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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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FOREWORD

The technical report on the Nuclear Pulse Space Vehicle Study, performed under National Aeronautics and Space Administration Contract NA58-11053, consists of four volumes:

Vol. I	Summary Report (Secret);
Vol. II	Vehicle Systems Performance and Costs (Secret);
Vol. III	Conceptual Vehicle Designs and Operational Systems (Secret--Restricted Data);
Vol. IV	Mission Velocity Requirements and System Comparisons (Unclassified), prepared by General Dynamics/Astronautics.

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

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

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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 ~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

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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 [redacted]).

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

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 ~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

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.

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 ΔV earth-departure maneuvers, since gravity losses are then insignificant. When lower departure thrust-to-weight ratios were used or the earth-departure maneuver ΔV was high, additional gravity loss penalities (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.

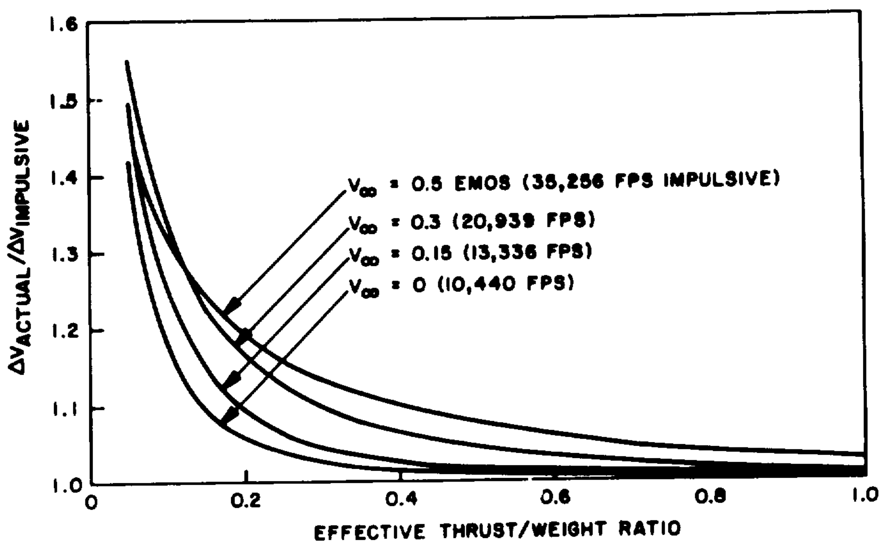

Fig. 2.1--Earth-departure gravity losses vs thrust-to-weight ratio (I_{sp} = 1,850 sec) for 10-m propulsion module

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

Maneuver	ΔV Requirements (m/sec)	(ft/sec)
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	3,234	10,610
Total Velocity Required	9,031	29,634

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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

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

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

Maneuver	ΔV Requirement (m/sec)	ΔV Requirement (ft/sec)
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

	Earth-approach Condition							
	M-1, Hyperbolic Rendezvous		M-2, 50,000-fps Reentry		M-3, 36,300-fps Reentry		M-4, Circular-orbit Capture	
Maneuver	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

		Mission Velocity Requirement					
	Departure	Outbound		Return		Total	
Year	Date	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 m/sec	300-day Mission (ft/sec)	150-day Mission m/sec	150-day Mission (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	4,030	13,200	10,700	35,000
Total velocity requirement	31,820	103,300	63,550	208,800
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	ΔV Requirement (m/sec)	ΔV Requirement (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	16,320	53,600
Total velocity required	63,740	209,000
Reference departure date	2/20/82	
Outbound transfer time, days	400	
Capture and window times, days	30, 20	
Return transfer time, days	460	
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_1, (EMPIRE[a] 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
Operational payload:						
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	41,847	40,167		42,230	75,170	
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	
Spares 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	27,838	26,517		32,280	51,260	
Contingency (~5 percent)	3,315	3,316		3,490	6,570	
Total operational payload[b]	73,000	70,000	73,000	78,000	133,000	133,000
In-transit payload allowance	250	250	1,000	1,000	250	2,000
Destination payload:						
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.)	---		4,000			8,000
Unmanned landers (3 nom.)	---		6,060			12,120
Unmanned returners (3 nom.)	---		11,800			23,600
Manned excursion vehicles (3 nom.)	---		48,000			96,000
Contingency (~5 percent)	50		3,740			7,480
Total destination payload	750	750	75,000	75,000	750	150,000
TOTAL MISSION PAYLOAD[b]	74,000	71,000	149,000	154,000	134,000	285,000
Total mission payload, lb	163,000	157,000	328,000	339,000	296,000	628,000
Payload reduction during mission:						
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	15,300	88,800	92,000	28,900	176,000
Earth approach	18,100	21,000	91,900	96,400	35,100	182,000

[a] 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 63-0001, January 31, 1963).

[b] If drag reentry vehicle and its maneuver stage is required for ~36,300 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

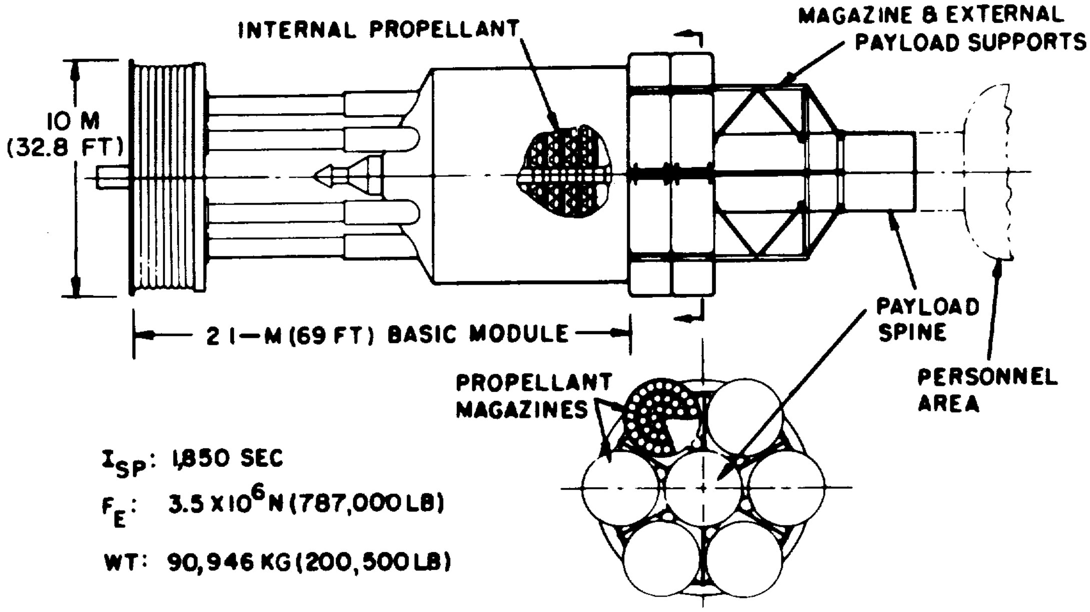

Fig. 3.1--Study configuration of the 10-m nuclear pulse propulsion module

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° 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

