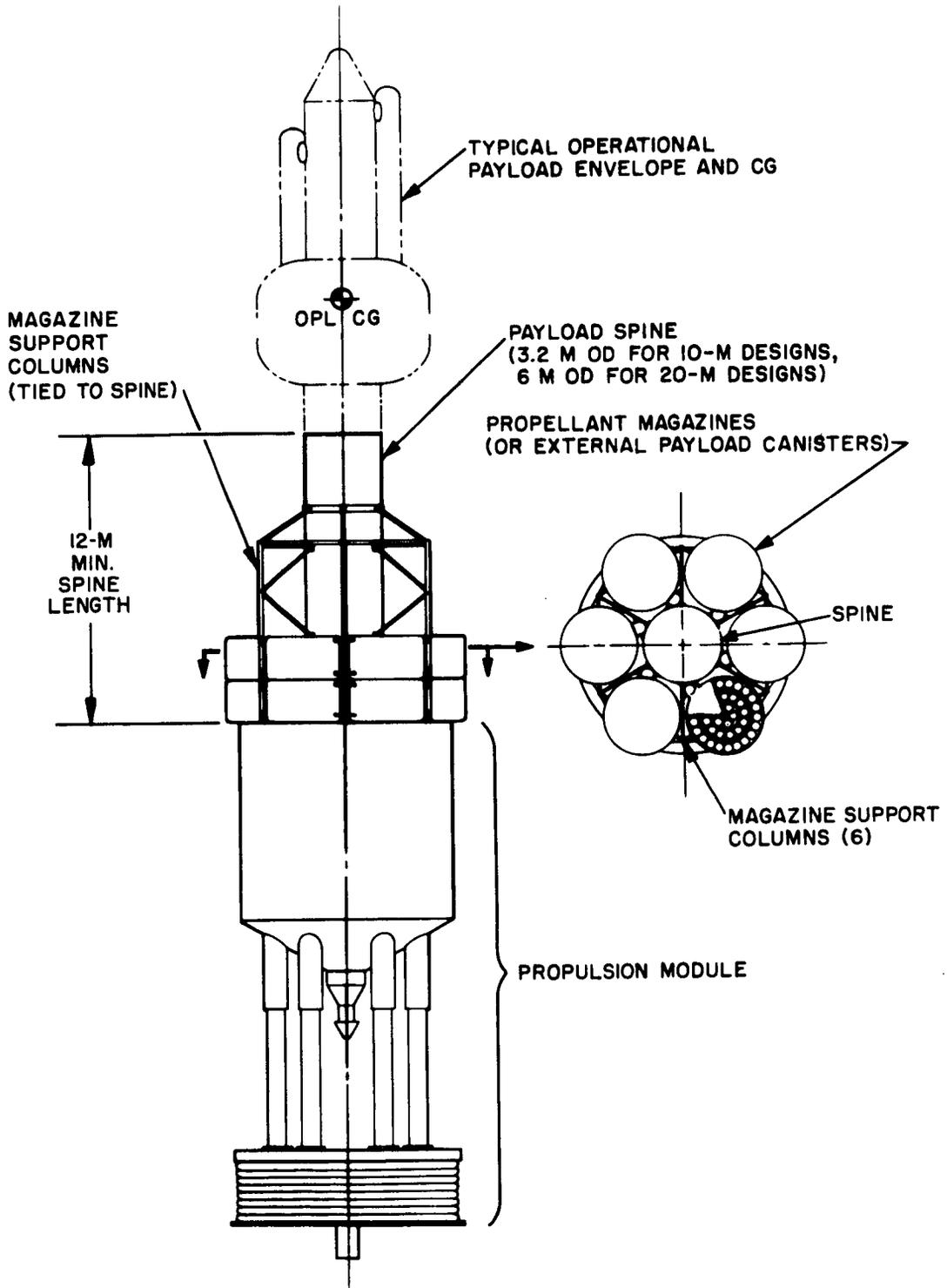


Fig. 3. 8--General arrangement of payload spine and magazine payload support columns

One other personnel-accommodations configuration for exploratory 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. However, 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 propulsion 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, personnel 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 structural 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 structurally 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

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\*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  $400 \text{ g/m}^3$  ( $25 \text{ lb/ft}^3$ ) 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.

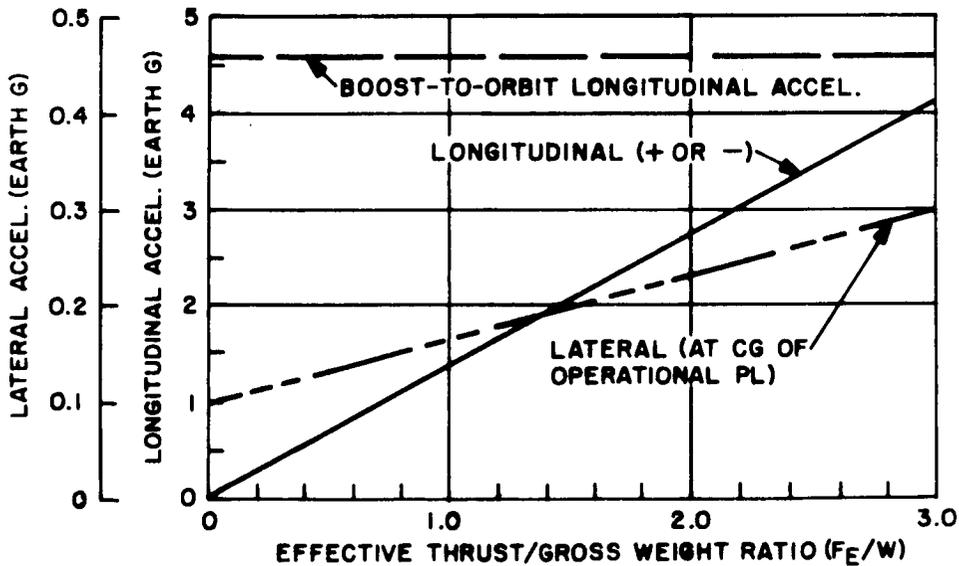


Fig. 3.9--Estimated critical design accelerations for upper vehicle

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 burn-out 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.)

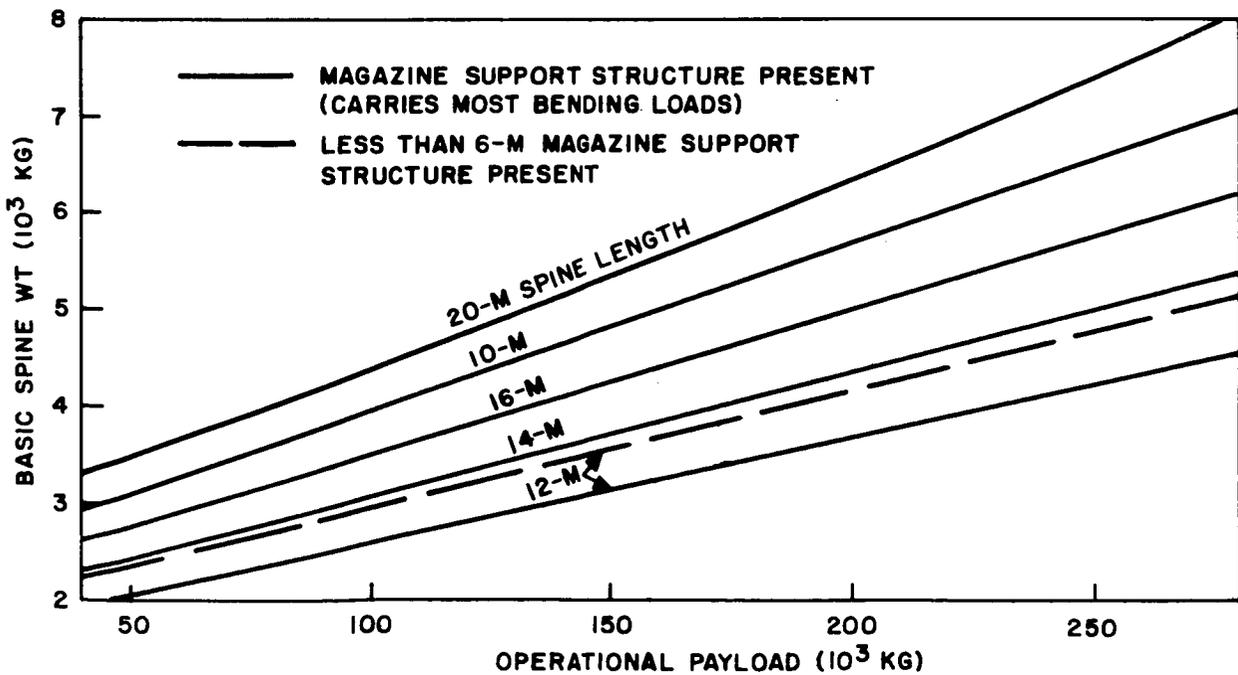


Fig. 3.10--Payload spine weights as functions of payload carried and length of spine for 10-m configurations

### 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 accessways. 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  $\Delta V$ ) 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.)

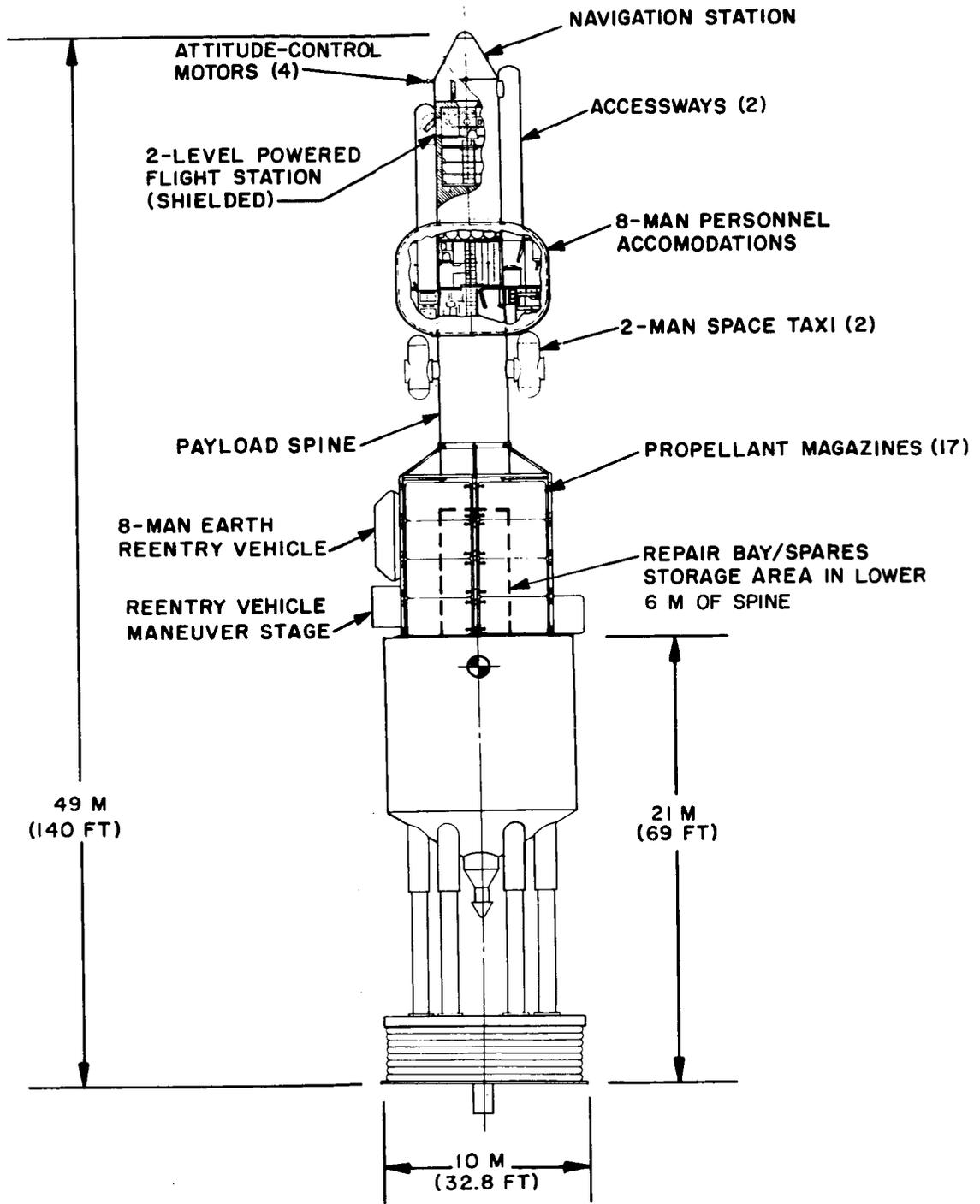


Fig. 3.11--Exploration vehicle configuration for Mars orbital capture mission using 10-m propulsion module

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 . . . . .	<u>11,600</u>
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 . . . . .	<u>9,100</u>
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 . . . . .	<u>2,977</u>
Expendables	
Main ecological system . . . . .	3,515
Emergency and reserve life-support system . . . . .	3,810
Food . . . . .	5,398
Total . . . . .	<u>12,723</u>
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

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.

\*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.

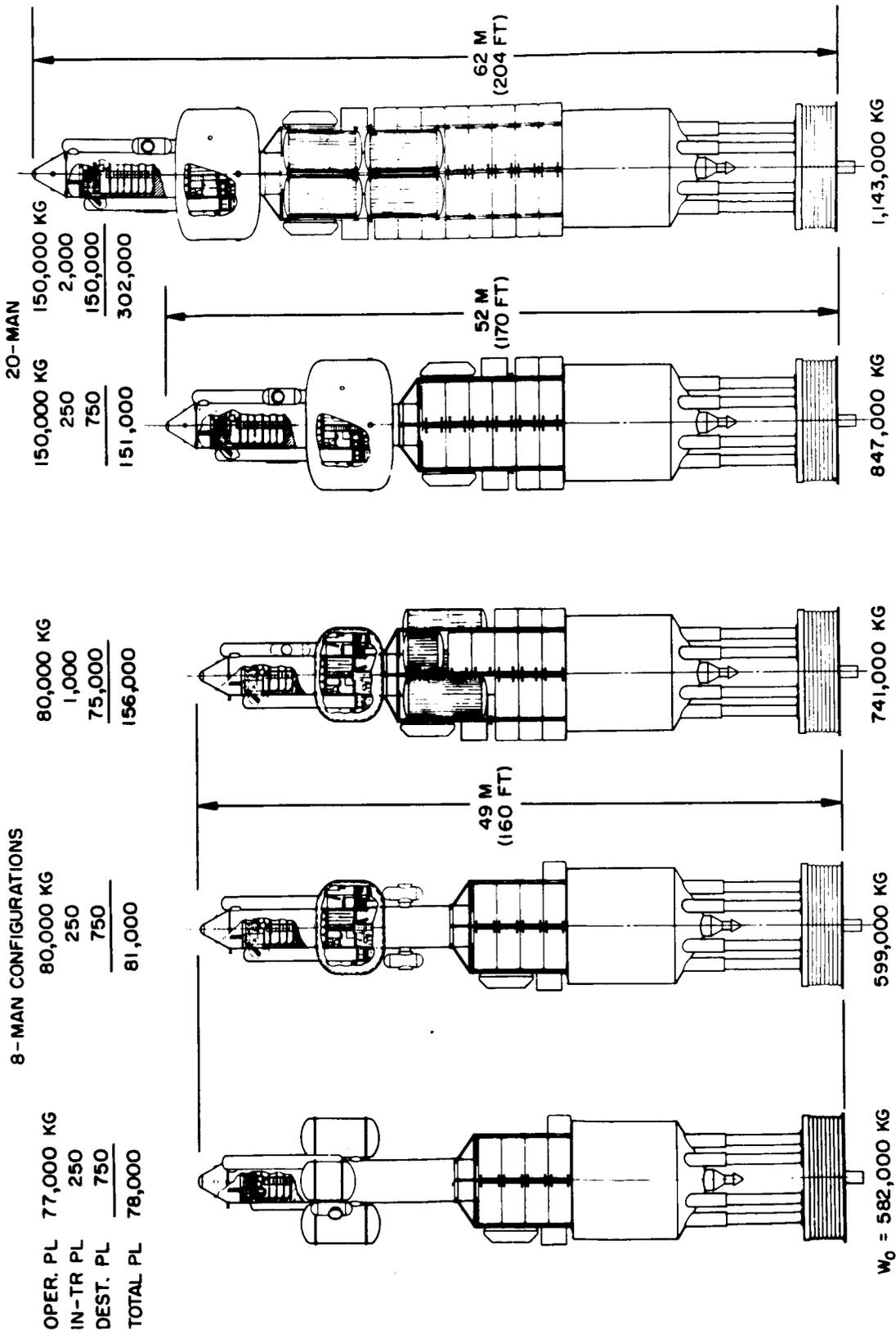


Fig. 3.12-- Various payload configurations on basic 10-m propulsion module (with departure weights for 72, 850 fps Mars mission)

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 performing 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

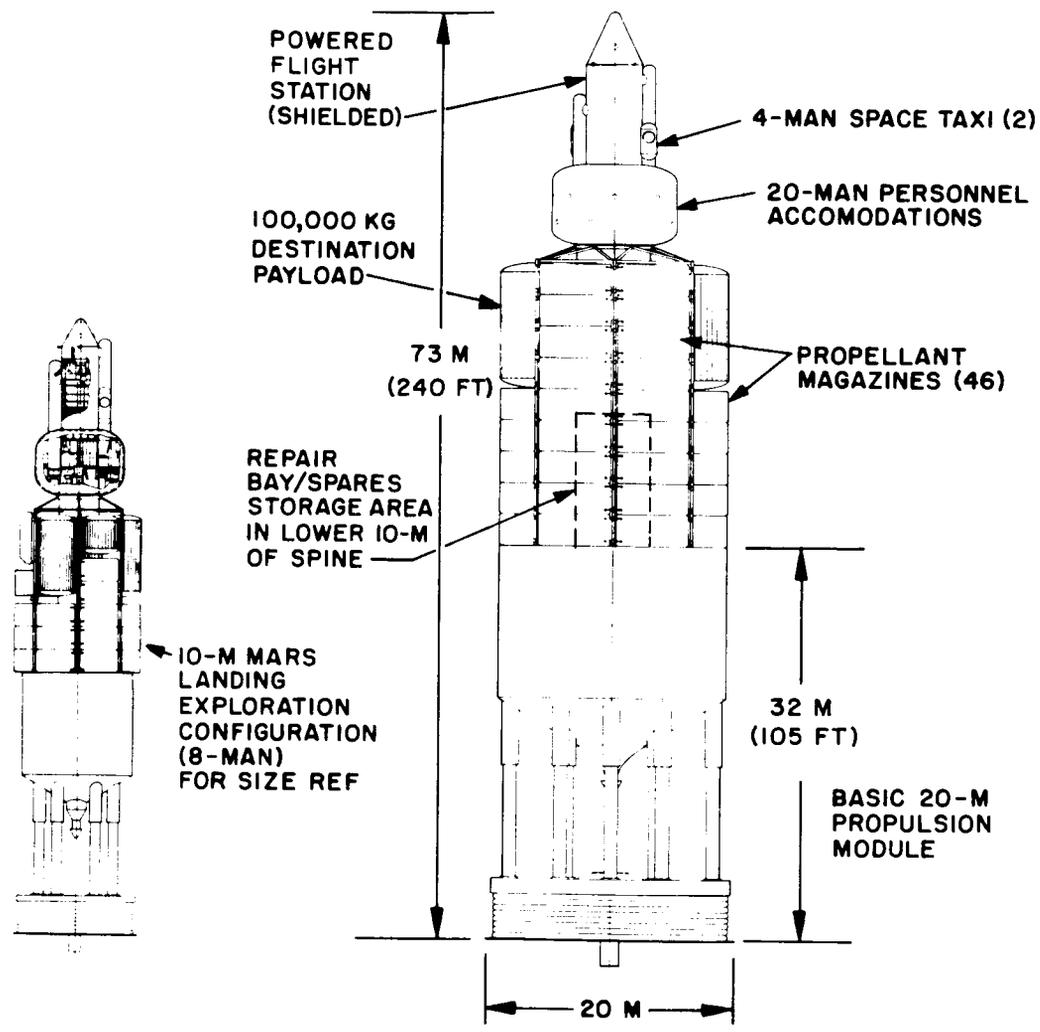


Fig. 3.13--Exploration vehicle configuration for Jupiter moon landing mission, 20-m propulsion module

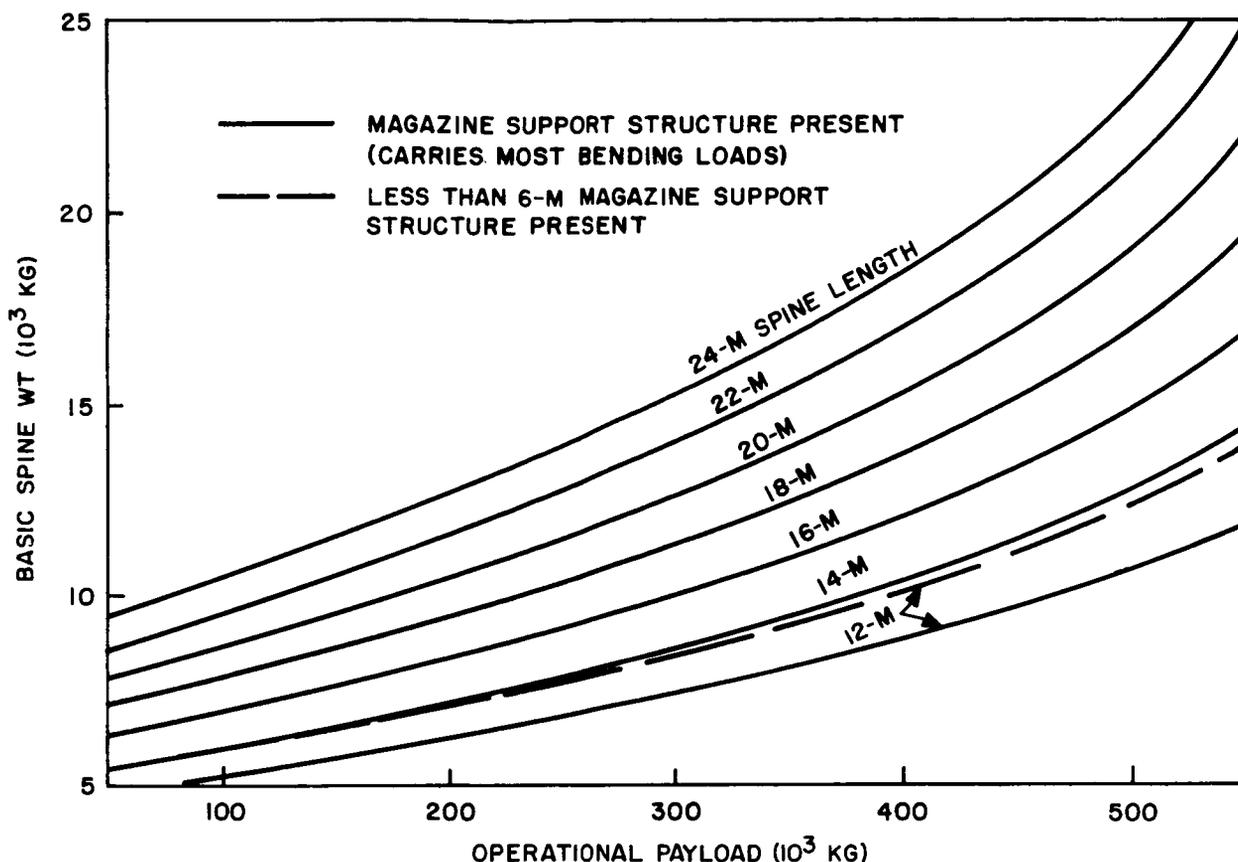


Fig. 3. 14--Payload spine weight as function of payload carried and length of spine for 20-m configurations

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.



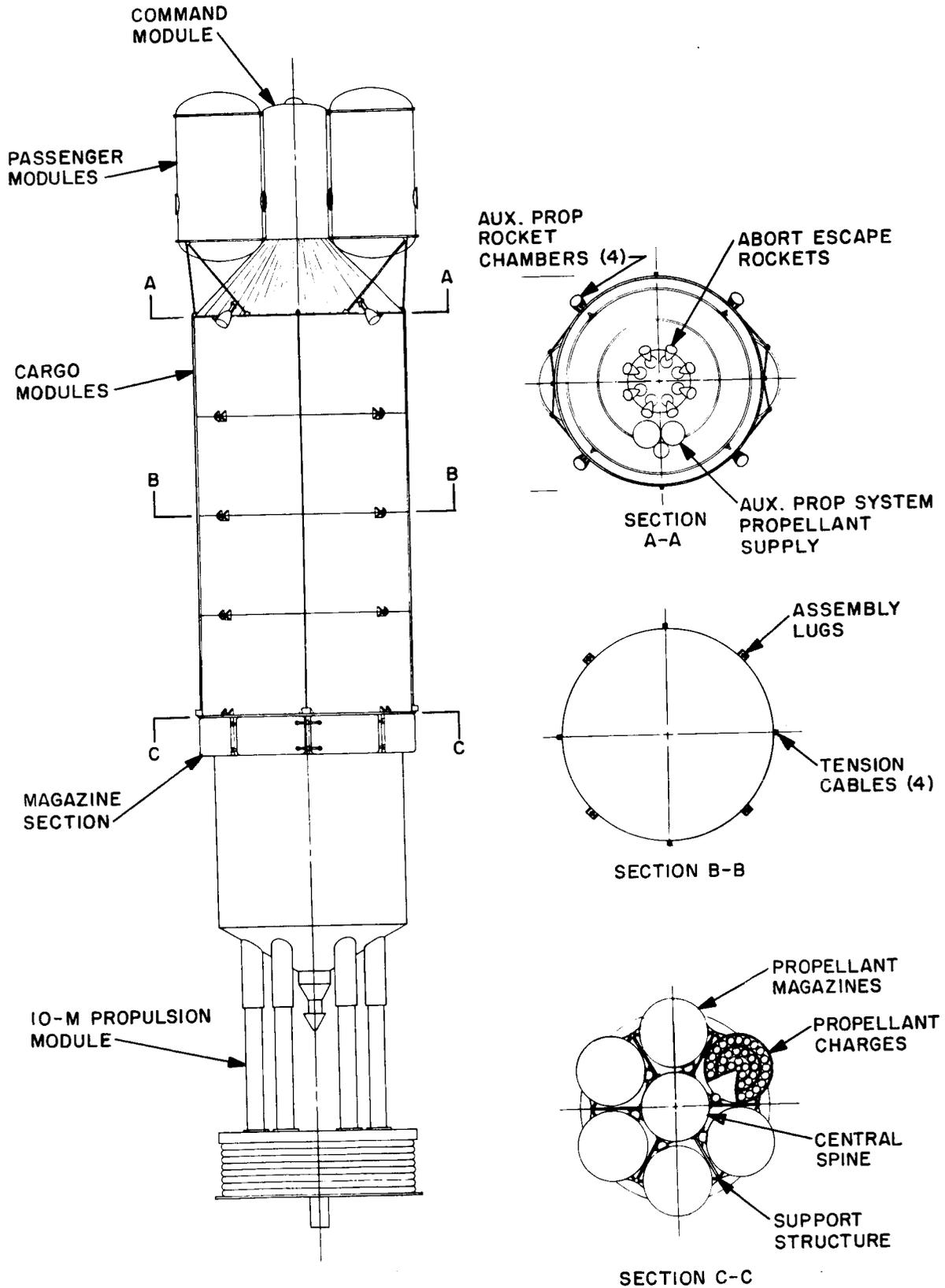


Fig. 3. 16--Earth-orbit-to-lunar-orbit ferry vehicle

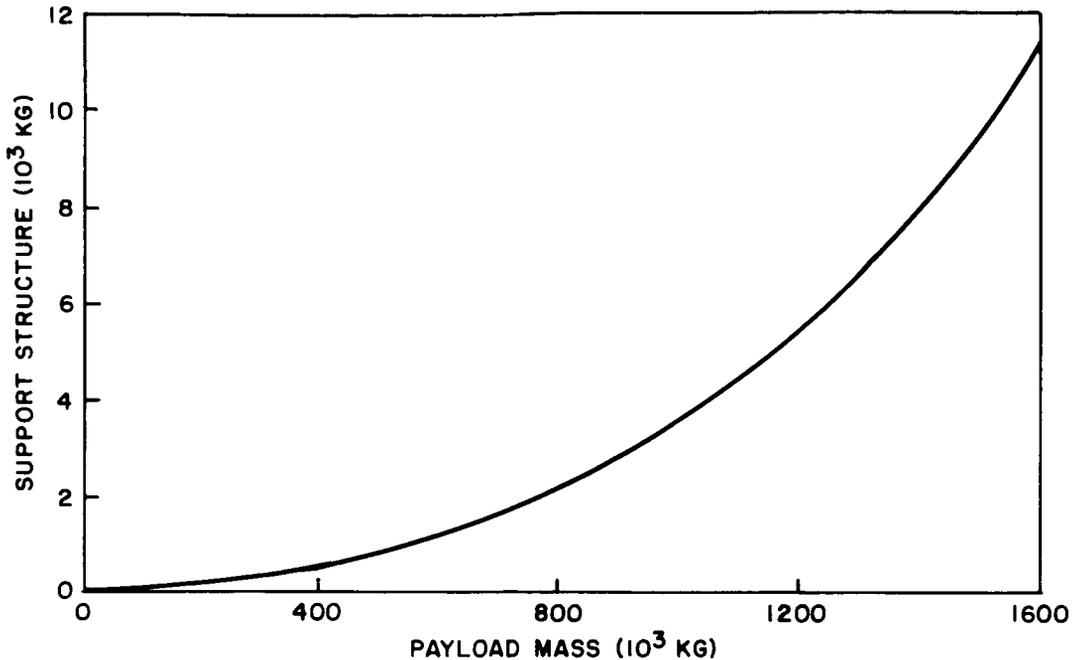


Fig. 3.17--Estimated support-structure mass

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<sup>3</sup> (20 lb/ft<sup>3</sup>; composite density of O<sub>2</sub>/H<sub>2</sub> 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 propulsion 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<sup>2</sup> (50 ft<sup>2</sup>) for the crew of three. The lower compartment is sized on the basis of 5 m<sup>3</sup> (180 ft<sup>3</sup>) per man, assuming that no more than two of the crew are off duty at one time.

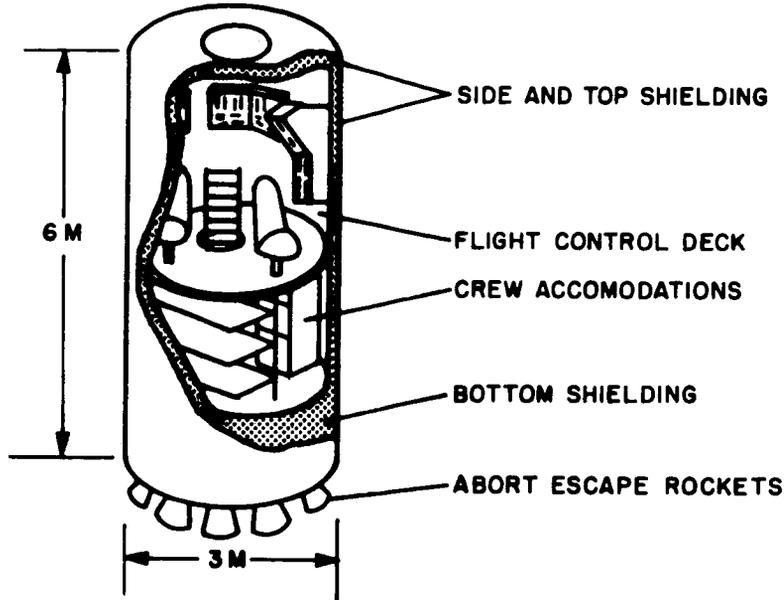


Fig. 3.18--Lunar-ferry-vehicle command module

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:	{ 110 g/cm <sup>2</sup> hydrogenous material (neutron),
	55 g/cm <sup>2</sup> lead or equivalent (gamma).
Side and top:	25 g/cm <sup>2</sup> 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:      { Polyethylene 110 cm  
                  { Lead                29 cm

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<sup>3</sup> (180 ft<sup>3</sup>) is provided for each passenger.

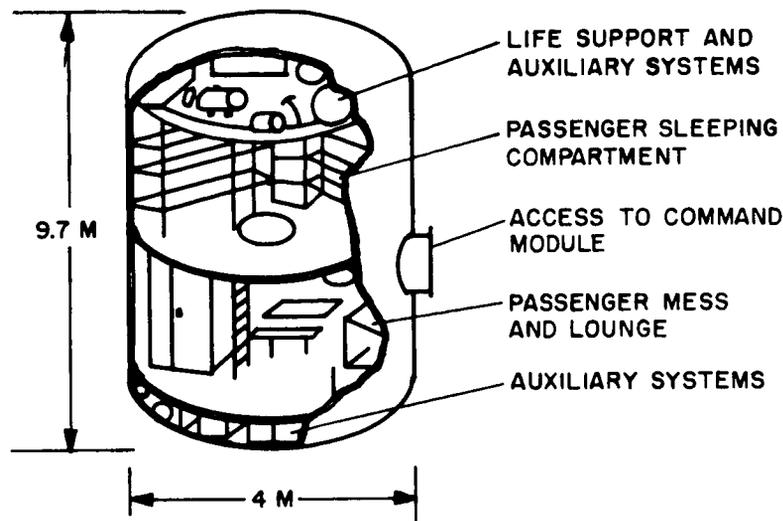


Fig. 3.19--Reference-design passenger module

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
	<hr/>
	4,180
Contingency (5 percent) . . . . .	210
Total mass . . . . .	<hr/>
	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

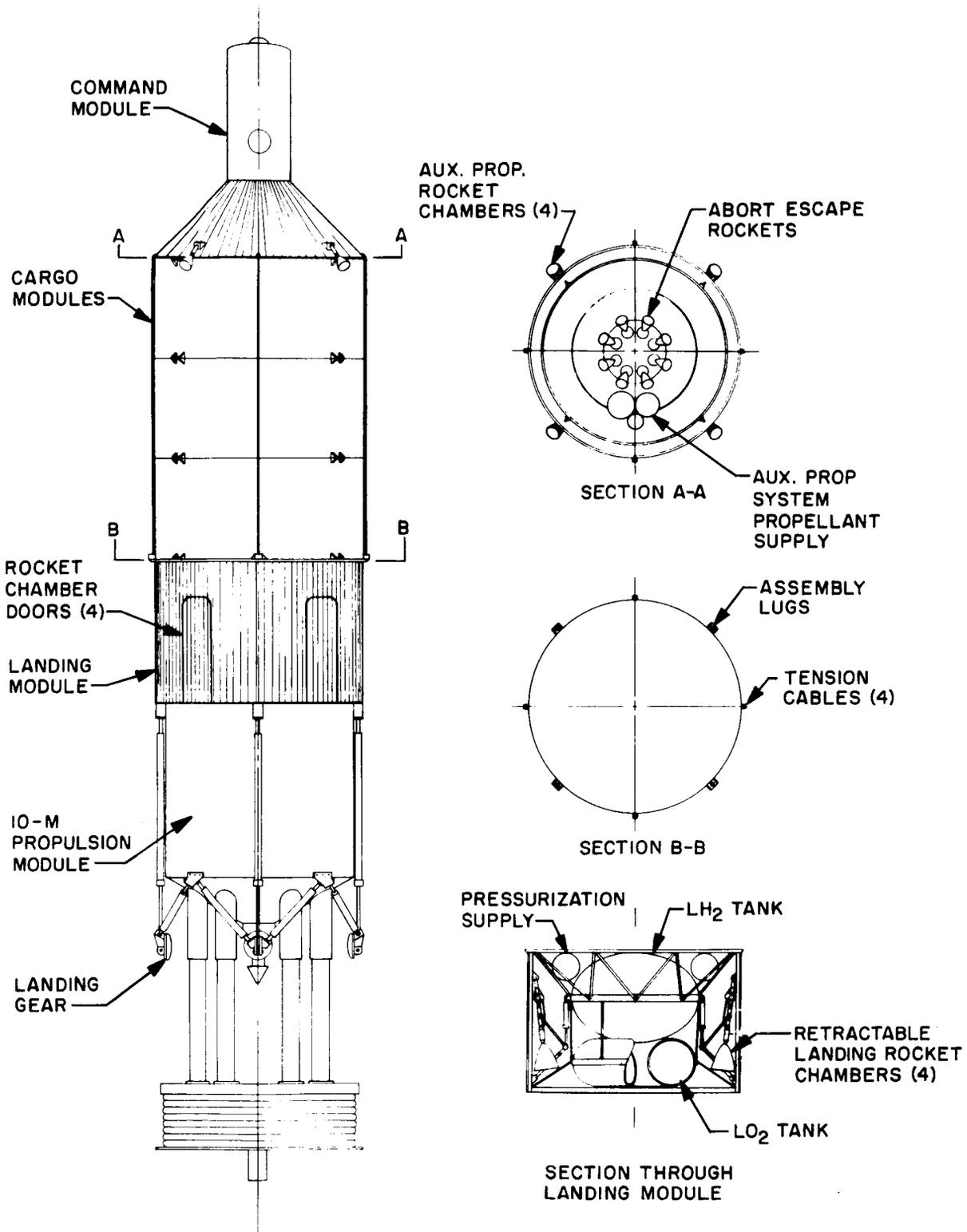


Fig. 3. 20--Earth-orbit-to-lunar-surface ferry vehicle

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

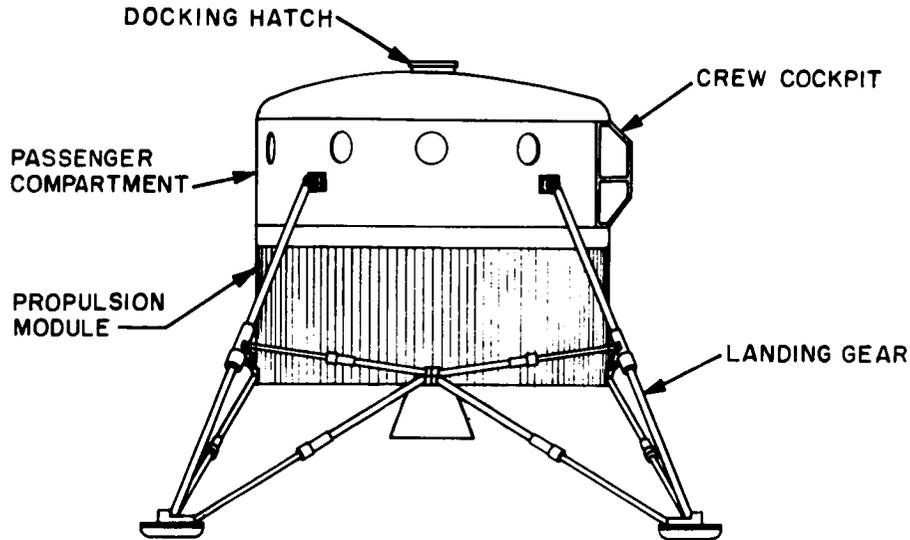


Fig. 3.21--Lunar passenger ferry

The weight summary for the passenger shuttle is given in Table 3.6. The passenger-cabin size was based on a requirement of  $2.5 \text{ m}^3$  ( $88 \text{ ft}^3$ ) 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 integrated 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.