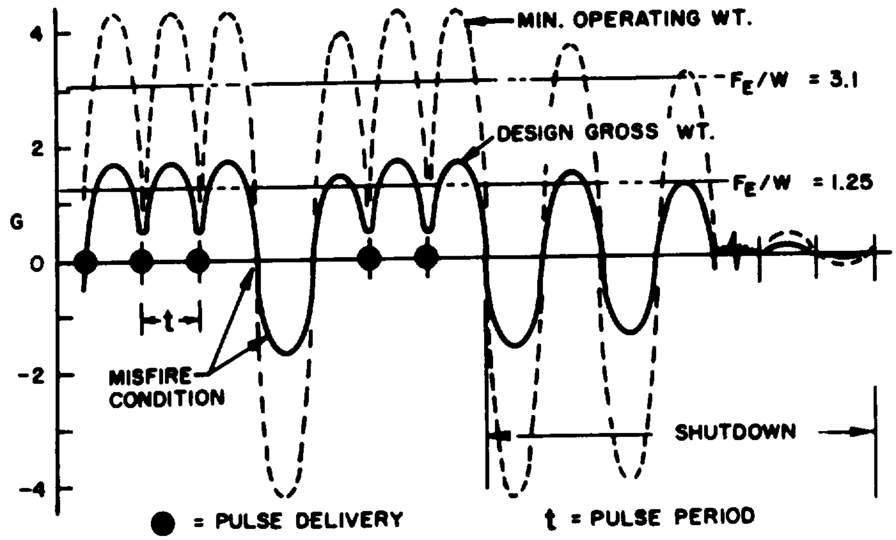

Fig. 3.2--Typical nuclear-pulse-vehicle acceleration profile

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 = 3.5×10^6N, or 787,000 lb) and 20-m module (I_{sp} = 3,150 sec, thrust = 16×10^6N, 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 44×10^6 newtons) \$64/kg. The reason is that

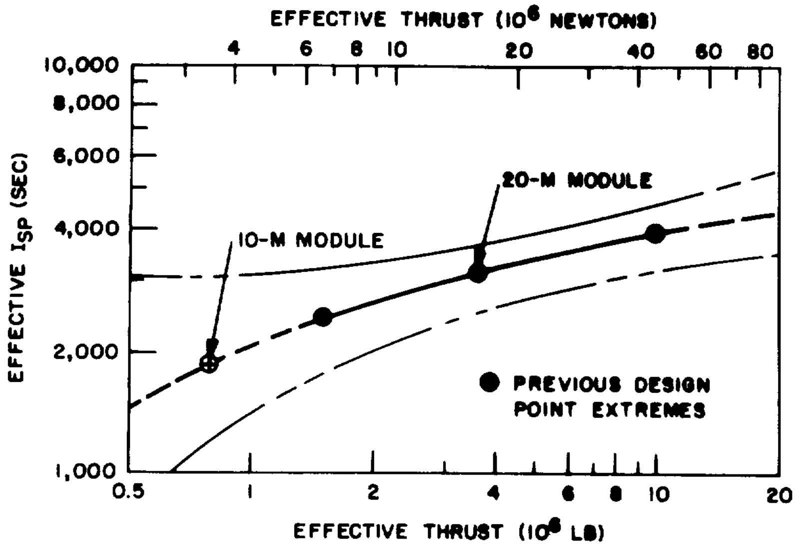

Fig. 3.3--Propulsion-module specific impulse

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 28×10^6 newtons (6×10^6 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 g/cm^2 of hydrogenous material plus some lead for direct radiation shielding (crew compartment bottom) and ~25 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 g/cm^2) is sufficient to attenuate probable solar-flare radiation to values similar to the 50 rem allowed for propulsion.

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 ~2,000 ft/sec ΔV) 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 m^3 (7,100 ft^3) or approximately 25 m^3 (880 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 ~1,000 ft/sec ΔV, is carried as shown for missions requiring its use at earth return.

A 20-m configuration, as arranged and loaded for a high-velocity (~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 m^3 (1,040 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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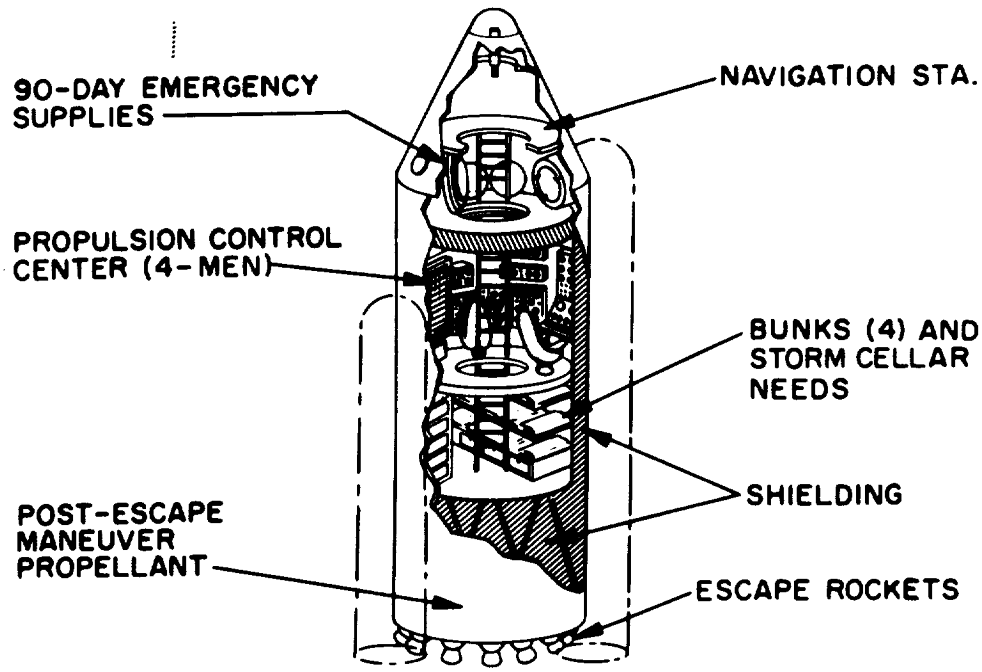

Fig. 3.4--Powered flight station—escape vehicle for 8-man exploration missions with 10-m configurations