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 . . . . .	<u>3, 350</u>
Total . . . . .	7, 930
Contingency (5 percent) . . . . .	<u>400</u>
Total passenger module . . . . .	8, 330
Propulsion module . . . . .	<u>1, 610</u>
Total weight . . . . .	<u><u>9, 940</u></u>

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 . . . . .	<u>400</u>
Total weight . . . . .	<u><u>3, 350</u></u>

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:

$$M_{BO} = 3, 520 + 0. 05 M_{\lambda} + 0. 14 M_P,$$

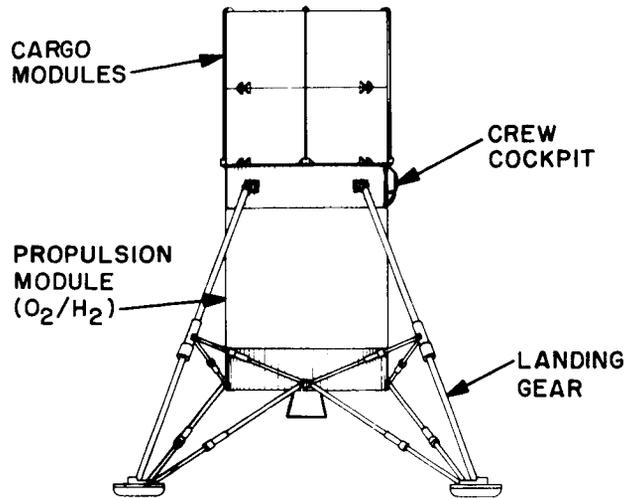


Fig. 3.22--Lunar cargo shuttle

where  $M_{BO}$  = burnout mass, kg,  
 $M_{\lambda}$  = 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

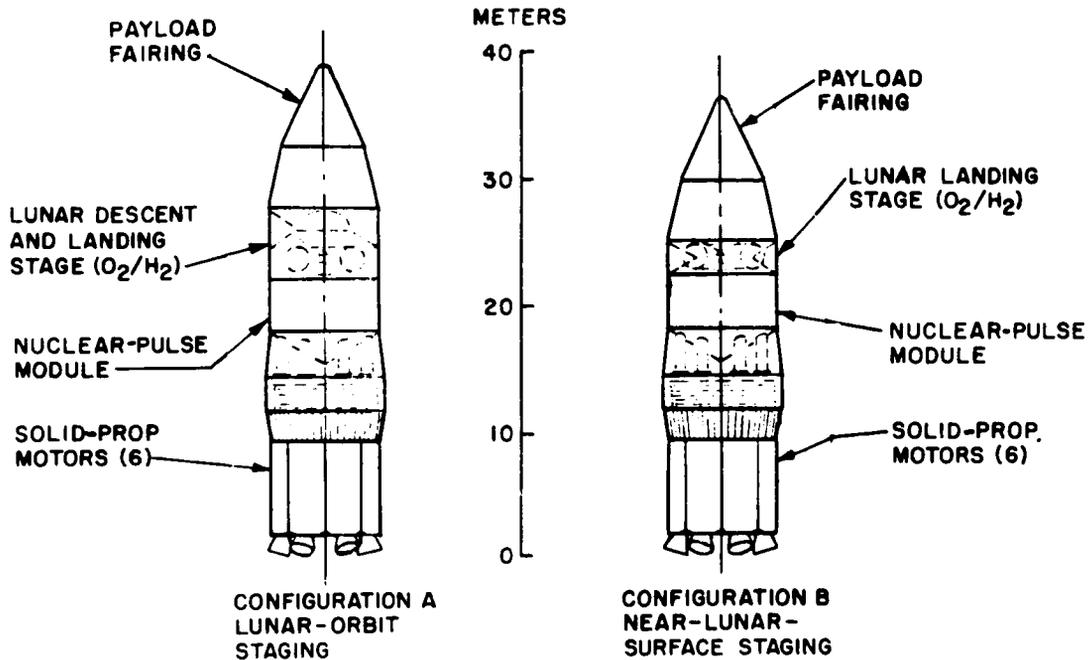


Fig. 3.23--Solid-propellant-boosted, earth-launched lunar logistic vehicles

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:

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

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

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

$M_P$  = propellant mass, kg,

$M_\lambda$  = 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.

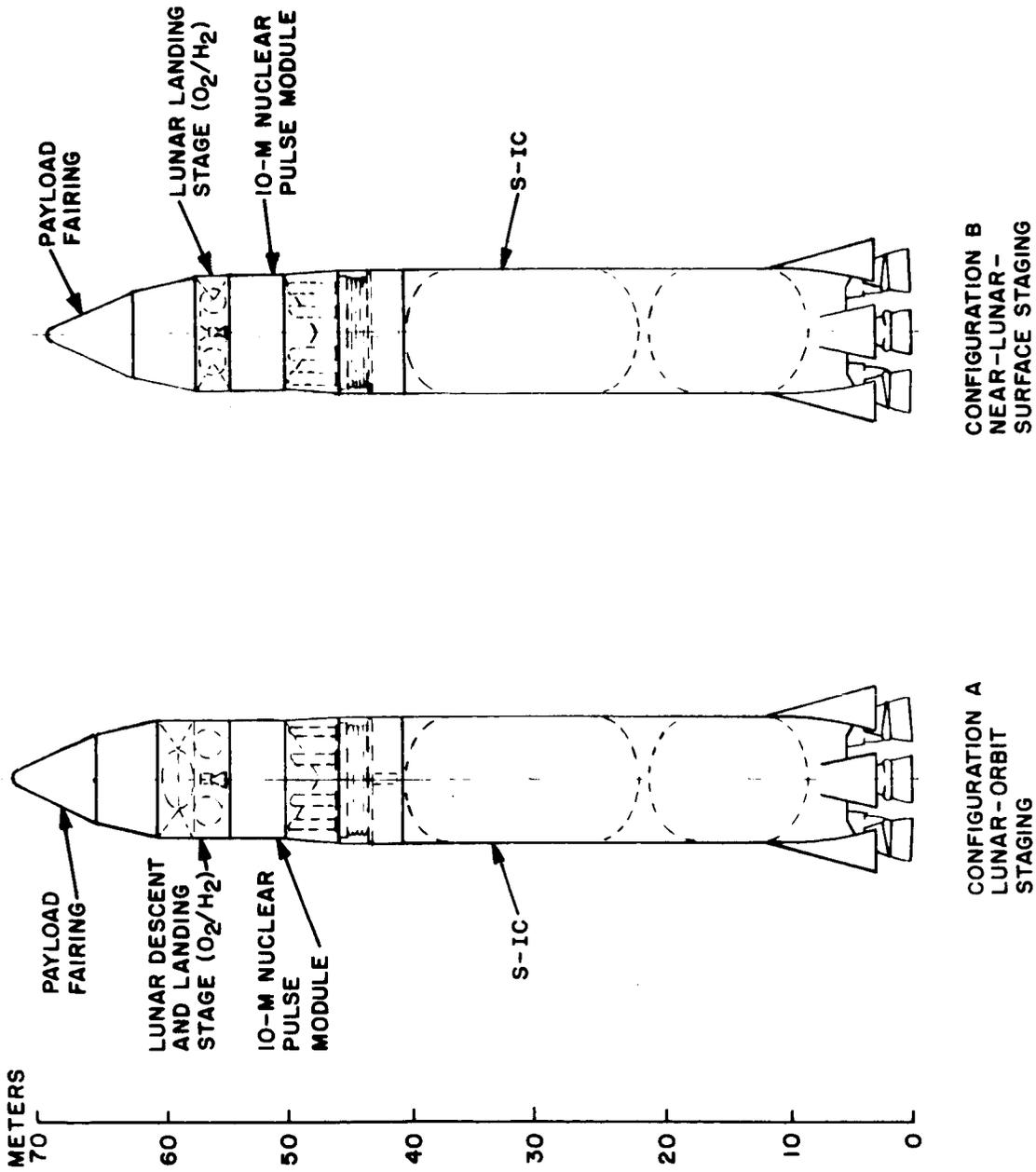


Fig. 3.24--S-1C boosted earth launched lunar logistic vehicle

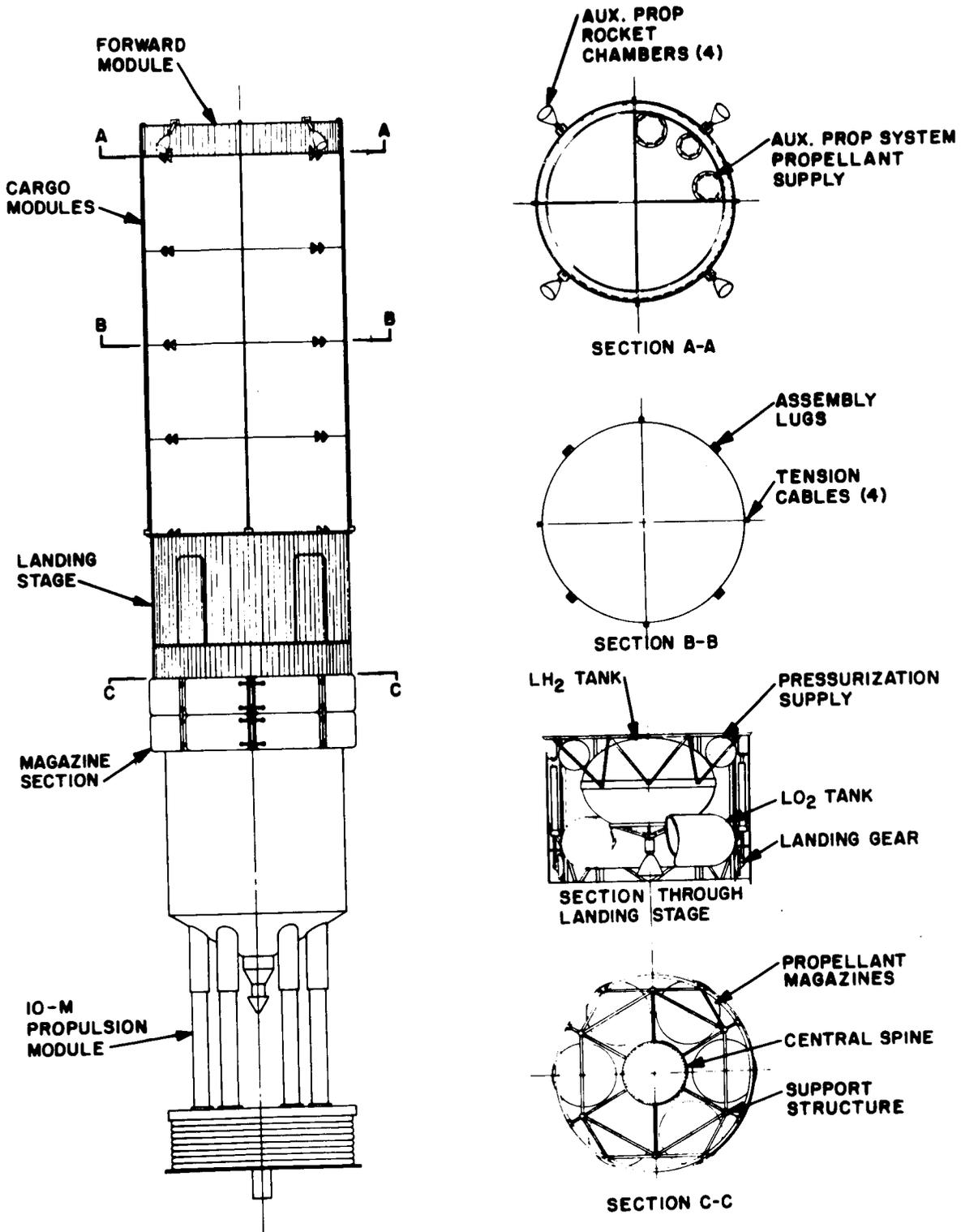


Fig. 3. 25--Orbit launched lunar logistic vehicle

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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,

$M_\lambda$  = payload mass, kg.

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

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#### 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 overall 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.

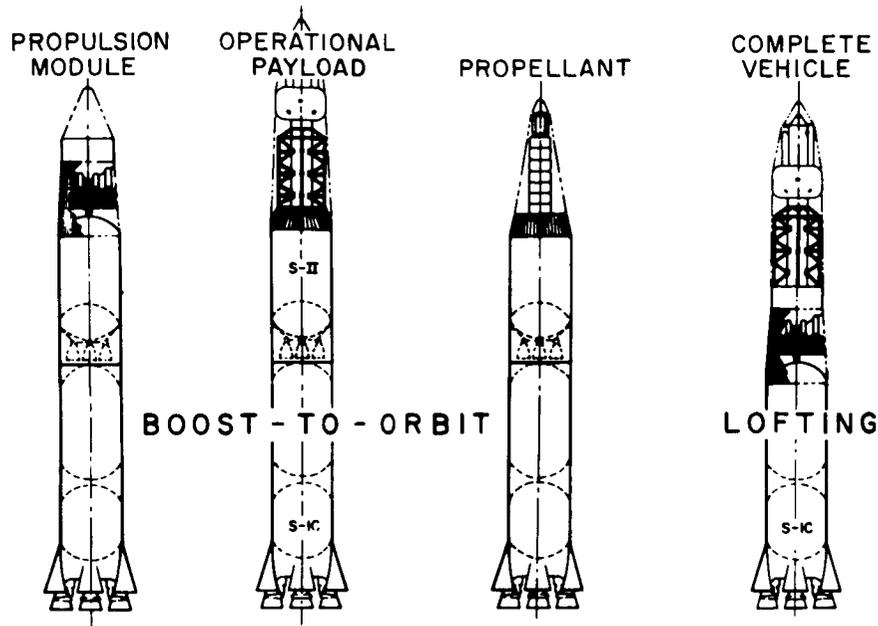


Fig. 4.1--Saturn V with three exploration mission payloads for orbital rendezvous and the S-1C stage as a complete-vehicle lifter

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 (~ 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

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

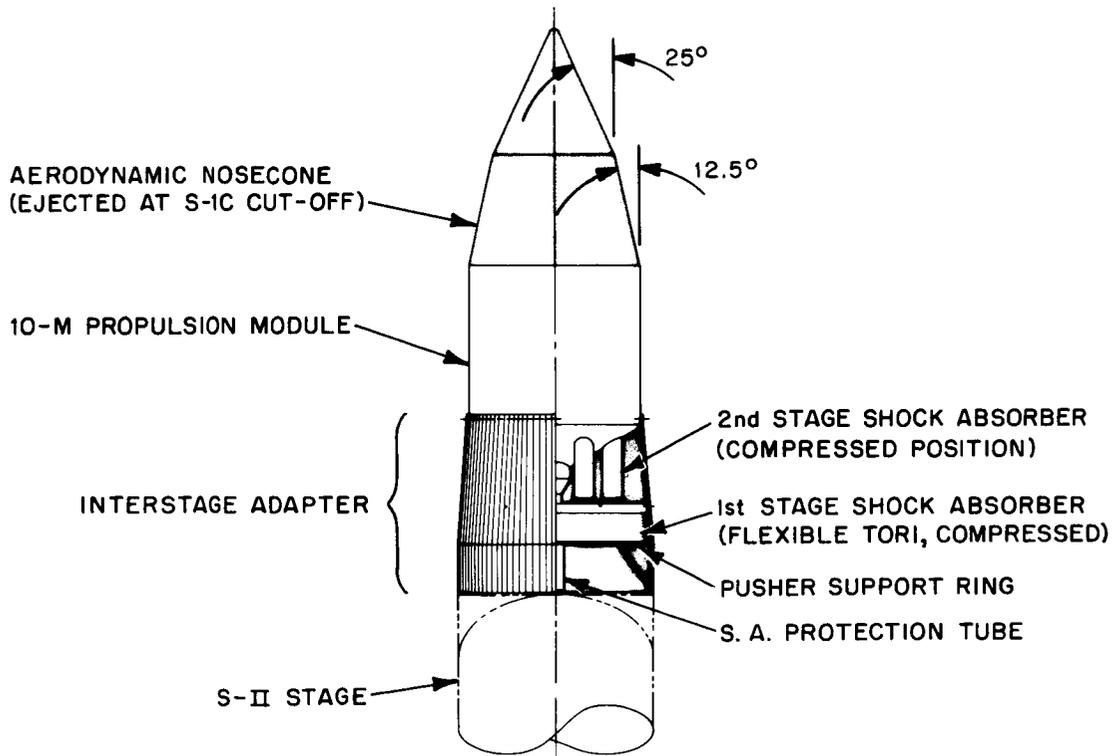


Fig. 4.2--Interstage adapter and nosecone concepts for 10-m propulsion module boost-to-orbit

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.) . . . . .	<u>1,500 lb</u>
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

\* L. K. Linsley, "Nose Shape on Saturn Vehicles," George C. Marshall Space Flight Center, Memorandum No. M-AERO-A-20-63, March 6, 1963.