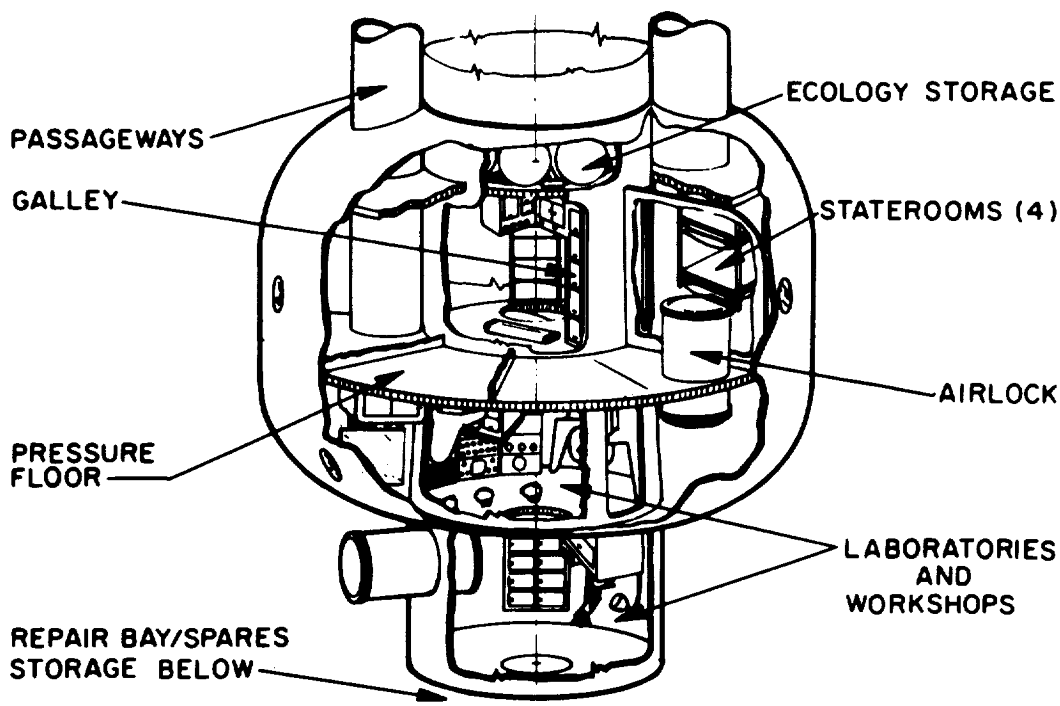

Fig. 3.5--Exploration-mission personnel accommodations for an 8-man complement

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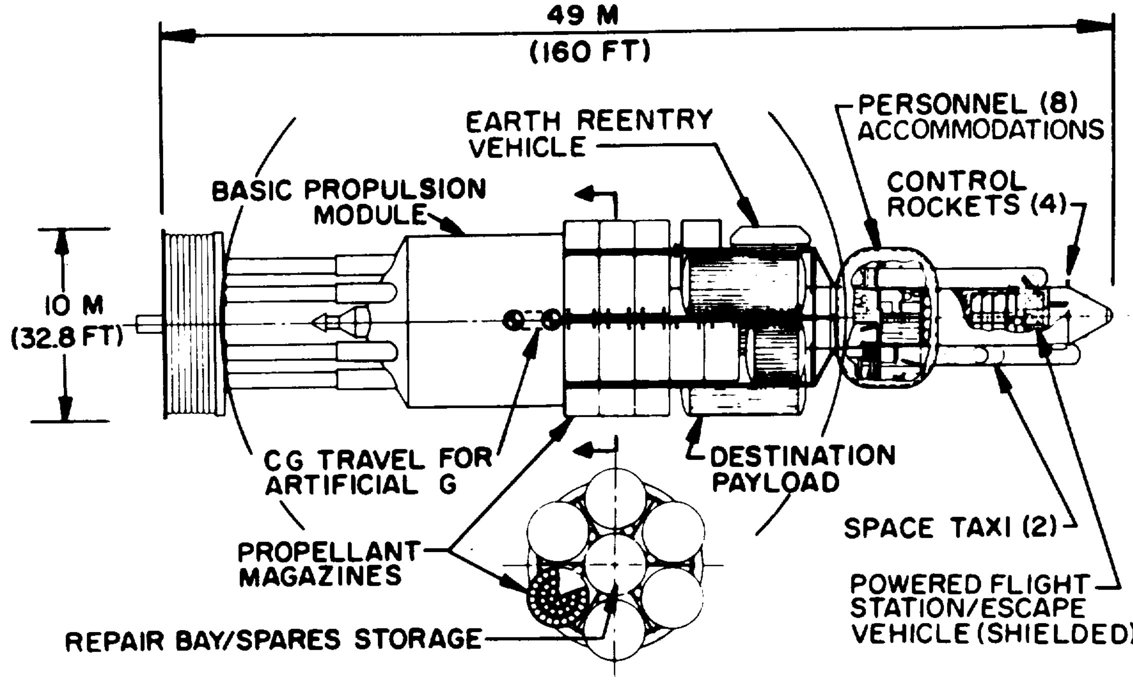


Fig. 3.6--Basic 10-m exploration configuration

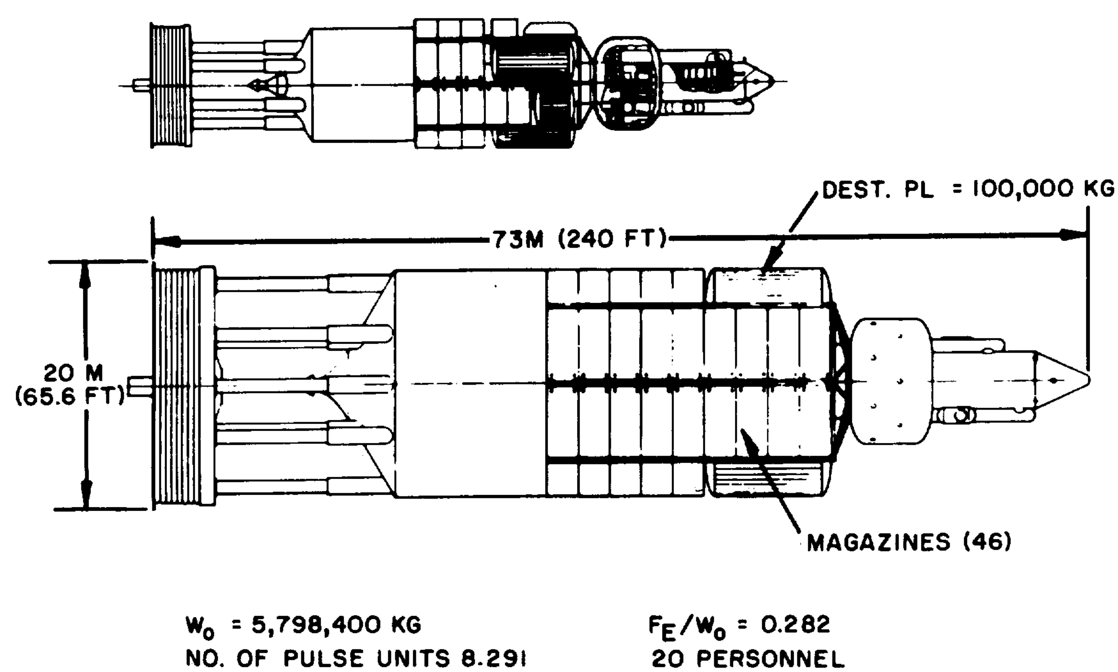


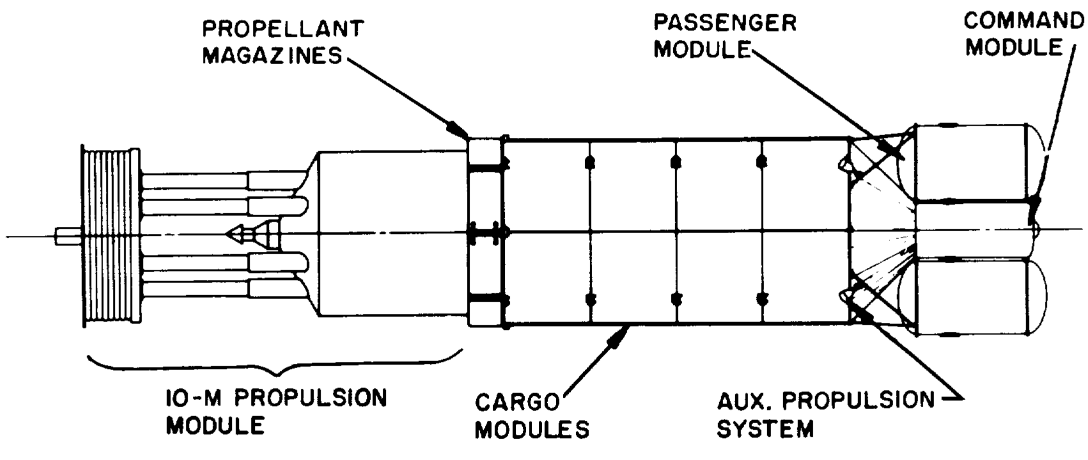
Fig. 3.7--The 20-m exploration configuration, with the 10-m Mars exploration system shown for comparison

[redacted]

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 kg/m^3 (20 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.



$F_E = 3.5 \times 10^6$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 (O_2/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.

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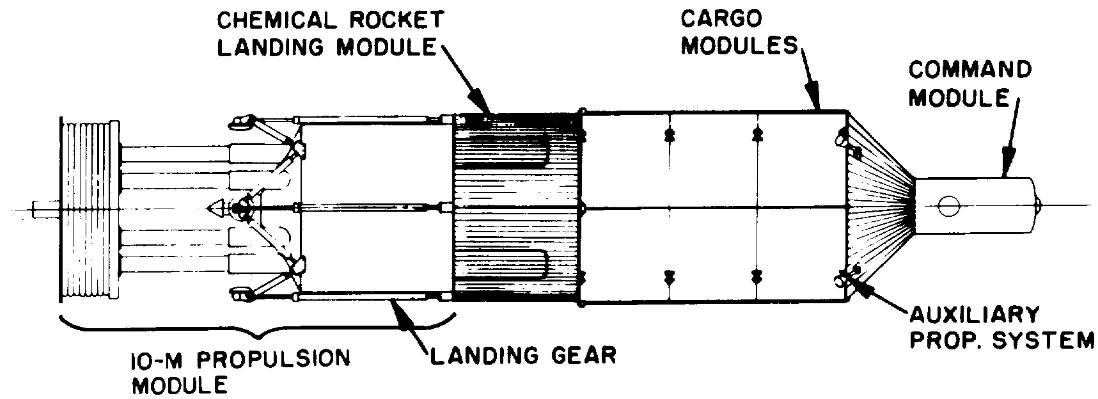

Fig. 3.9--Earth-orbit-to-lunar-surface ferry concept

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 lofter. 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.

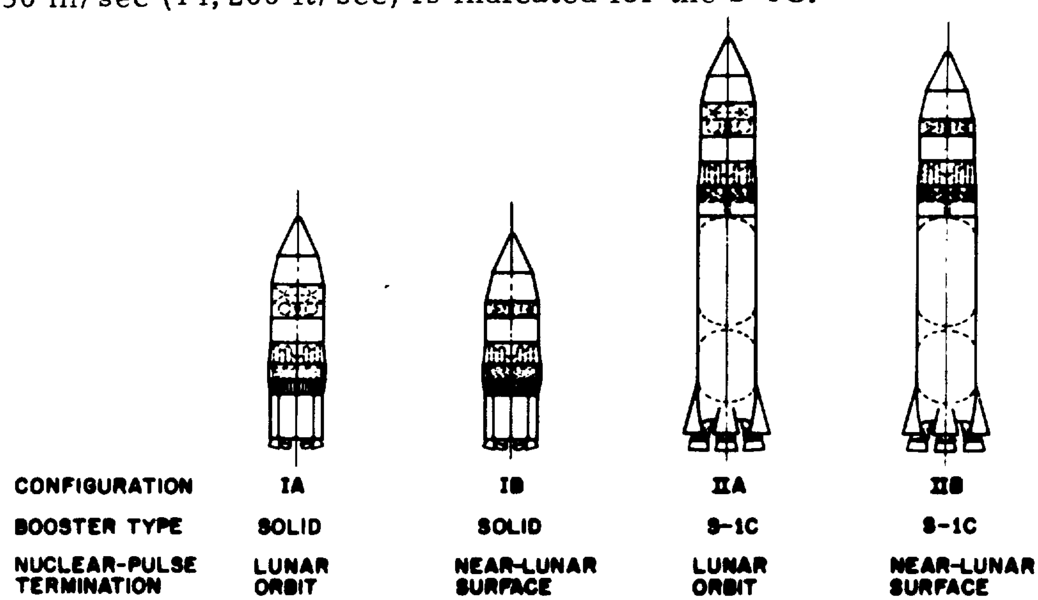

Fig. 3.10--Earth-launched lunar logistic systems

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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.

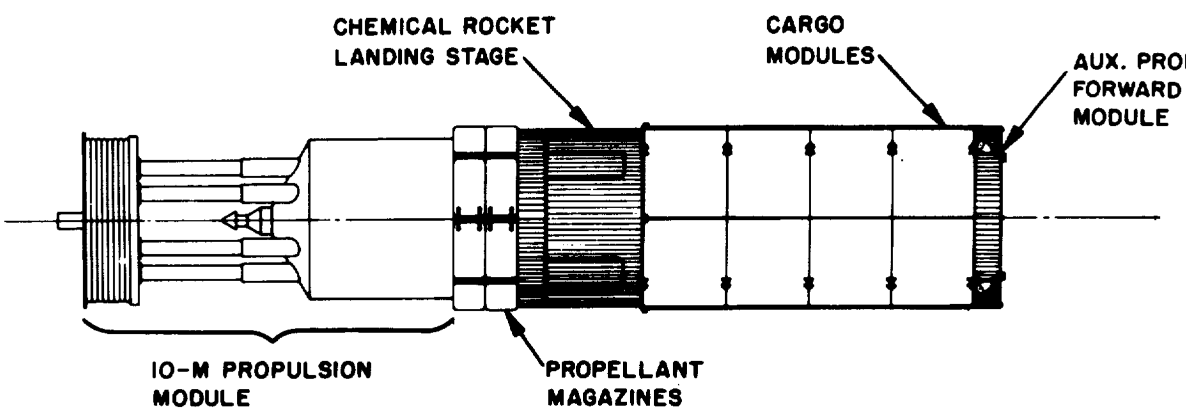

Fig. 3.11--Orbit-launched lunar logistic vehicle

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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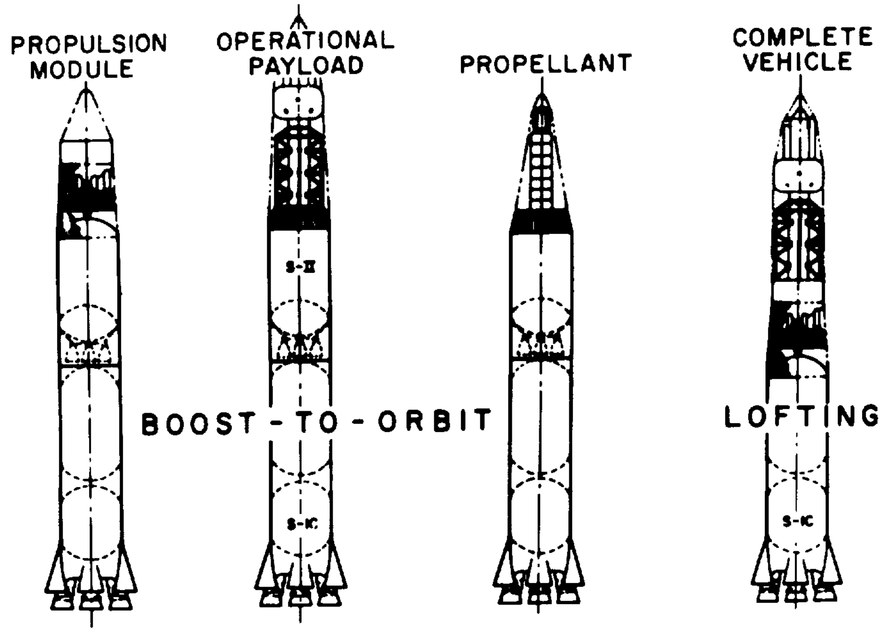

Fig. 3.12--Saturn V with three exploration mission payloads for orbital rendezvous and the S-1C stage as a complete-vehicle lofter

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.

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 (ΔV'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_0'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.

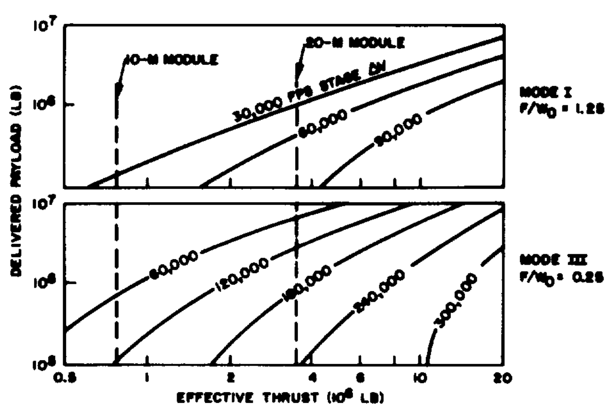

Fig. 4.1--Performance benefits of lower thrust-to-weight-ratio mode of operation

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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 ΔV 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.