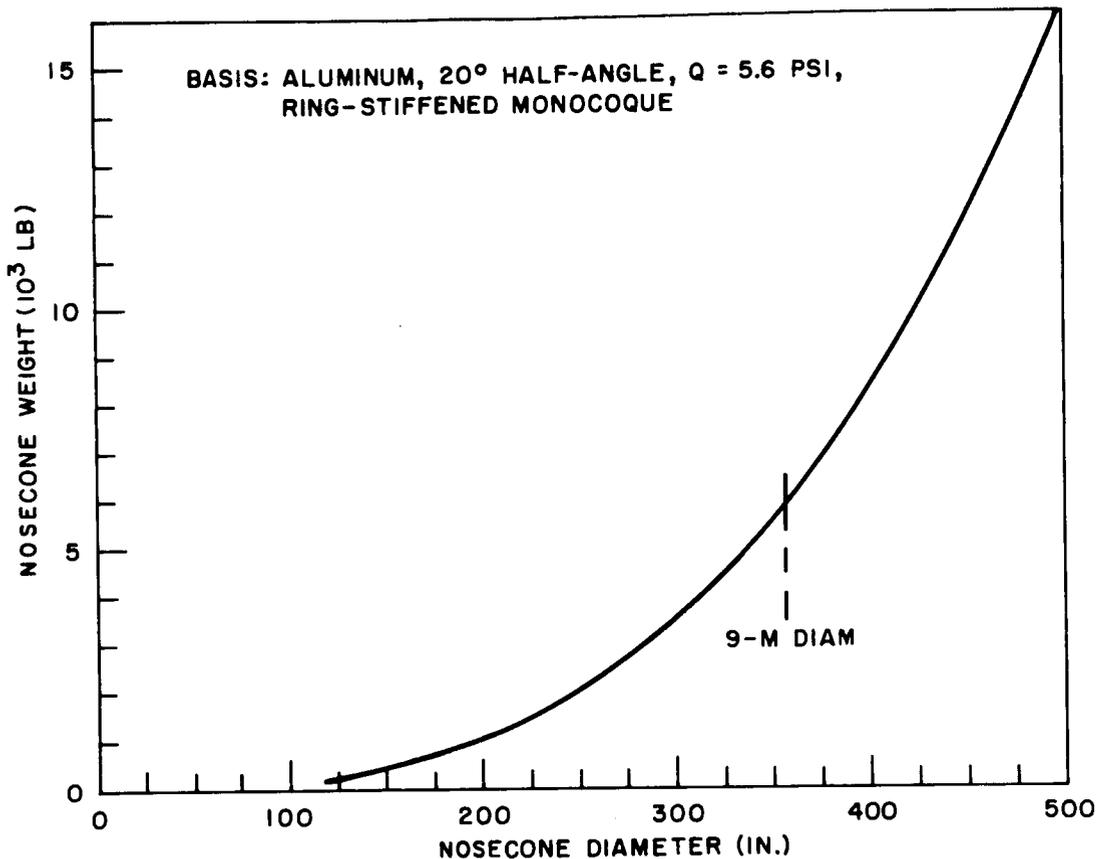


Fig. 4.3--Typical nosecone weight versus diameter data (taken from Nevins and Helton)

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.

\*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) . . . . .	<u>2,000 lb</u>
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

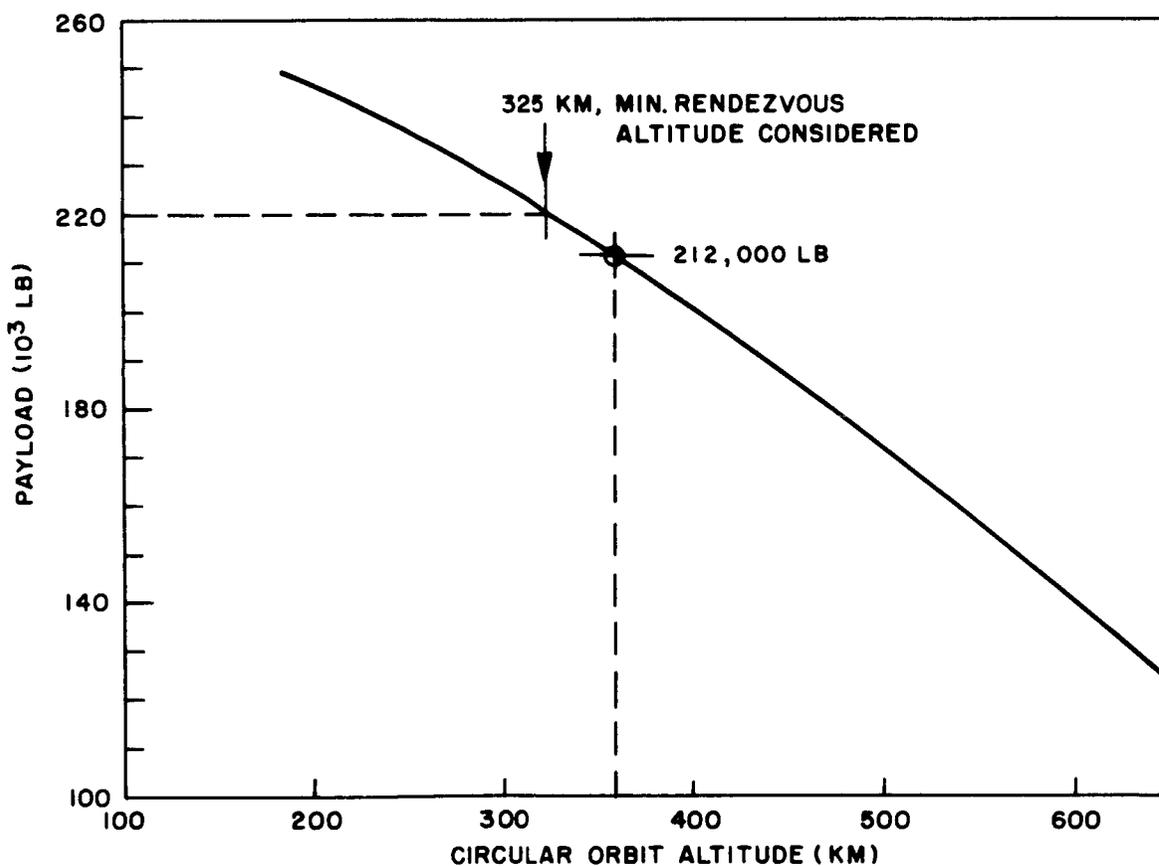


Fig. 4.4--Payload versus orbital altitude for a two-stage Saturn V launch vehicle (11-20-63 data)

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 lifter 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 lifters 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  $\$60 \times 10^6$ . Its DOC per pound of gross payload is therefore  $\$60/\text{lb}$  (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

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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.

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## 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  $2\pi$  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  $2 \times 10^3$  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  $2\pi$  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  $\beta^-$  decay of fission products. The STARFISH tests were detonated reasonably close to the magnetic equator and injected  $\beta$  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 latitude  $40^\circ$  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.

\*  $L$  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.

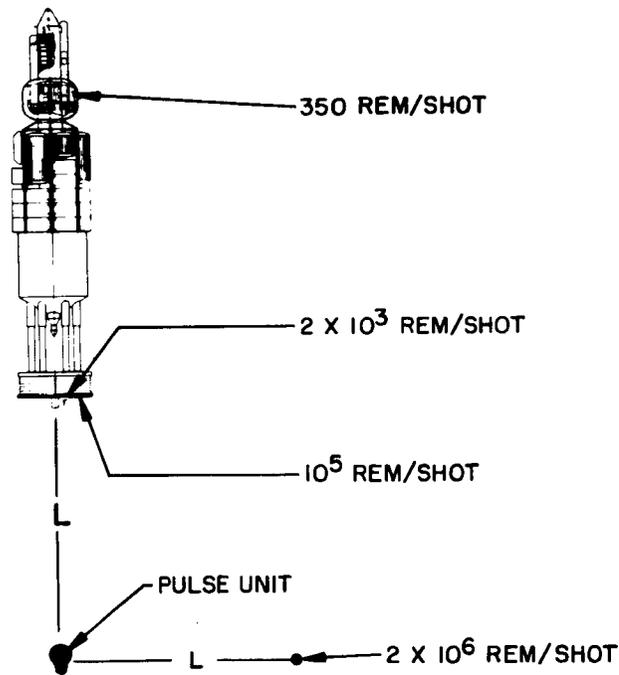


Fig. 5.1--Operational nuclear environment about the 10-m propulsion module

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  $3.5 \times 10^6$  newtons (10-m module) to  $10^3$  rem/shot at a thrust of  $100 \times 10^6$  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.

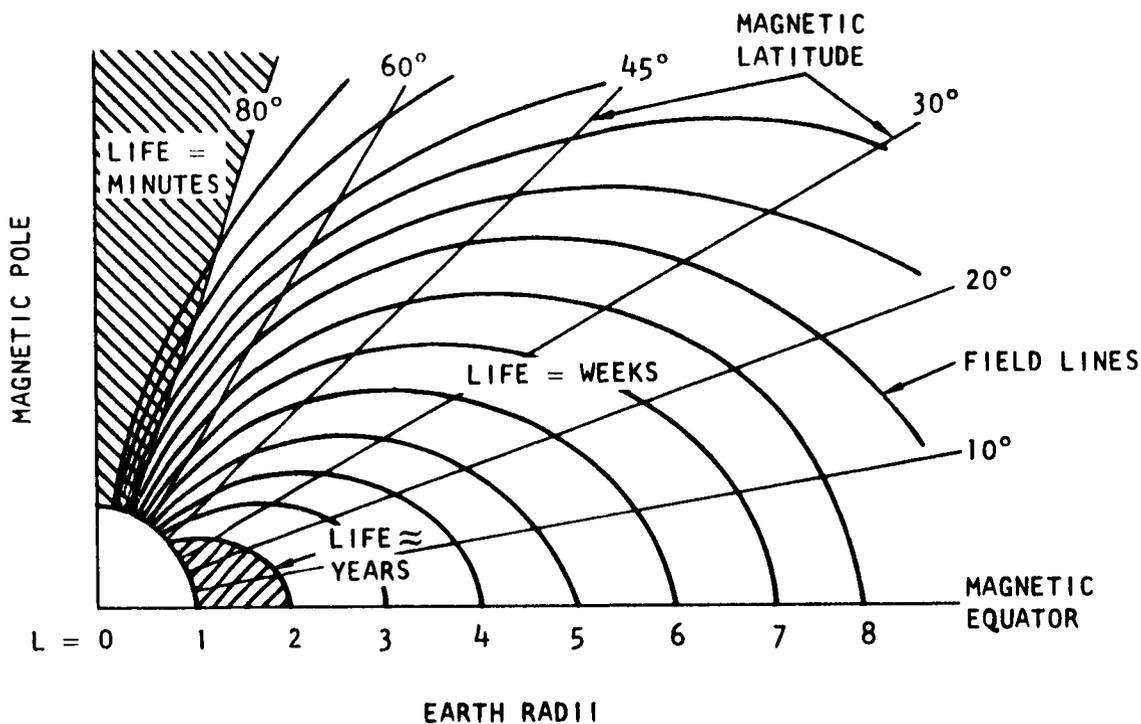


Fig. 5.2--Trapped electron lifetime summary

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  $\eta$  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 requirements as a function of altitude. It is estimated that a  $\phi'$  greater than  $0.6 \text{ cal/cm}^2$  can cause retinal burn. By evaluating the radiative flux for several different yields by use of the PEEP code,\* it was determined conservatively 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 existence (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.

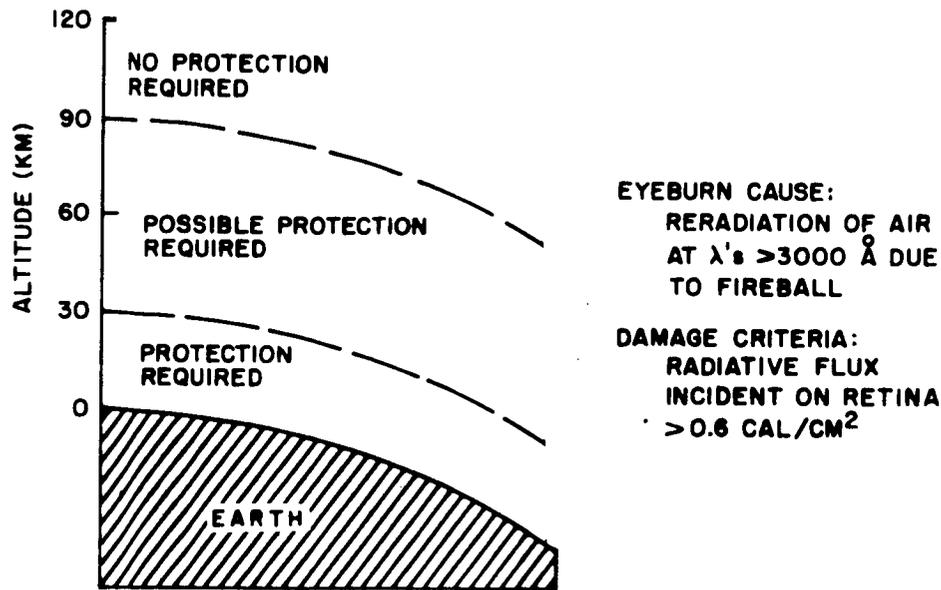


Fig. 5.3--Preliminary assessment of eyeburn conditions

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.

\* 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.

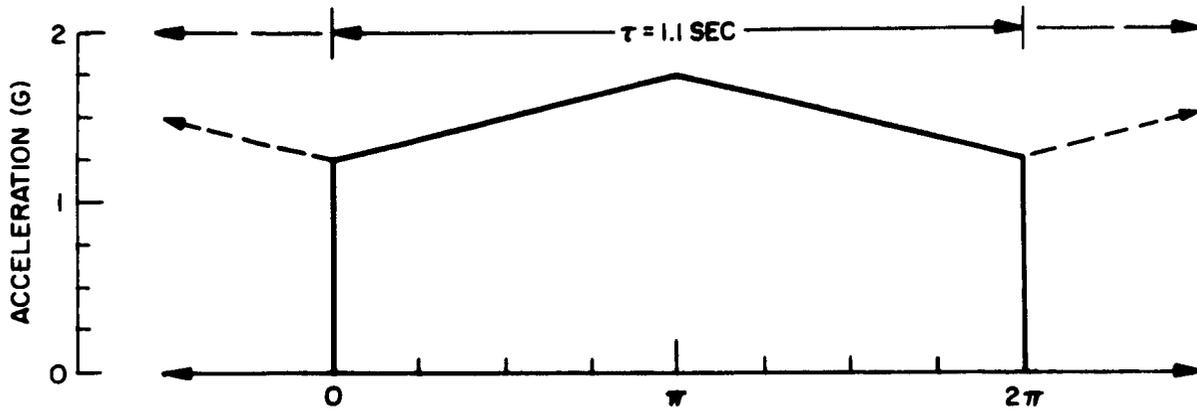
Figure 5.5 shows the envelope curves drawn through the harmonic amplitudes of the functions shown in Fig. 5.4. The important points to be noted are: (1) Below about 6 cps, the relative harmonic amplitudes between Case 1 and Case 2 are within one order of magnitude. (2) At the higher frequencies, Case 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

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

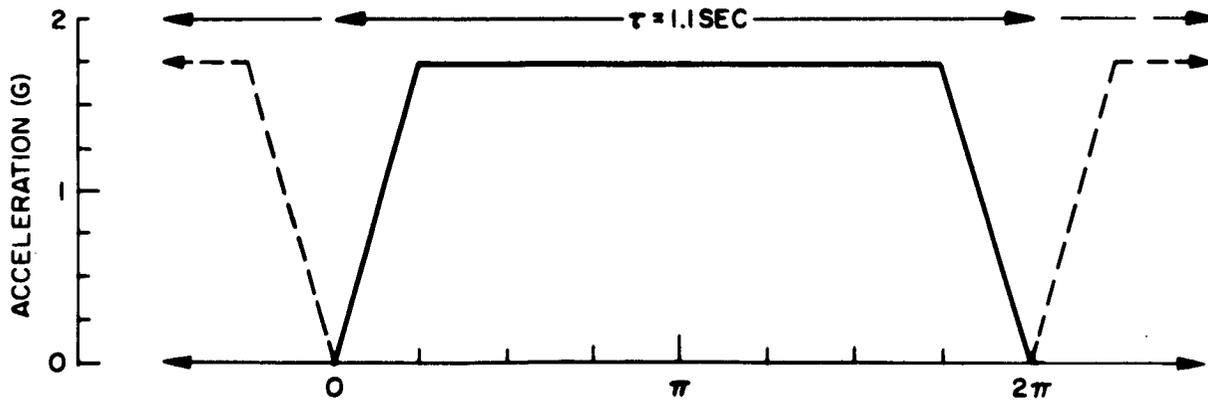
where  $C_L$  is the longitudinal wave speed in steel,  $A$  is the floor area, and  $H$  is the equivalent thickness of the floor. For the floor dimensions provided, the modal density computes to be about 0.10 modes/cps.