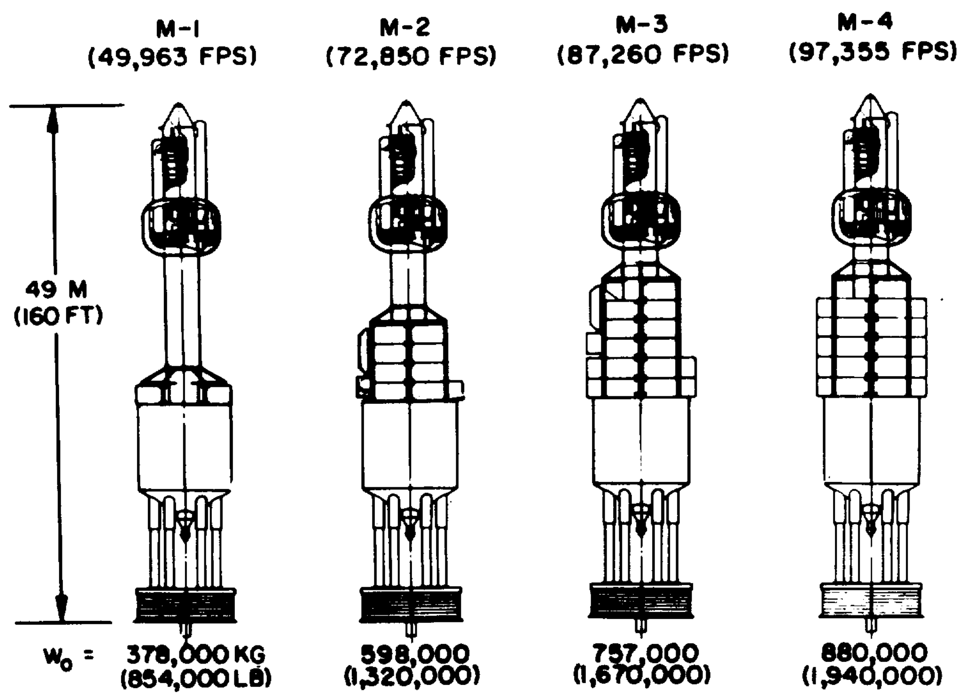

Fig. 4.2--Vehicle loading differences for different mission velocities

The effect of departure year on Mars mission ΔV 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.

[redacted]

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	750	750	750	750
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	212,800	405,266	556,650	677,450
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 ΔV requirements is mission duration. Typical ΔV requirements for a 300-day and 150-day Mars round-trip mission were given in Table 2.6. The 300-day mission (ΔV = 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	750
Total payload	73,800
Propulsion module	95,380
Propellant magazines	38,150
Guidance and start-up fluids	14,940
Propellant	922,730
Earth-orbit departure weight	1,145,000
Earth-orbit departure weight, lb	2,522,000

[redacted]

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.

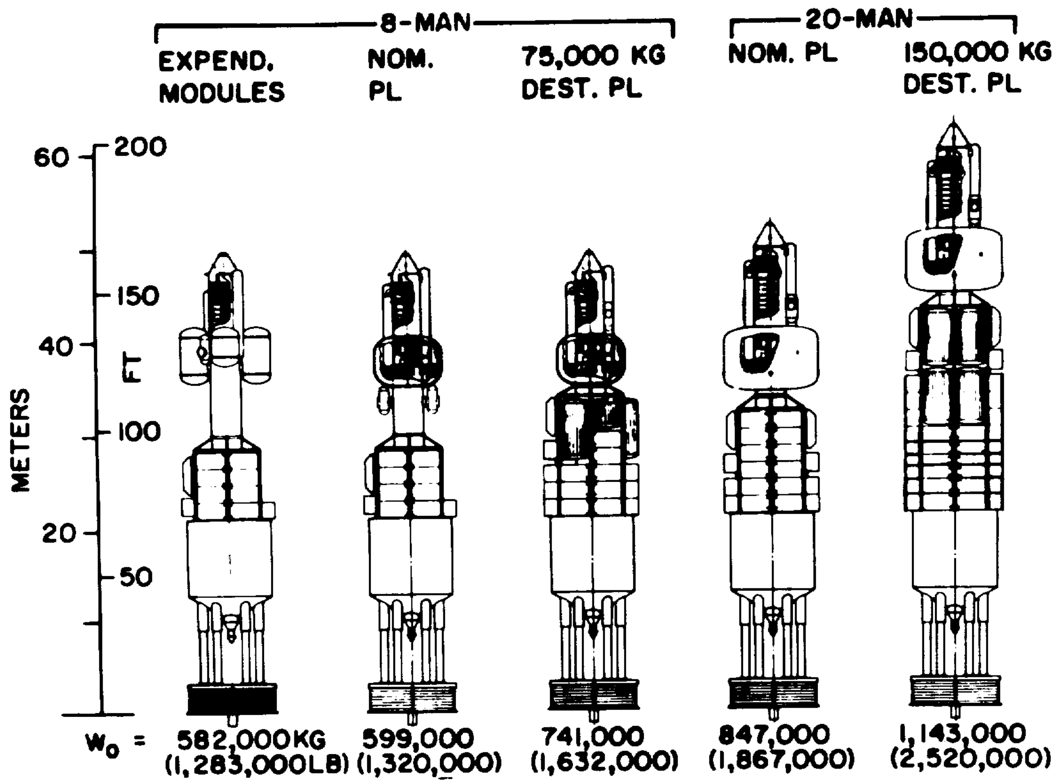

Fig. 4.3--Vehicle configurations and loading differences for various mission payloads

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 <u>not</u> constitute an upper limit--scientists as well as multiskilled astronauts can make the trip; it is <u>not</u> necessary to rely on a faster-than-Apollo reentry at the conclusion of the trip; extra shielding <u>can</u> be carried if required;

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 ΔV = 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
Total Launches Required		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

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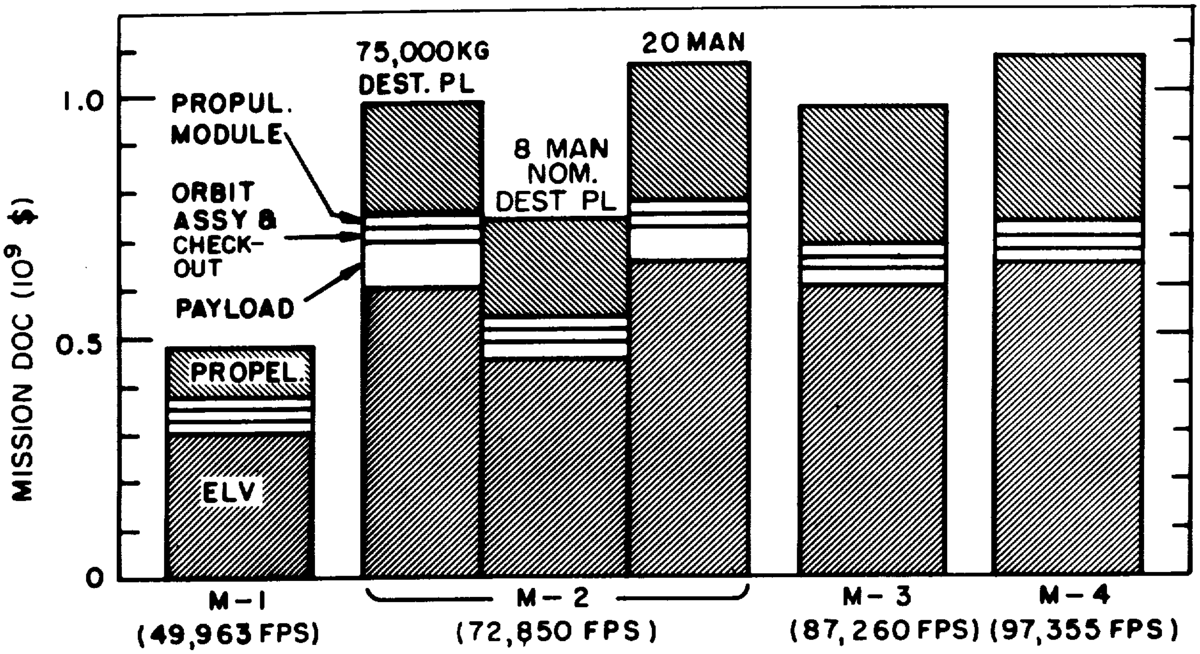

Fig. 4.4--Direct-operating-cost components for different Mars exploration missions

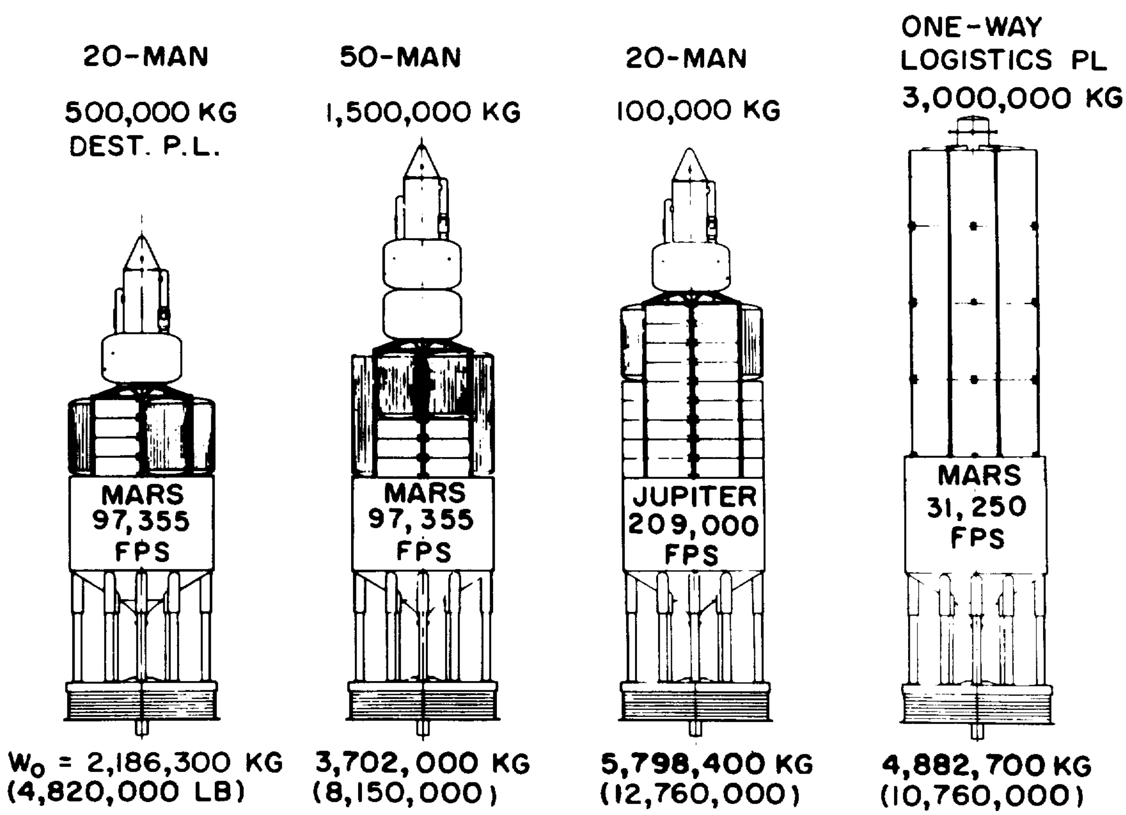

Fig. 4.5--20-m configuration and loading difference for three exploration missions and one logistics mission

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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,	Logistic,	910-day, 20-man, 209,000 fps	
	20-man, 97,355 fps	50-man, 97,355 fps	20-man, 208,800 fps	One-way, 31,250 fps	100,000 PL	500,000 PL
Payloads						
Operational	192,700	271,700	178,700	178,000	289,500	297,500
In-transit	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
Total payload	694,000	1,703,000	680,000	3,178,000	392,000	800,000
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
Earth-orbit-departure weight	2,186,000	3,702,000	7,338,340	4,882,700	5,798,000	7,584,530
Earth-orbit-departure weight, lb	4,815,000	8,150,000	16,200,000	10,760,000	12,786,000	16,750,000

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

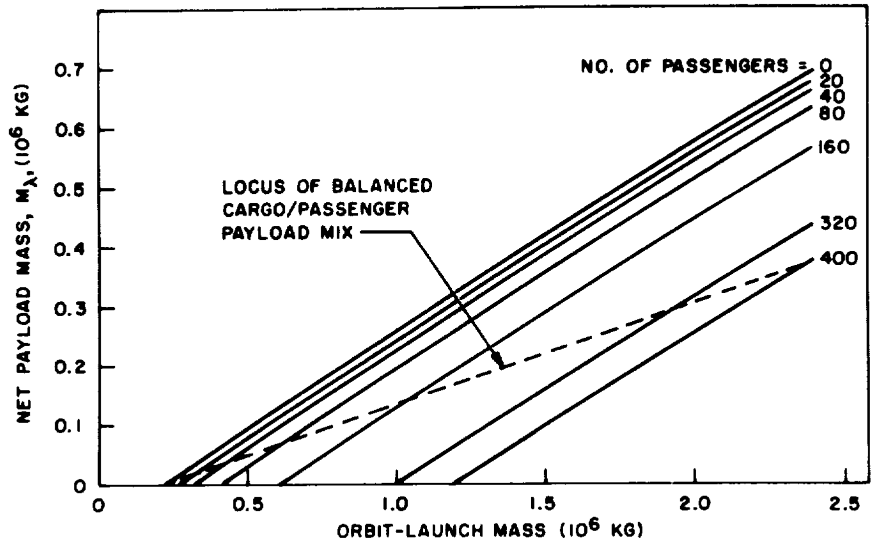

Fig. 4.6--Passenger transportation capability of earth-orbit-to-lunar-orbit ferry system

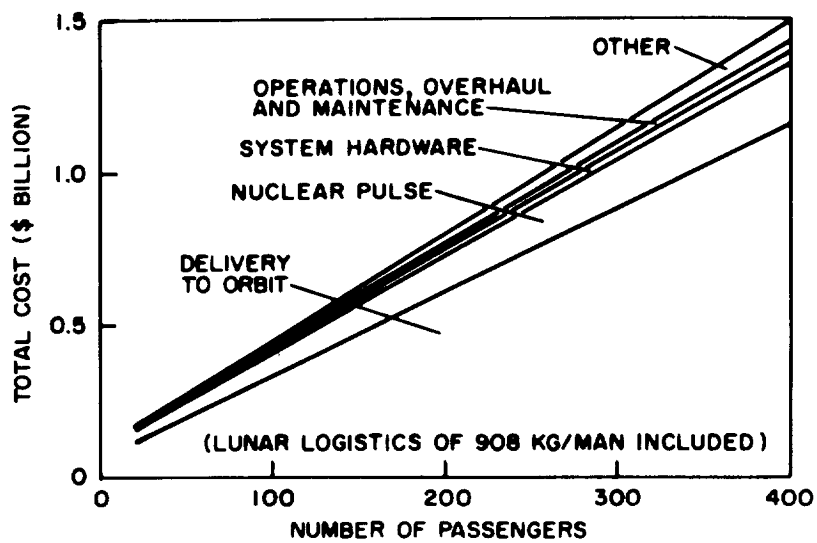

Fig. 4.7--Lunar-ferry direct operating cost for balanced passenger-cargo payload

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 10^6kg. Its more massive and complex lunar