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.



$$\begin{aligned} f(x) &= 0 && \text{FOR } x = 0 \text{ AND } 2\pi \\ &= (1.25 + mx) && \text{FOR } 0 < x < \pi \\ &= 1.75 && \text{FOR } x = \pi \\ &= 1.25 + m(2\pi - x) && \text{FOR } \pi < x < 2\pi \end{aligned}$$

CASE 1



$$\begin{aligned} f(x) &= 0 && \text{FOR } x = 0 \text{ AND } 2\pi \\ &= 7x/\pi && \text{FOR } 0 < x < \pi/4 \\ &= 1.75 && \text{FOR } \pi/4 < x < 7\pi/4 \\ &= 14 - 7x/\pi && \text{FOR } 7\pi/4 < x < 2\pi \end{aligned}$$

CASE 2

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

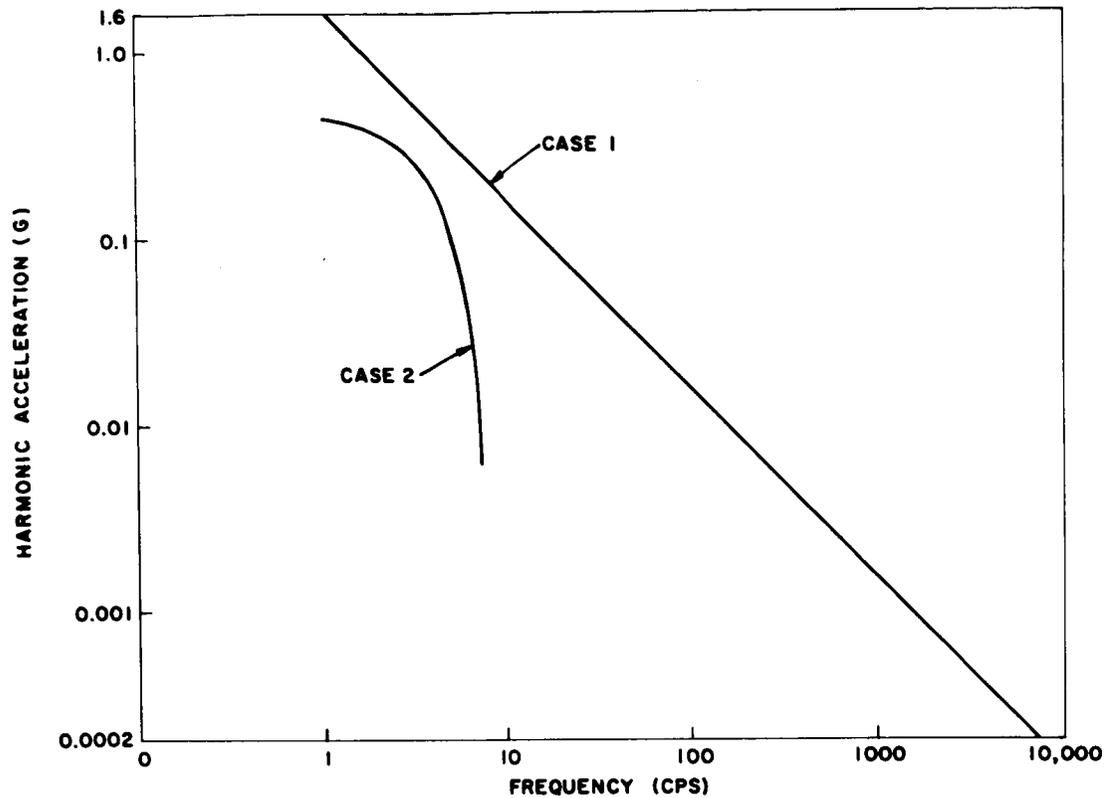


Fig. 5.5--Spectrum curves (smoothed) of harmonic amplitudes

The next assumption was that the structural damping of the floor (from all causes),  $\zeta$ , 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.

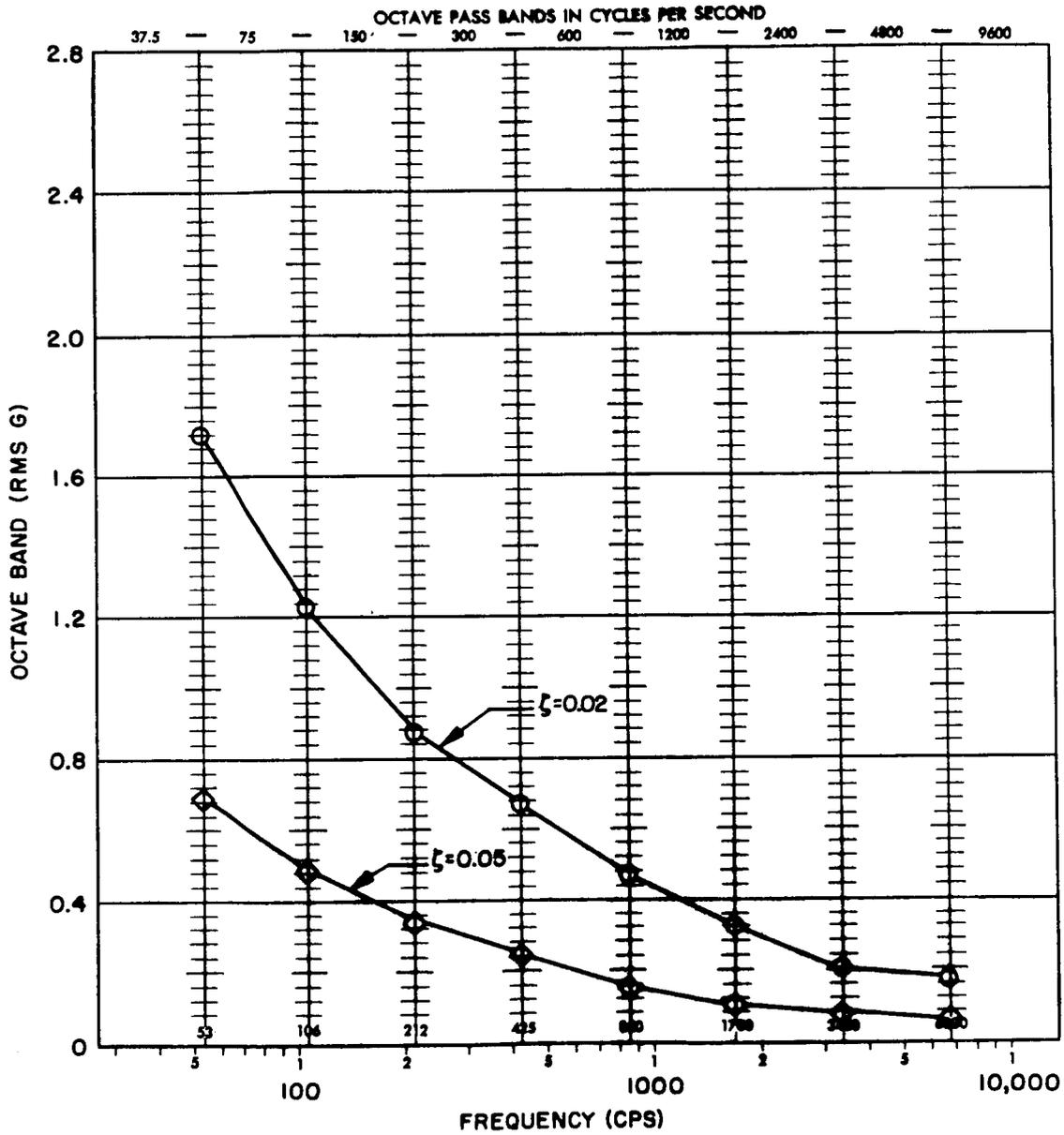


Fig. 5.6--Crew-compartment floor vibration (Case 1)

In estimating the acoustic output provided by the vibrating floor, it was considered that for this very preliminary exercise the most straightforward procedure would be to assume an acoustical-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

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

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

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

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

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

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

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

then  $R = 31.3$  and  $SPL = PWL - 9 \text{ db.}$

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

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.

\* 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.

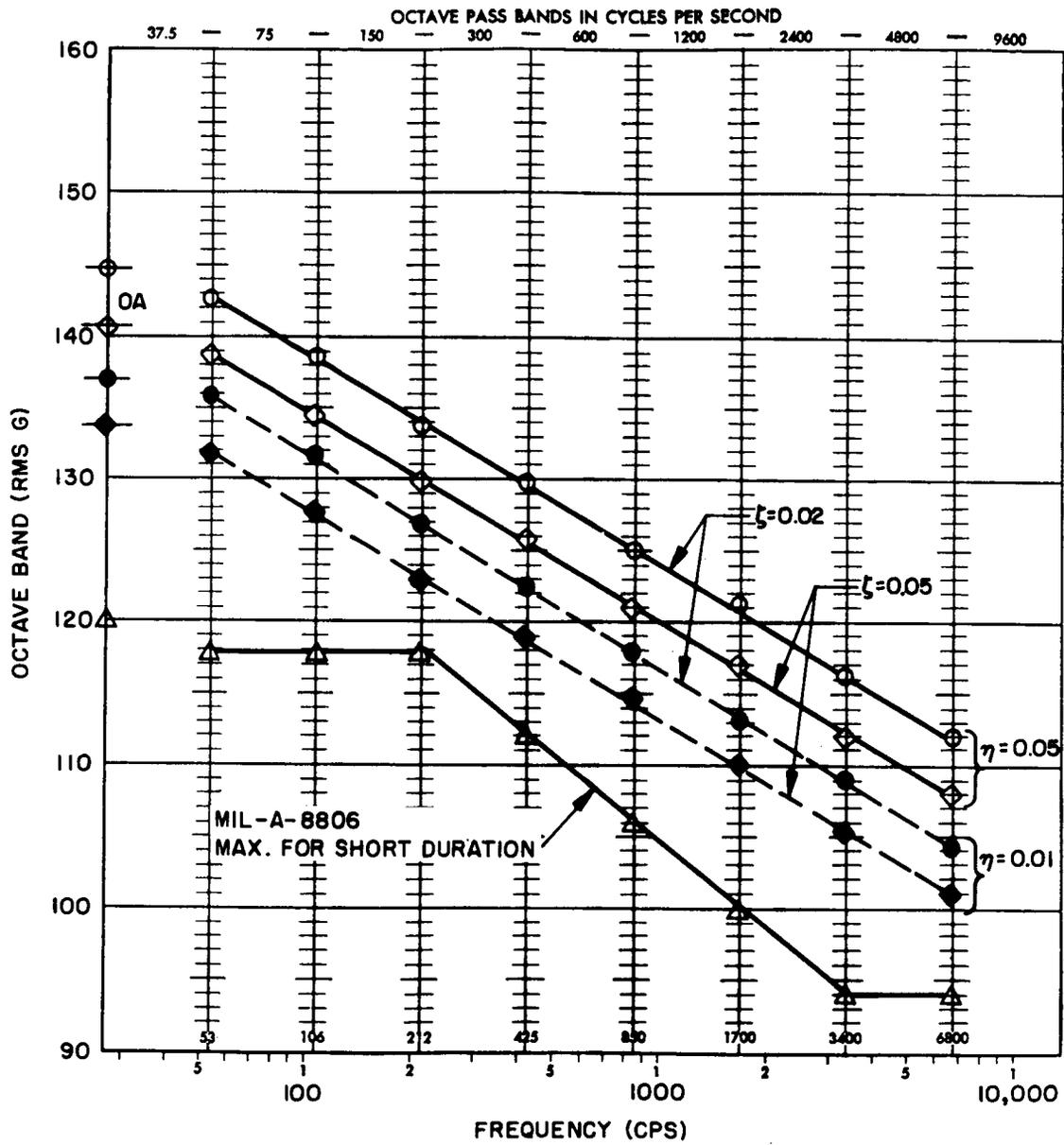


Fig. 5.7--Crew-compartment noise (case 1) at 10-psi pressurization

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.

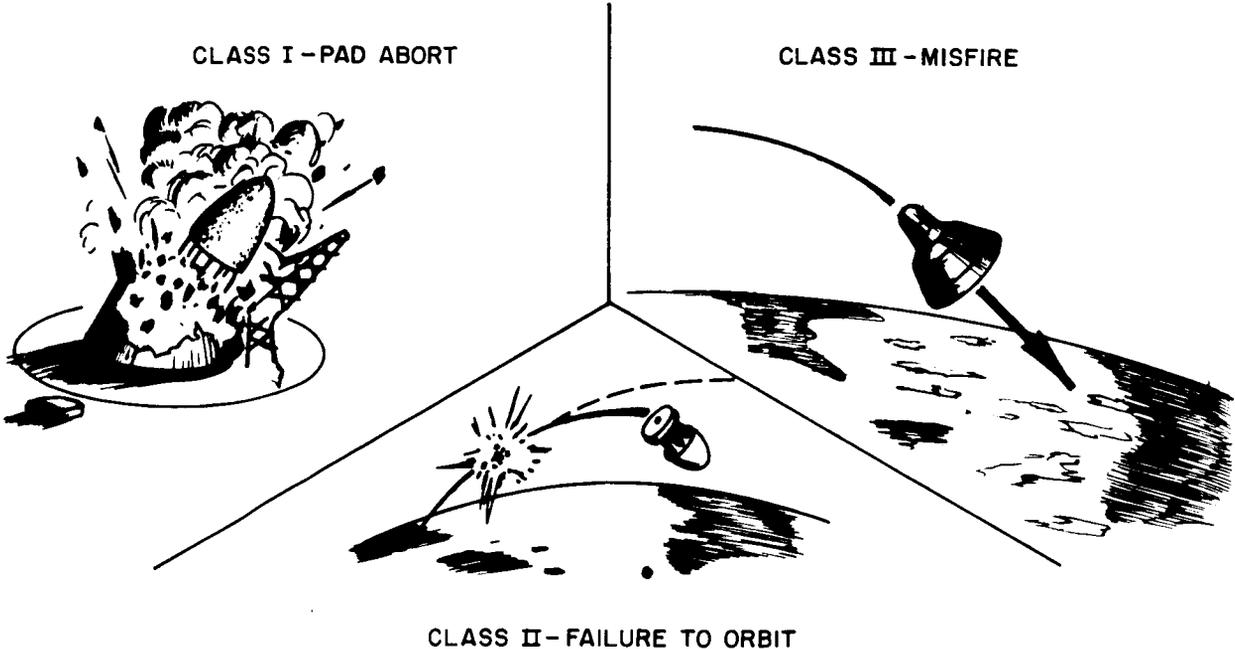


Fig. 5. 8--Operational failure mode classification for ground hazards

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 vaporization of plutonium, which could produce a downwind inhalation hazard and ground-contamination hazard. Tolerances on maximum permissible concentrations 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 combination 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

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

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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

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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  $\sim 2000$  ft/sec  $\Delta V$  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 catastrophe. 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 meteoroid 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  $\Delta V$  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 countermeasure 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  $20 \text{ g/cm}^2$  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  $\Delta V$ ). 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 ( $\leq 200^{\circ}$  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

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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

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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  $44 \times 10^6$  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  $3 \times 10^7$  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  $3 \times 10^7$  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  $2 \times 10^4$  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  $3 \times 10^7$  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  $3 \times 10^7$  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  $3 \times 10^7$  kg/yr)

Available Production Method	Plutonium Production/Year	Estimated Cost/kg*
1. Conversion of 1/5 of U produced, with 55% extraction efficiency	$2 \times 10^4$ 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	$2 \times 10^6$ 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  $40 \times 10^6$  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  $3 \times 10^7$  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

## 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.

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 payload 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 preparation 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.

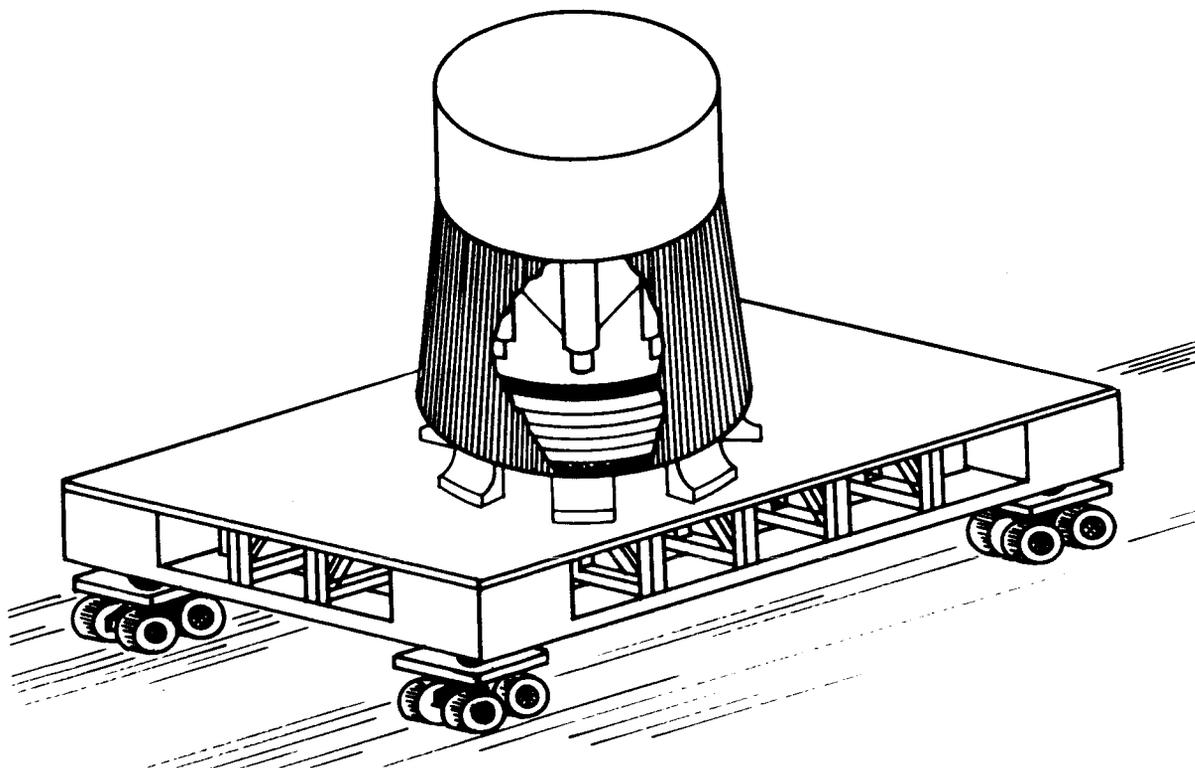


Fig. 6.1-- Conceptual transporter for the nuclear pulse propulsion module

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. Crawler-Transporter. 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. Arming Tower. 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. Transport Barges. 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. Operations

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

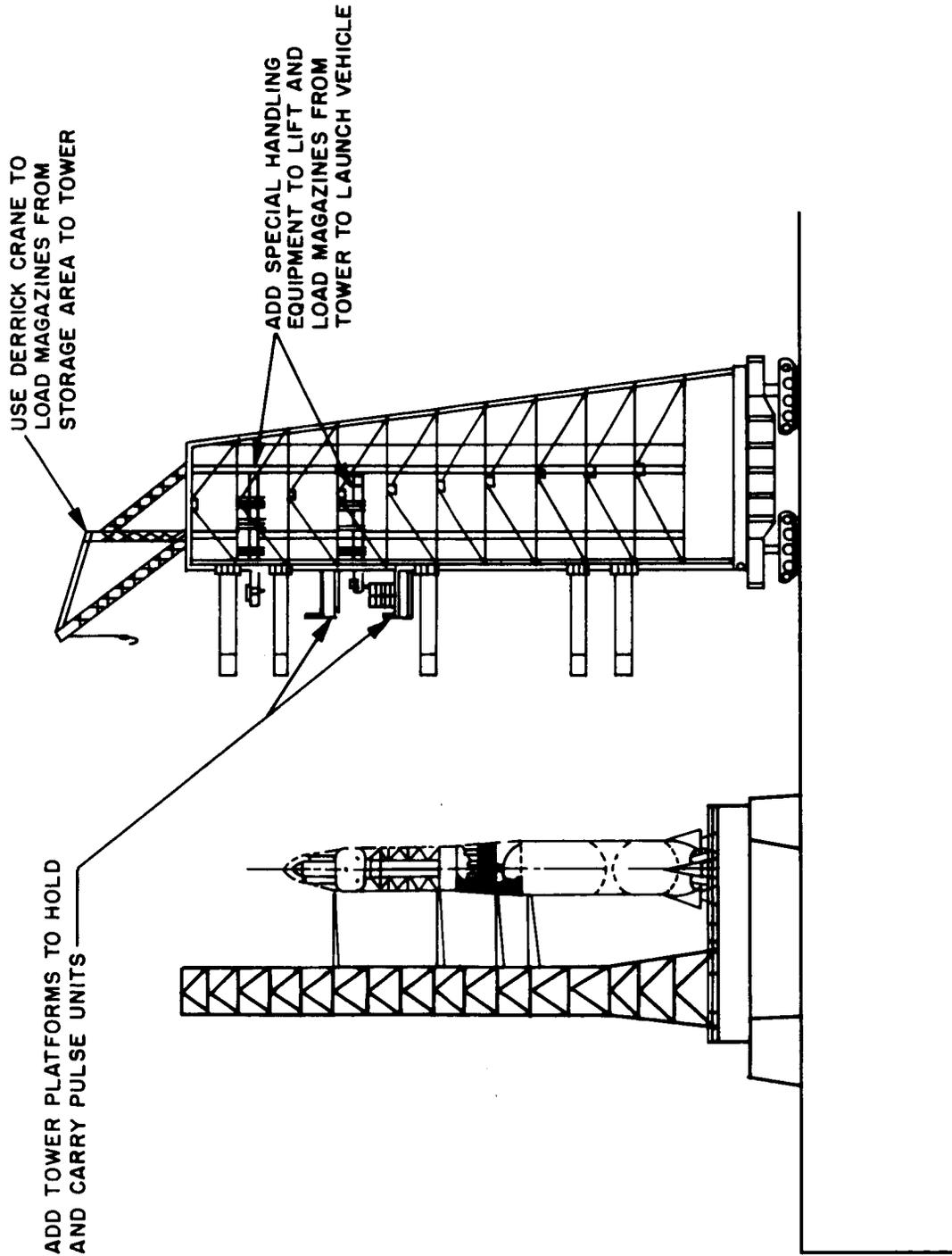


Fig. 6.2--Conceptual arming tower modification for nuclear pulse 10-m vehicle operations

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 loftier, 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<sup>6</sup>-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

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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.