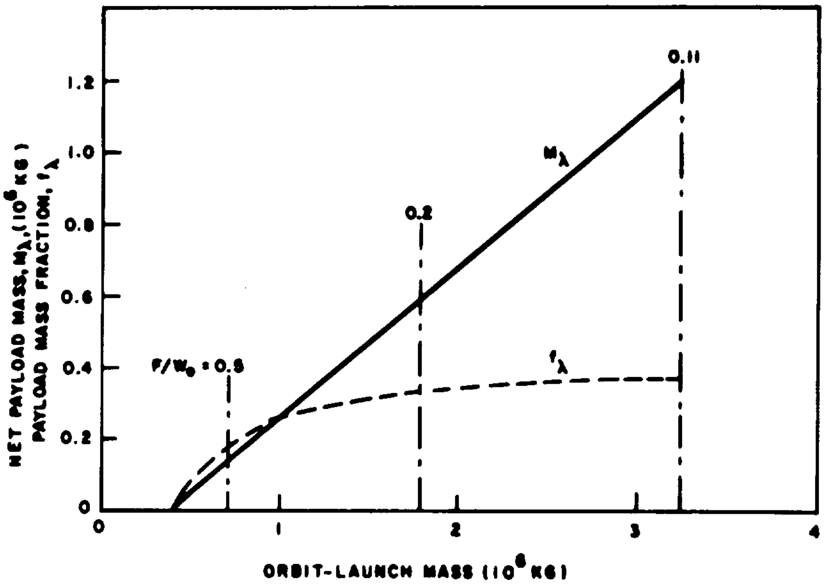

Fig. 4.8--Performance of earth-orbit-to-lunar-surface ferry system

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, ΔV of the S-1C boosted systems (the II's) results in a net payload more than twice that for the minimum ΔV, 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

Configuration	Net Payload (kg)	Net Payload (lb)	Launch Mass (kg)	Launch Mass (lb)	Net Payload Mass Fraction
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

[redacted]

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 ΔV 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)

	Solid-propellant First Stage		Saturn S-1C First Stage	
Cost Element	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
Total direct operating costs	91,761	90,268	110,629	106,220
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.

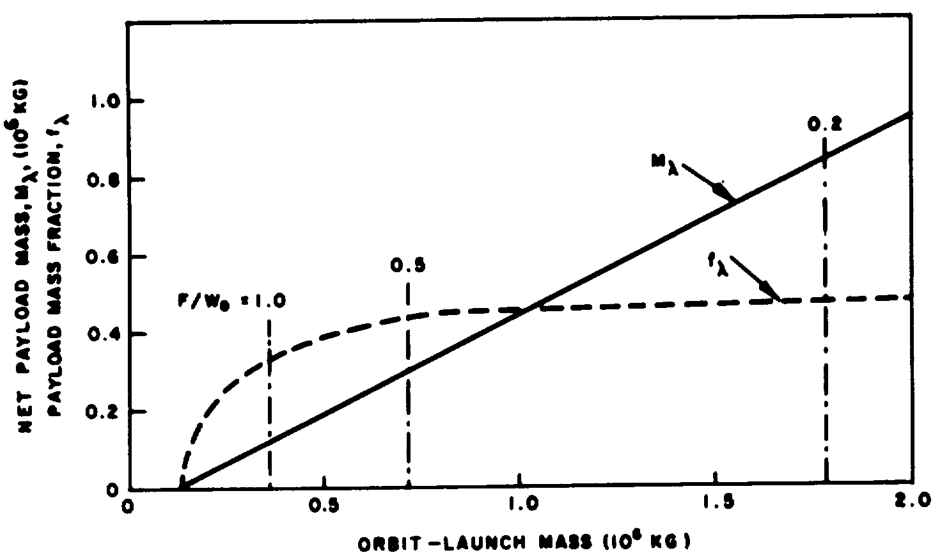

Fig. 4.9--Orbit-launched lunar-logistic-system performance

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 ΔV 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

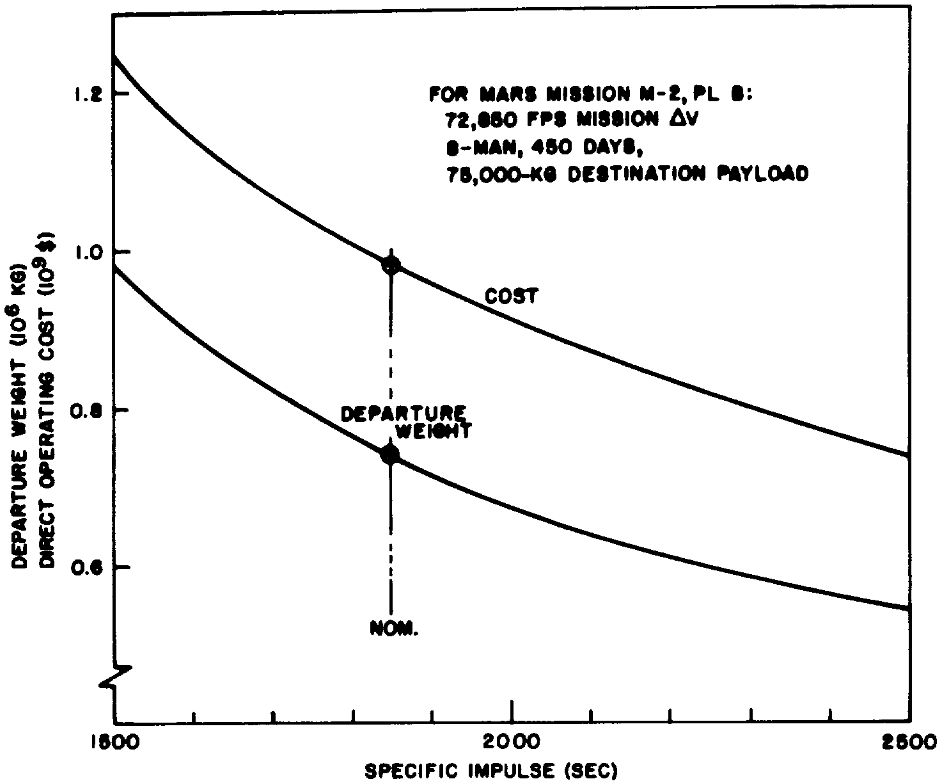

Fig. 4.10--Typical exploration-system sensitivity to specific impulse

($\Delta W_o/\Delta W_i$) is ~3.7/1. Similarly, the operating cost to inert weight ratio ($\Delta DOC/\Delta W_i$) is ~\$3,300/kg (~\$1,500/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 10×10^6 lb to 40×10^6 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 10×10^6 lb; B was assigned an I_{sp} of 20,000 sec and a thrust of 40×10^6 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 ΔV's in excess of 500,000 ft/sec; its DOC/lb is under \$300 until ΔV'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.