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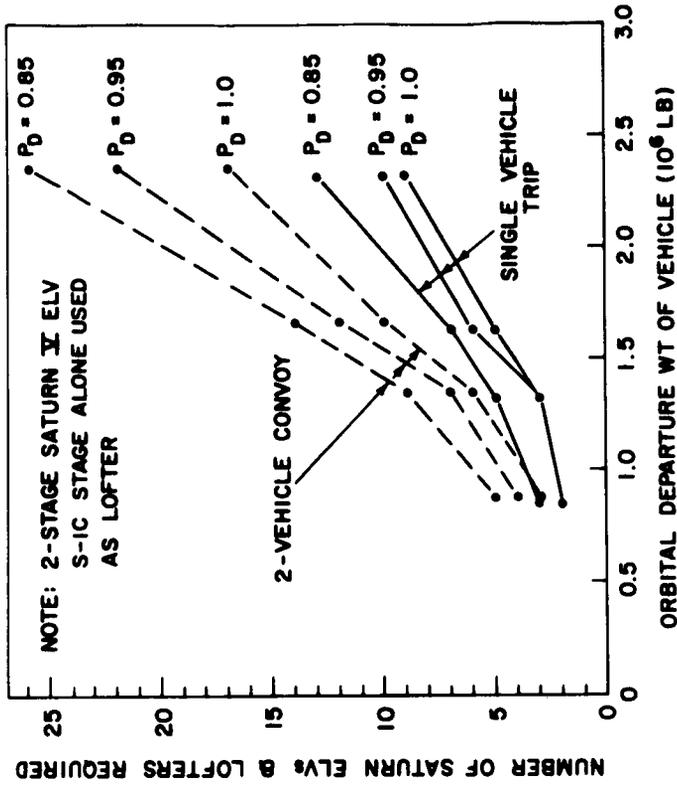


Fig. 6.4--Mode II operation launch-vehicle requirements for selected earth-orbit departure weights (exploration missions)

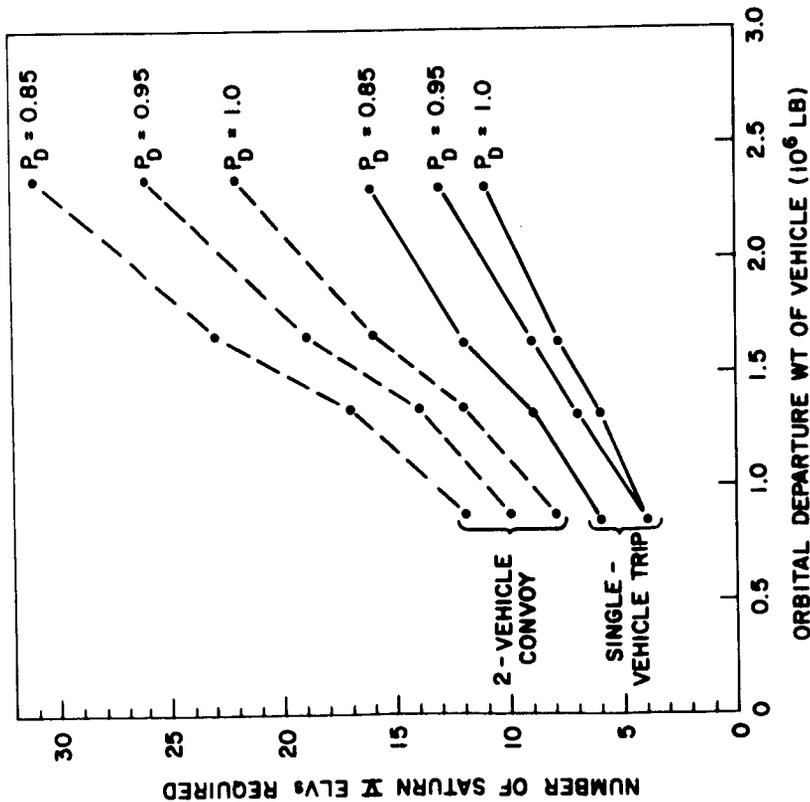


Fig. 6.3--Mode III operation launch-vehicle requirements for selected earth-orbit departure weights (exploration missions)

## 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

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

### 7.1. OBJECTIVE: 10-M VEHICLE

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. DEVELOPMENT APPROACH

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

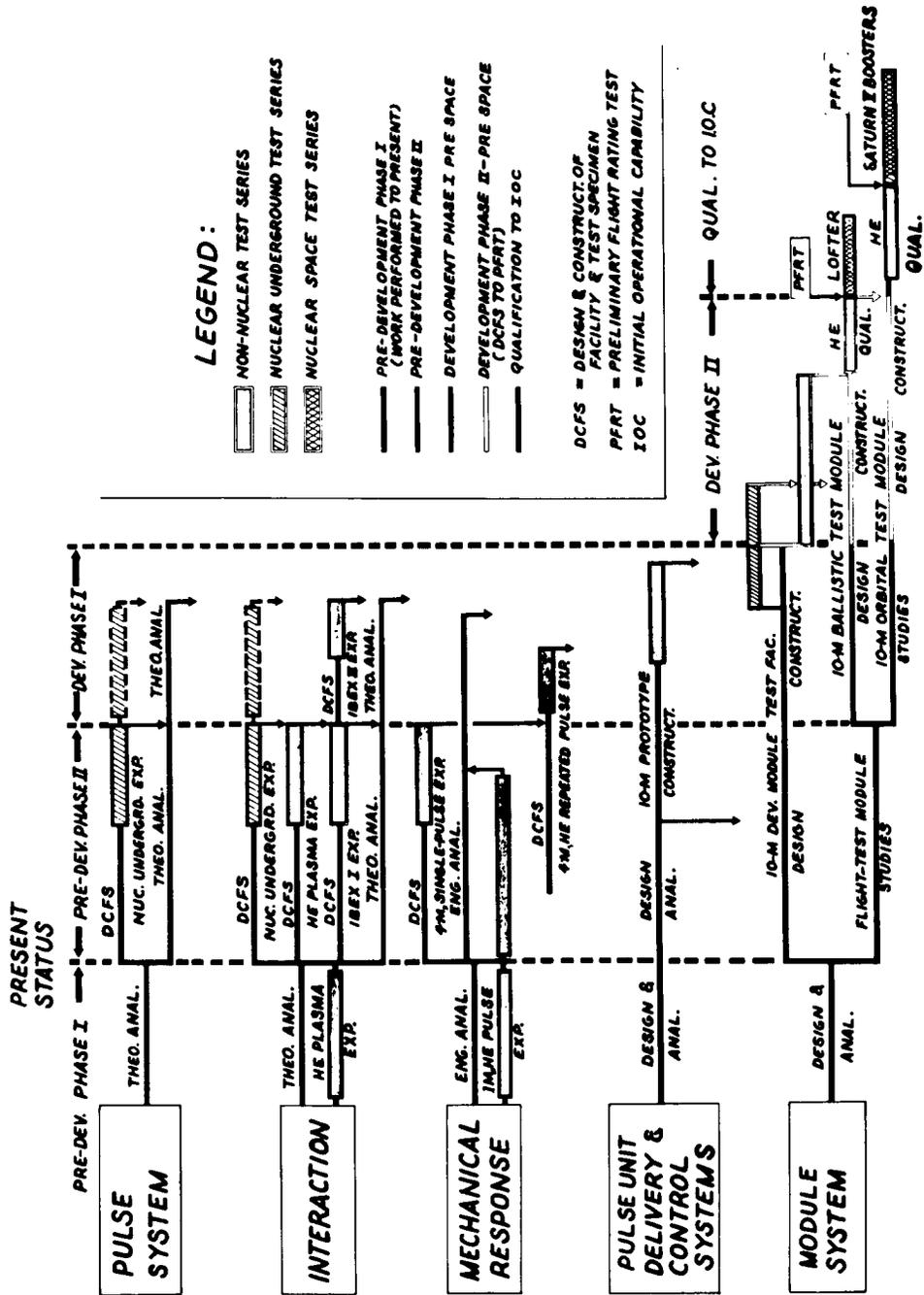


Fig. 7. 1--Development phases

DEVELOPMENT AREAS	EXPERIMENTAL TECHNIQUES									
	HE PLASMA	EM PLASMA	COMP. TEST (VACUUM)	HE PULSE	HE REP. PULSES	NUCLEAR UNDERGROUND		NUCLEAR REP. BALLISTIC	NUCLEAR REP. ORBIT	
						PUL. UNIT	SCALED MODULE			
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 4a. REP. S.A. & ATTACH. RESPONSE 4b. S.A. COOLING			S.A. ONLY S.A. ONLY	● ●	● ● ●		●	● ●		
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

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.

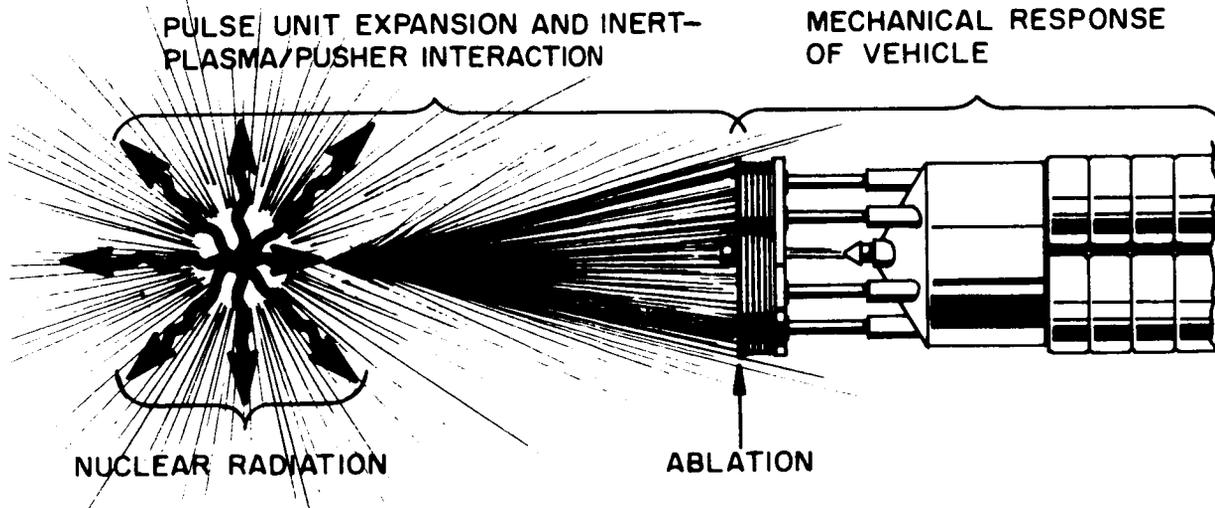


Fig. 7.3--Separability of nuclear-pulse problems

### 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 convective instability) that may not be amenable to calculation.

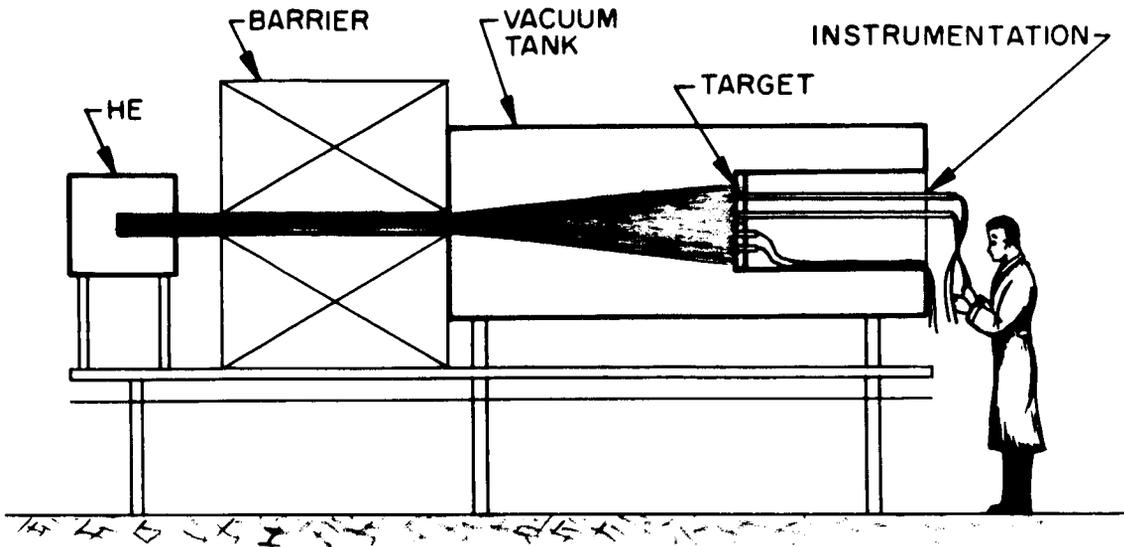


Fig. 7.4--HE plasma generator

### 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.

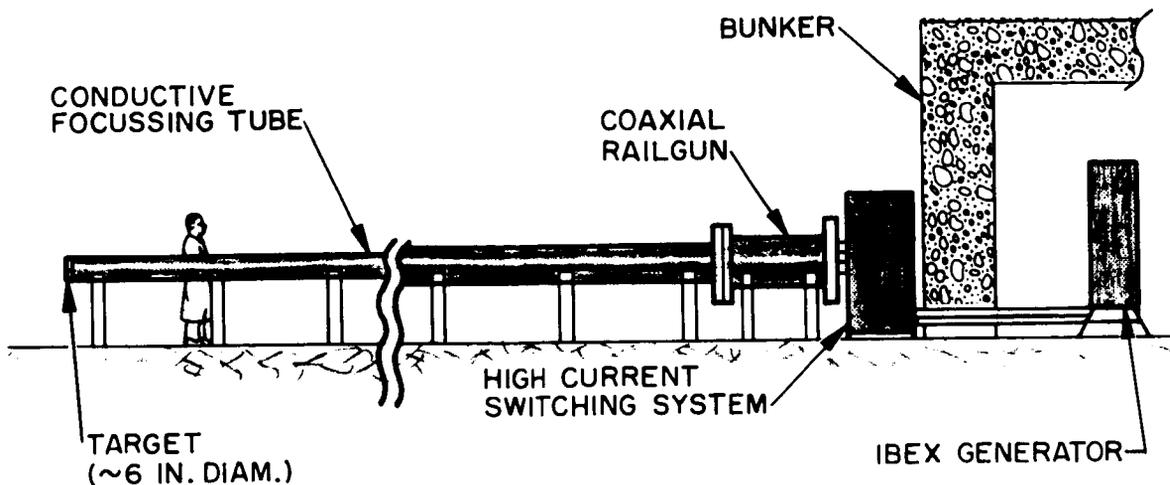


Fig. 7.5--Pulsed high-energy electromagnetic plasma accelerator

### 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 prototype. 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 laboratory 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 intercepted 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  $\mu$ sec, during which time the propellant temperature rises to  $\sim 100,000^{\circ}$ 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  $600^{\circ}$ 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. Tests at Existing Facilities. 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. Four-meter Test Stand. 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. Ten-meter Test Stand. 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.

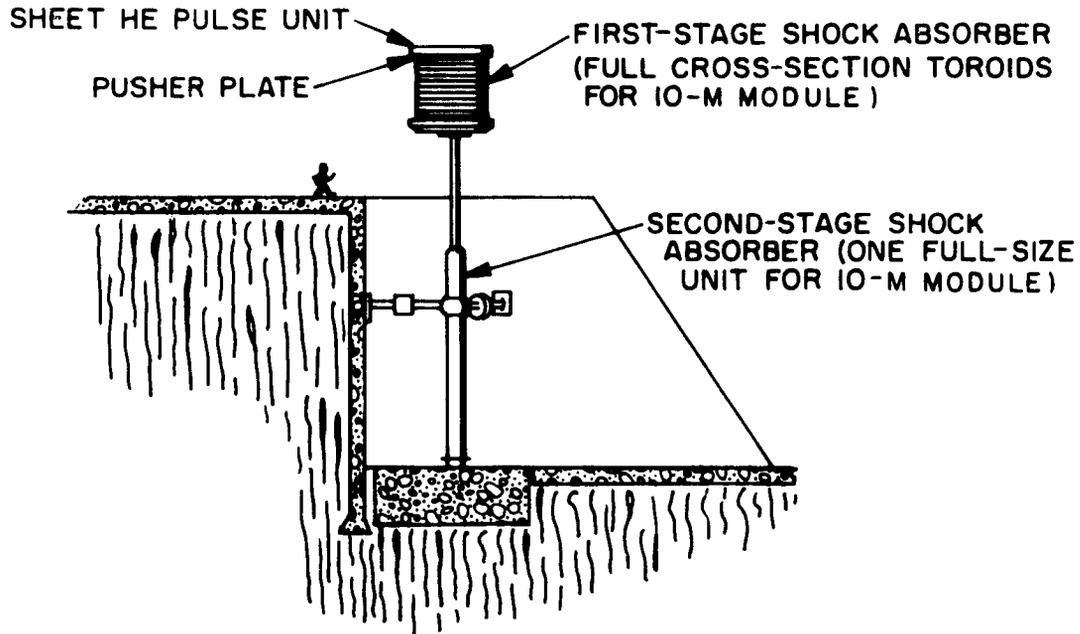


Fig. 7.6--4-m-diam single-pulse HE test facility

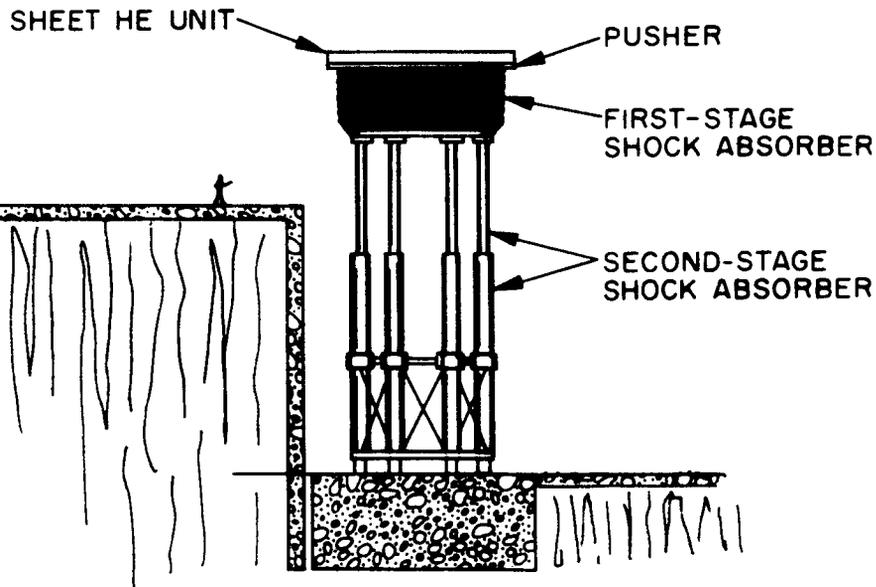


Fig. 7.7--10-m-diam single-pulse HE test facility

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)

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.

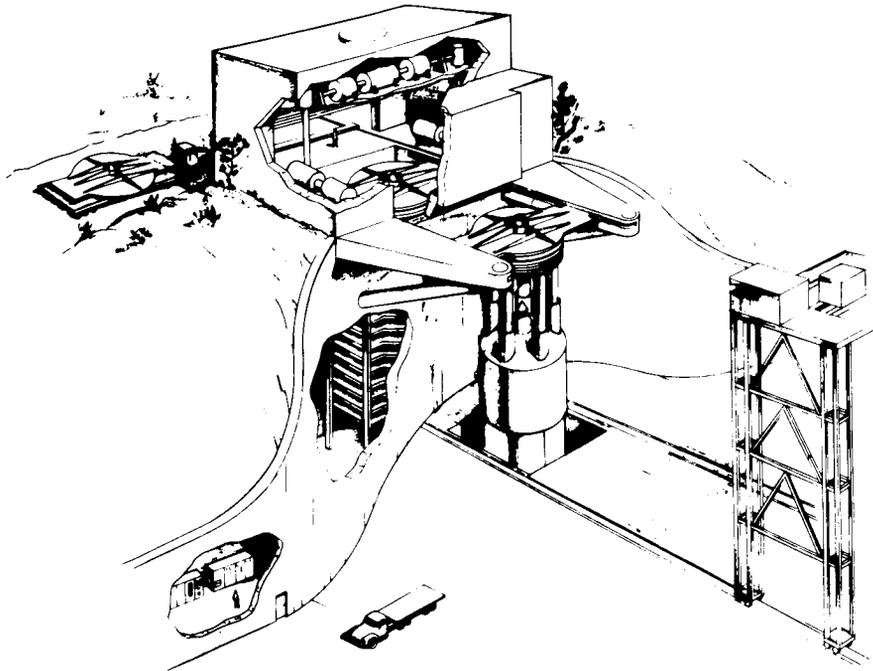


Fig. 7.8--Repetitive-HE-pulse test facility with inverted 10-m module in test position

### 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.

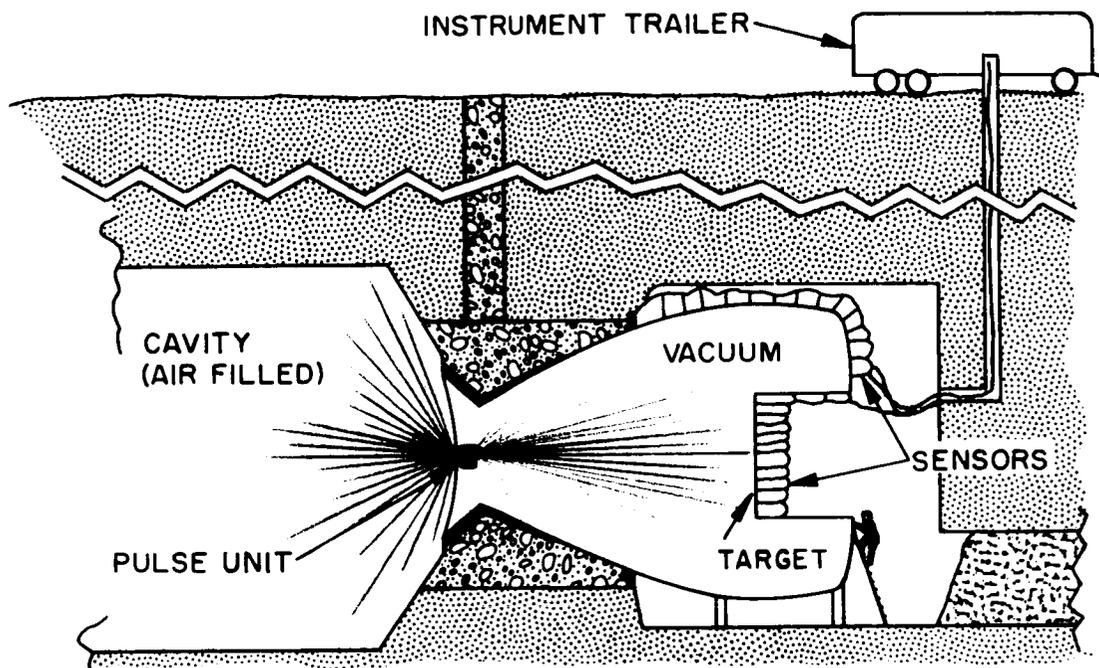


Fig. 7.9--Nuclear underground tests for pulse-unit development

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 interaction 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.

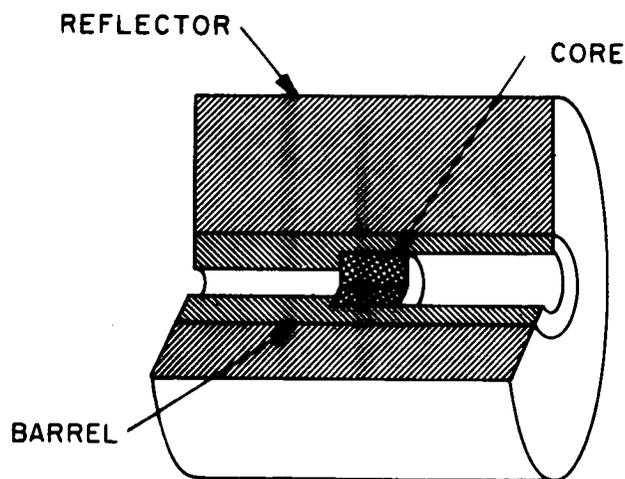


Fig. 7.10--Low-energy (yield) nuclear source (LENS) for nuclear underground tests

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.

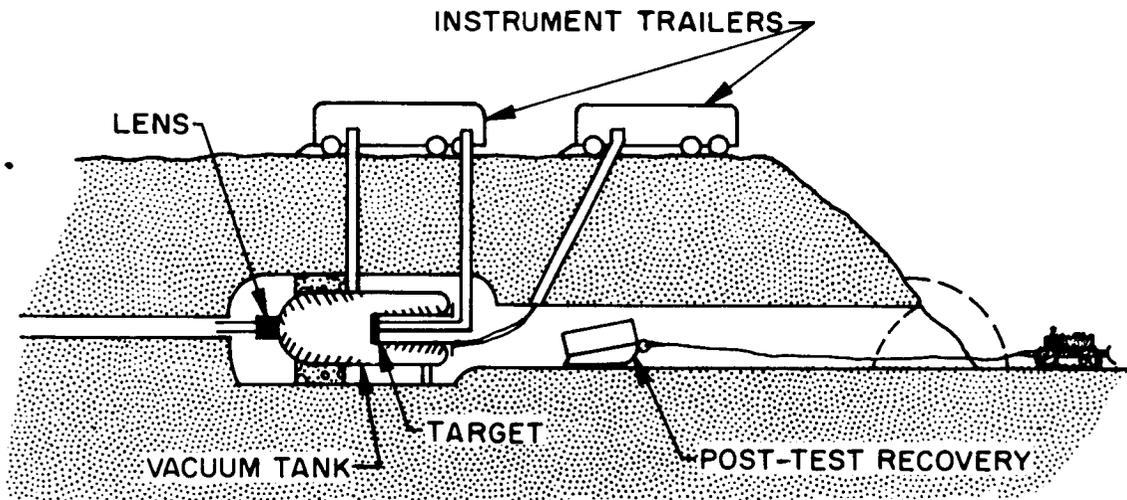


Fig. 7.11--Test facility for nuclear-underground-tests interaction studies

7.3.6.3. Full-scale Module Interaction. 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.

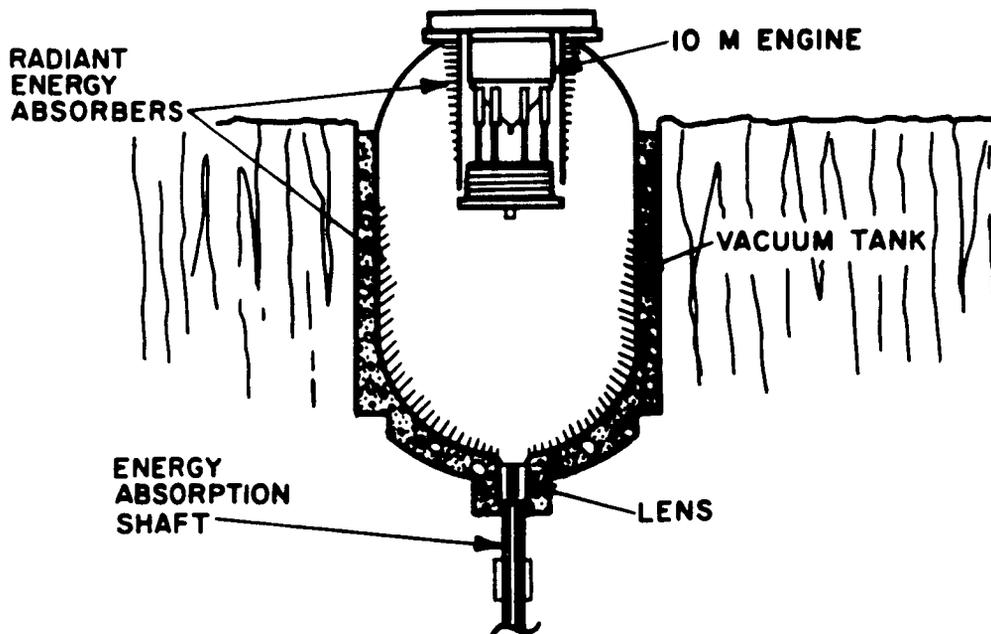


Fig. 7. 12--Nuclear underground tests full-scale interaction

### 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, subsystems, 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 measurements much like those made for the ballistic tests will be made to

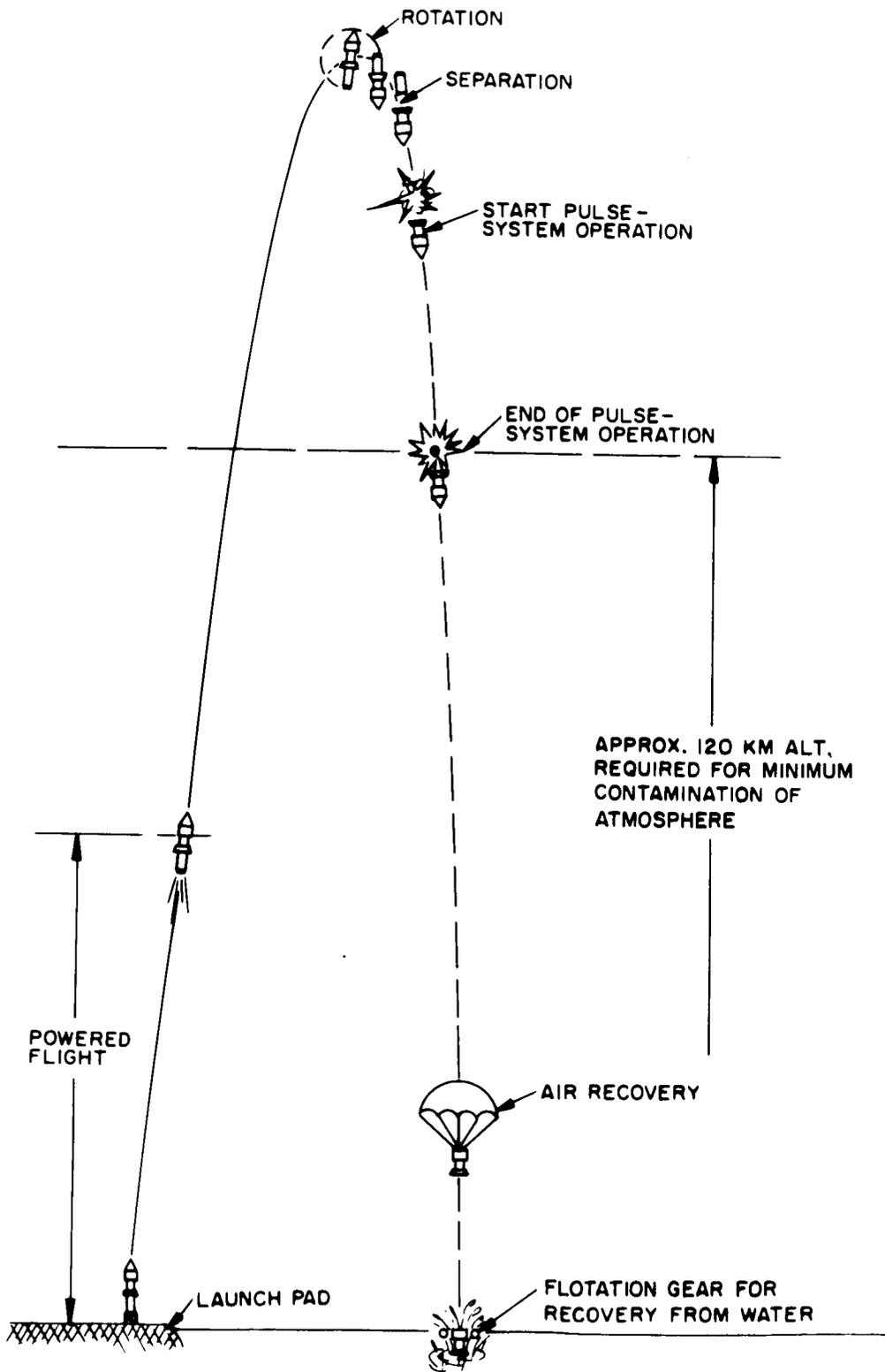


Fig. 7. 13--Trajectory for nuclear ballistic tests

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).

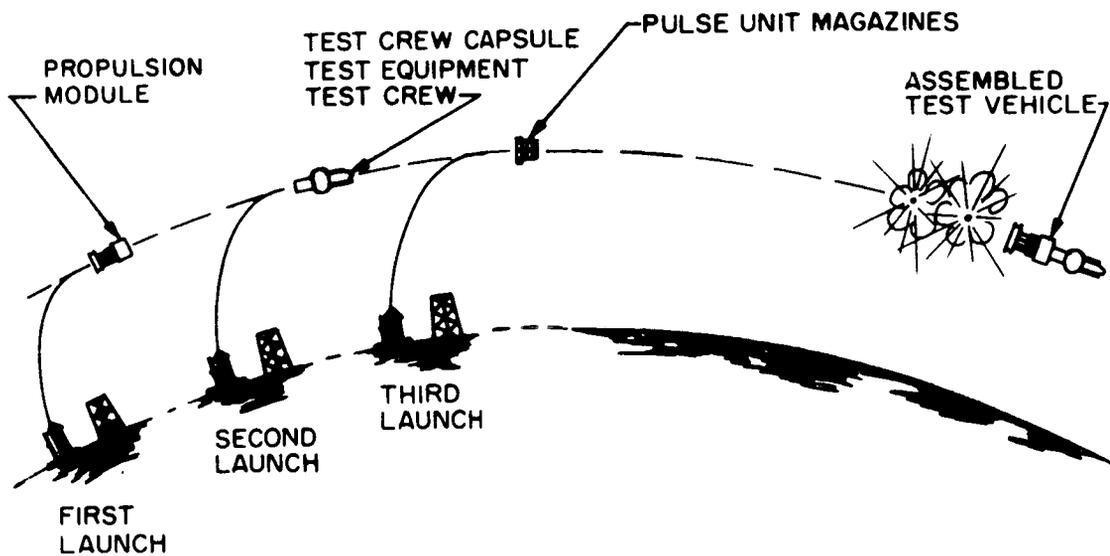


Fig. 7.14--Orbital tests

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

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. Therefore, 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.

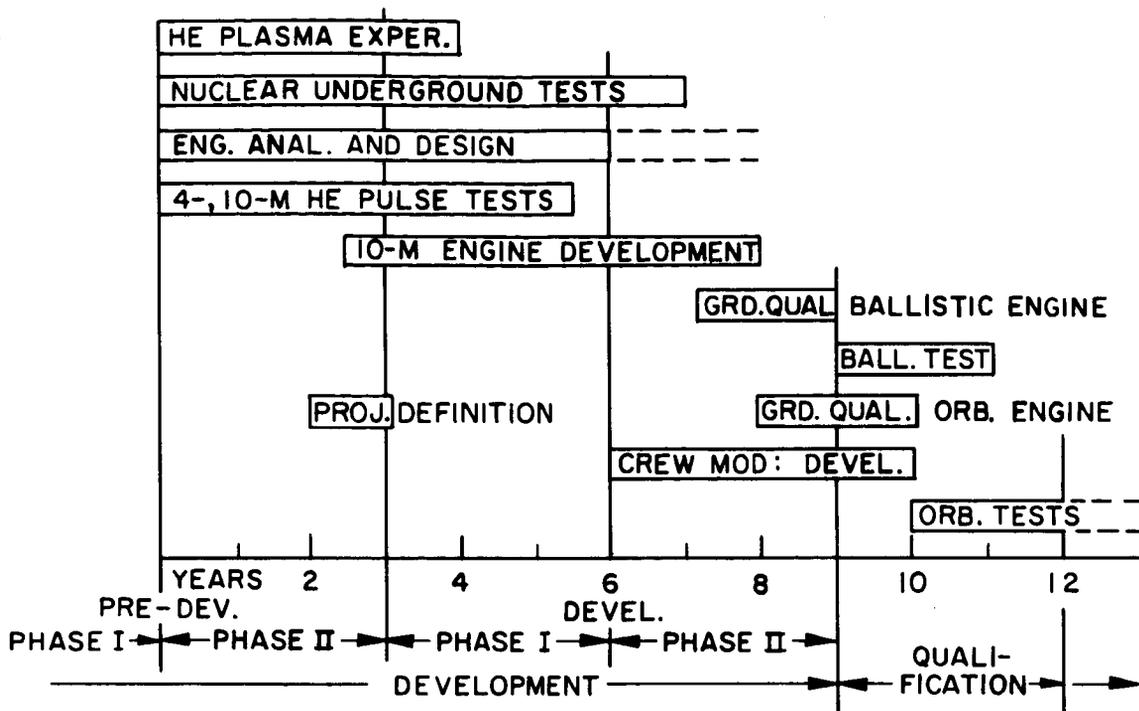


Fig. 7.15--Condensed development schedule

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).



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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 components, 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

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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 Development 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

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:

	<u>\$ Millions</u>
Predevelopment Phase II . . . . .	63
Development Phase I . . . . .	152
Development Phase II . . . . .	239
Qualification . . . . .	<u>1,037</u>
Total . . . . .	1,491

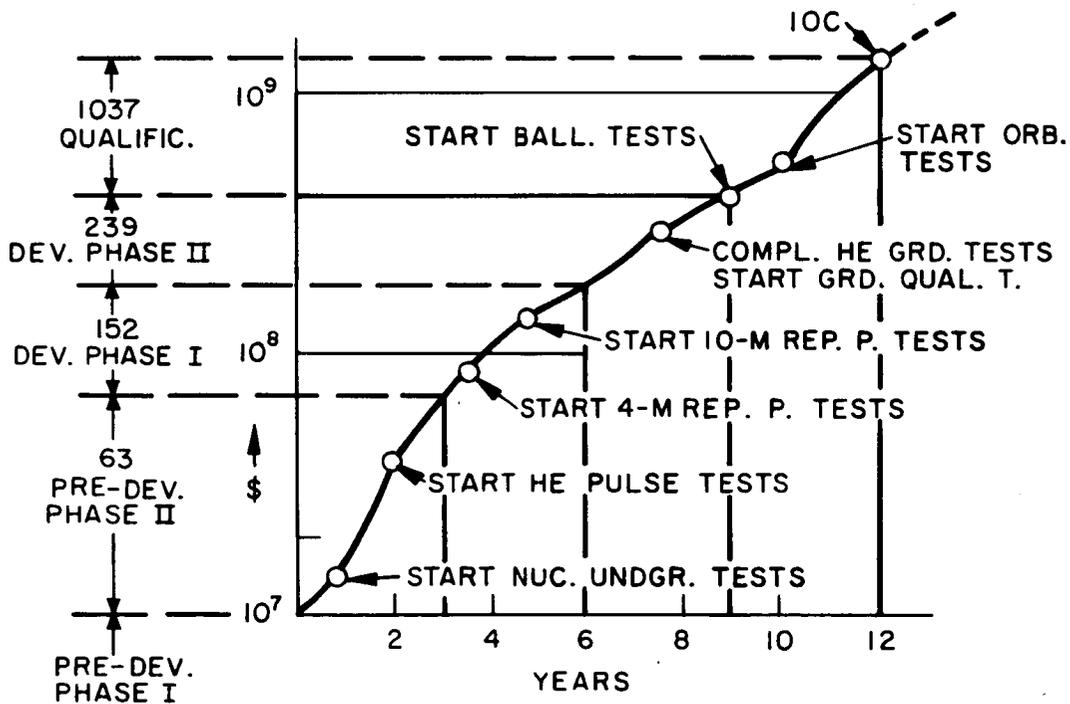


Fig. 7.17--Cumulative cost summary

## 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 accounts 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.

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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  $\$16 \times 10^6$ , whereas the actual expected cost is about  $\$30 \times 10^6$ .) One reason for the lower cost is that a large portion of the nuclear-pulse module's mass is concentrated in a few large components, 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.

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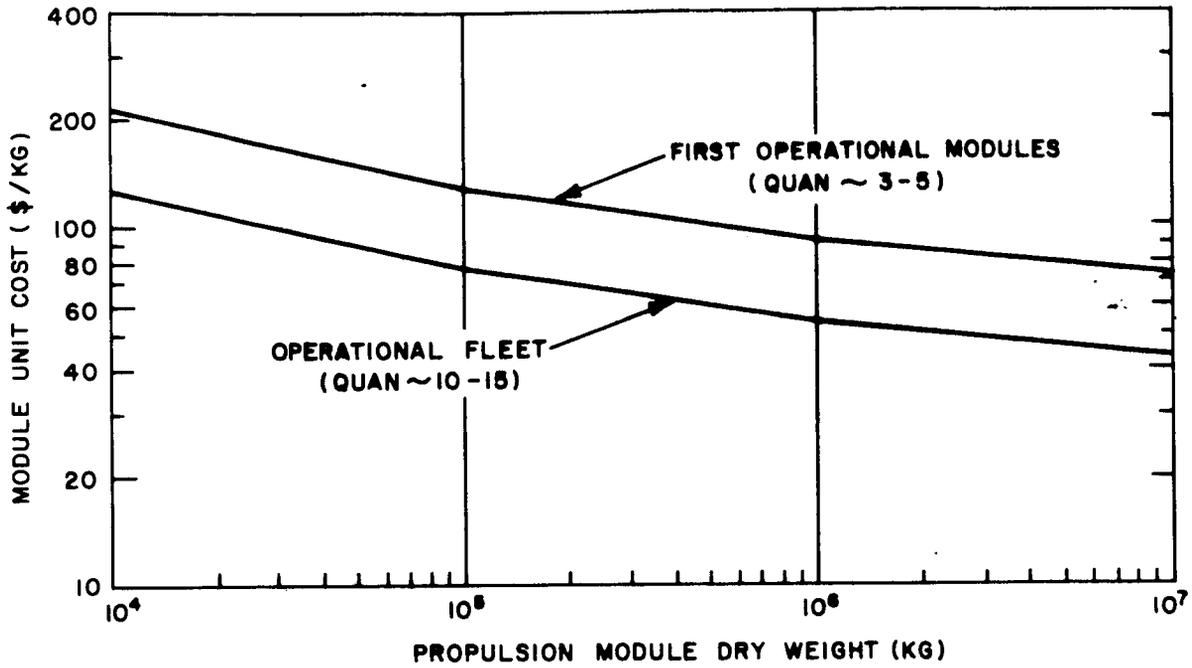


Fig. 8.1 -- Estimated fabrication cost per kilogram for nuclear-pulse-propulsion modules

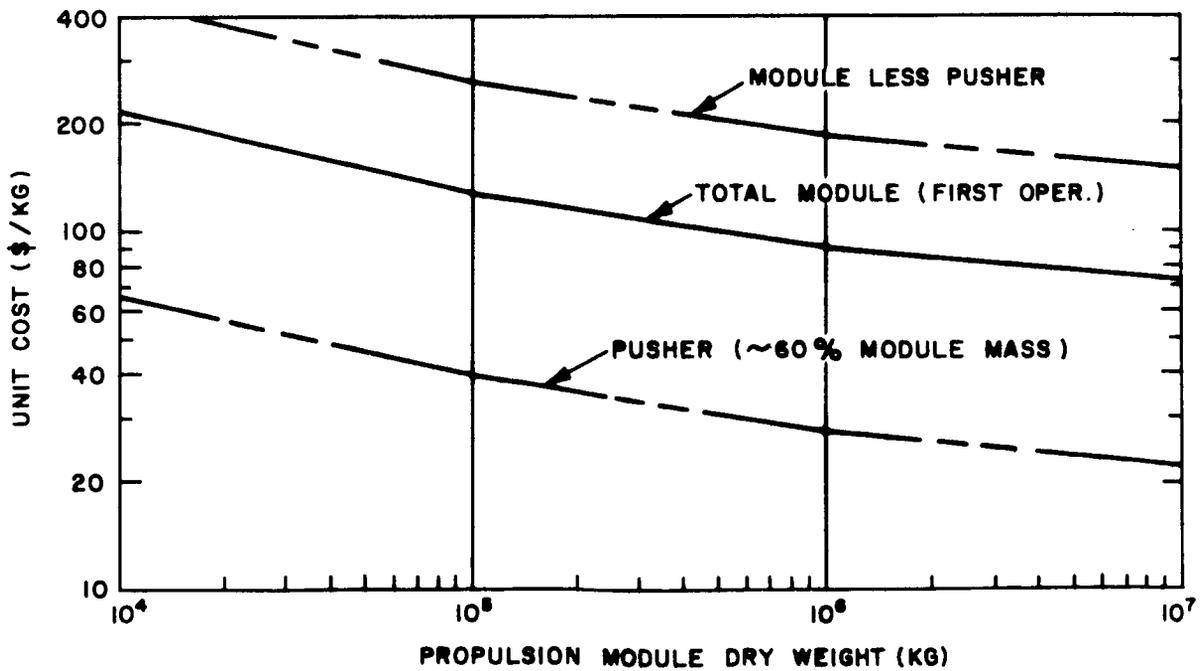


Fig. 8.2 -- Effect of pusher cost on estimated cost per kilogram of complete propulsion module

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	$\$150/\text{kg} \times 0.26 = 39$	
~50% main and emergency power supply	$\$400/\text{kg} \times 0.50 = 200$	
~ 5% communications, navigation, autopilot, and guidance systems	$\$2,000/\text{kg} \times 0.05 = 100$	
~19% control, spin system and abort chemical motors, tanks, and plumbing	$\$300/\text{kg} \times 0.19 = \frac{57}{396} \approx$	
100%		\$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: $101/132 \times \$250/\text{kg}$ ) . . . . .		\$191/kg
Equipment for 20-m vehicles (same basis as for structure: $101/132 \times \$400/\text{kg}$ ) . . . . .		\$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
<u>In-transit Payload Allowance</u>		
(In-transit payload was assumed a mix of scientific/electronic equipment and "guinea pig" experiments; modest total weight) . . . . .		\$1,000/kg
<u>Destination Payload Allowance</u>		
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	$\$2,000/\text{kg} \times 0.013 = 26$	
~ 0.7% data-handling and storage	$\$2,000/\text{kg} \times 0.007 = 14$	
~ 5.6% environmental satellites	$\$5,000/\text{kg} \times 0.56 = 280$	
~ 8.5% unmanned landers and propellant	$\$500/\text{kg} \times 0.085 = 42$	
~16.5% unmanned returners and propellant	$\$300/\text{kg} \times 0.965 = 50$	
~67.4% manned excursion vehicle and propellant	$\$500/\text{kg} \times 0.674 = \frac{337}{749} \approx$	
100.0%		\$750/kg
Very large destination payloads ( $\geq 250,000$ kg; a mix similar to the above assumed, but with larger manned excursion vehicles and more propellant; estimated 2/3 the above) . . . . .		\$500/kg

### 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  $28 \times 10^6$  N, or  $6 \times 10^6$  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

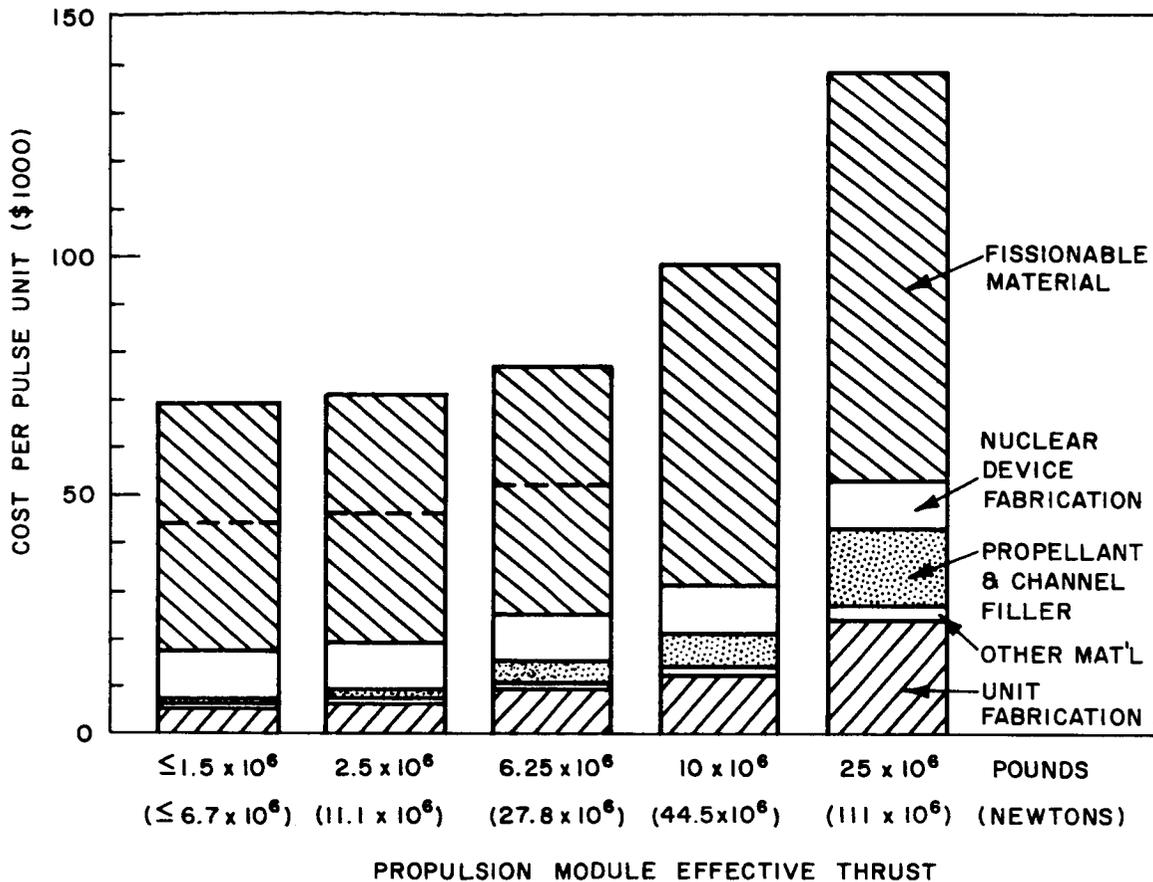


Fig. 8.3--Nuclear-pulse-propellant cost breakdown per pulse for a range of propulsion module thrusts

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

\* 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).

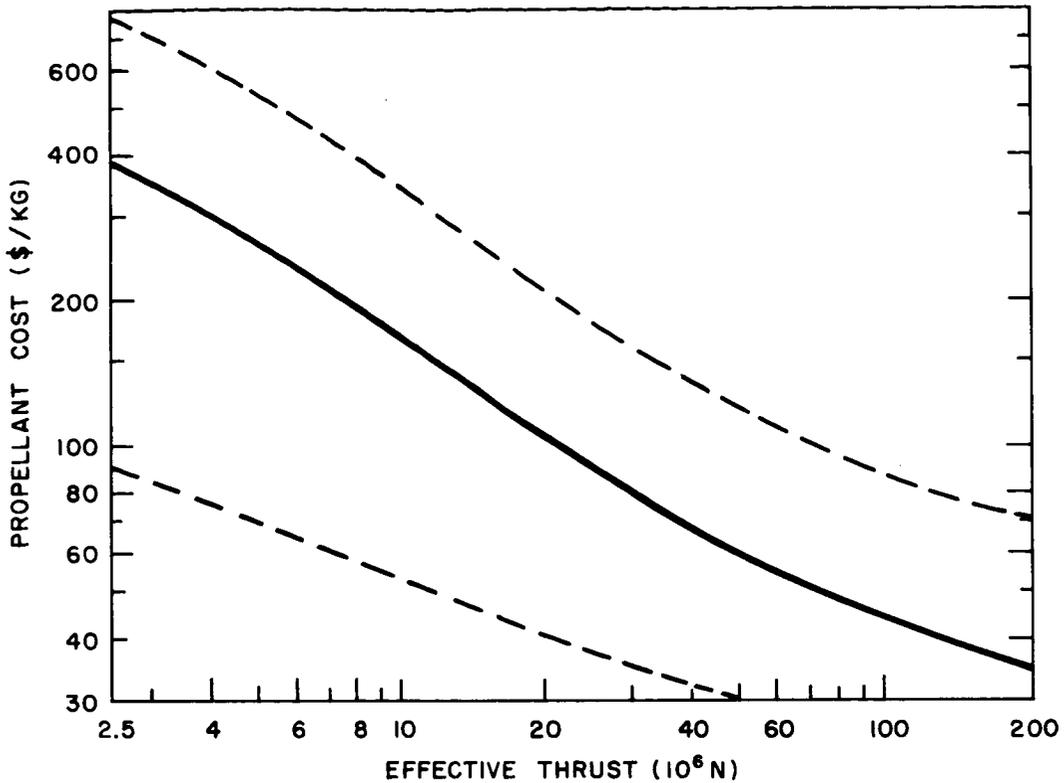


Fig. 8. 4--Nuclear-pulse-propellant estimated over-all cost per kilogram as a function of effective thrust

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 <sup>6</sup>
S-II stage, including engines . . . . .	\$16×10 <sup>6</sup>
S-4B stage, including engines . . . . .	\$ 8×10 <sup>6</sup>
Instrument unit and guidance . . . . .	\$ 6×10 <sup>6</sup>
System integration . . . . .	\$10×10 <sup>6</sup>
Launch operations . . . . .	\$ 4×10 <sup>6</sup>
Total	<u>\$74×10<sup>6</sup></u>