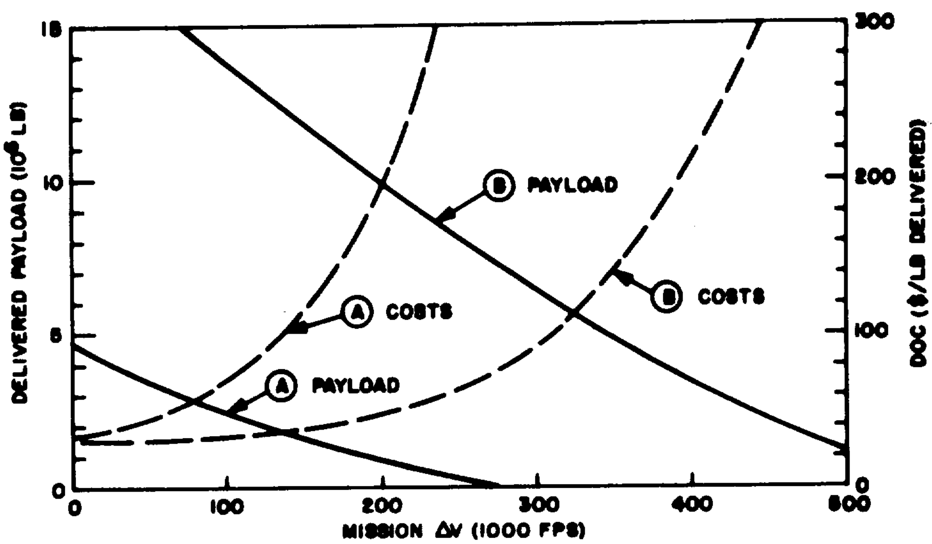

Fig. 4.11--Predicted performance and operating costs for advanced-version vehicles

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 $L >> 2$, 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 $\geq 40°$ north or south, where $L \geq 2$.

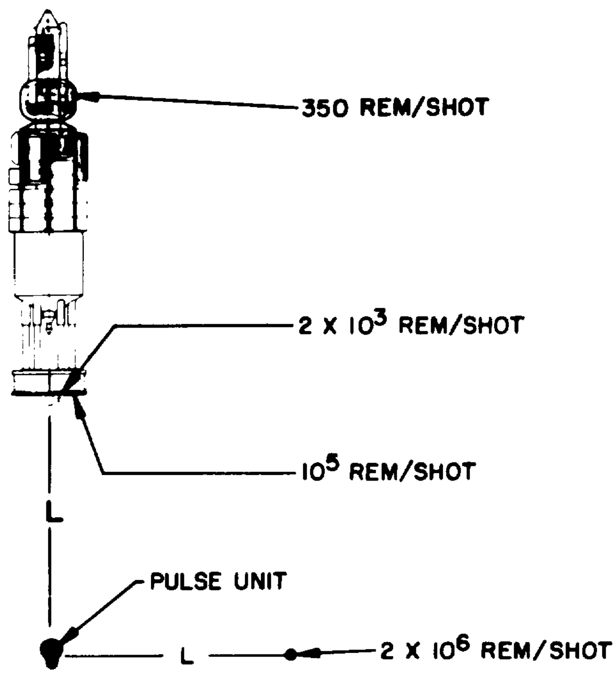

Fig. 5.1--Operational Nuclear Environment About the 10-M Propulsion Module

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 reradiation 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 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 fiberglas 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 dollys 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 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 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 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 meteroid 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 meteroid 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 ($\leq 600°F$) 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.

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 3×10^7 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 3×10^7 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 3×10^7 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 ~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 separting 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.

[redacted]

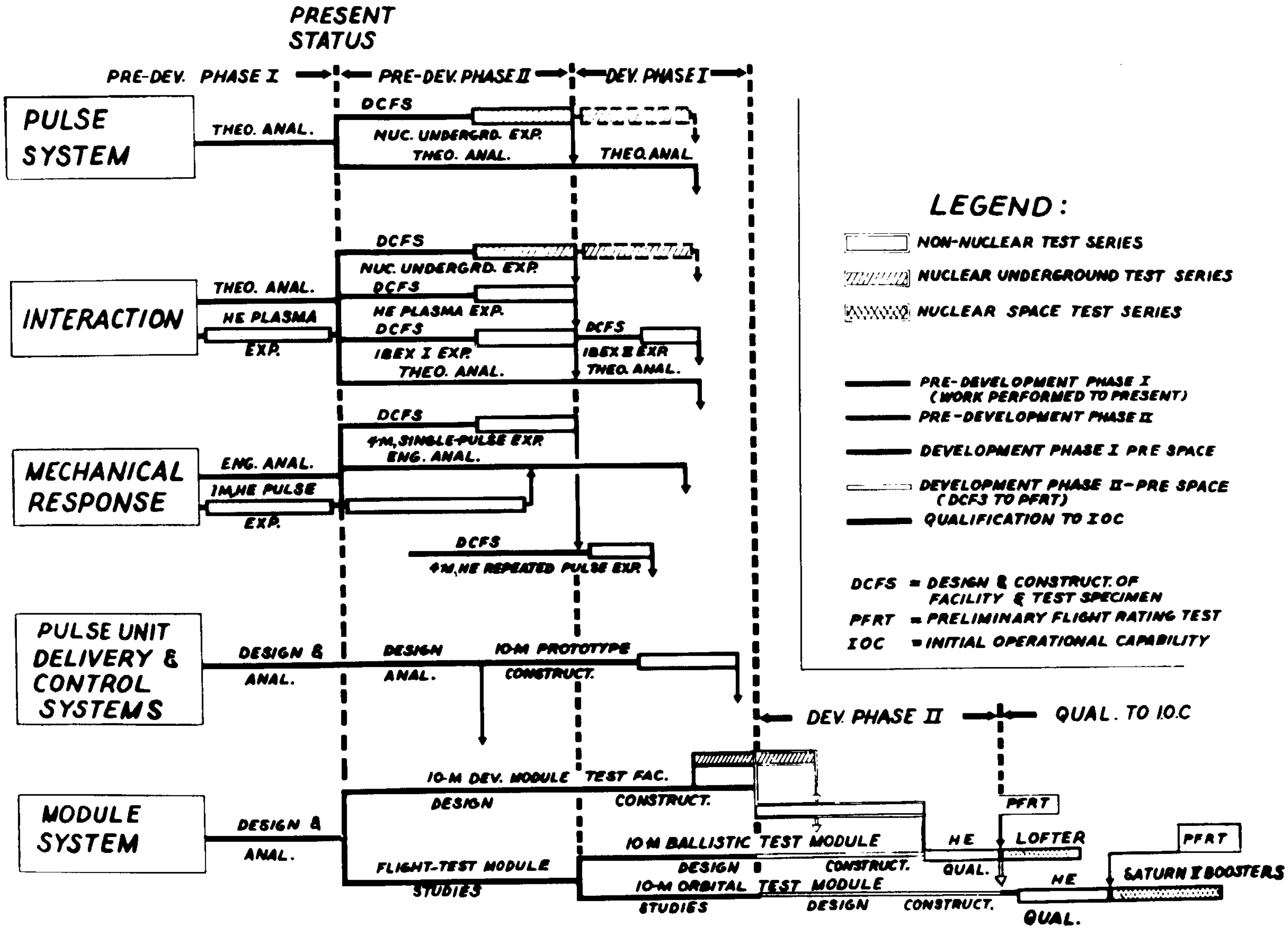

Fig. 6.1--Development phases

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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: PUL. UNIT	NUCLEAR UNDERGROUND: SCALED	NUCLEAR UNDERGROUND: MODULE	NUCLEAR REP. BALLISTIC	NUCLEAR REP. ORBIT
1. PULSE UNIT EXPANSION 1a. PLASMA/PUSHER DIAGNOSIS						● ●		●	● ●	
2. INTERACTION EFFECTS 2a. REP. INTERACTION EFFECTS 2b. PUSHER ABLATION PROTECTION	● ●	● ●	●				● ●	● ●	● ● ● ●	
3. PUSHER RESPONSE 3a. OFF-DES PUSHER RESPONSE 3b. REPEATED PUSHER RESPONSE				● ●	● ● ●		●	●	● ●	
4. S.A. & ATTACH. RESPONSE 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.

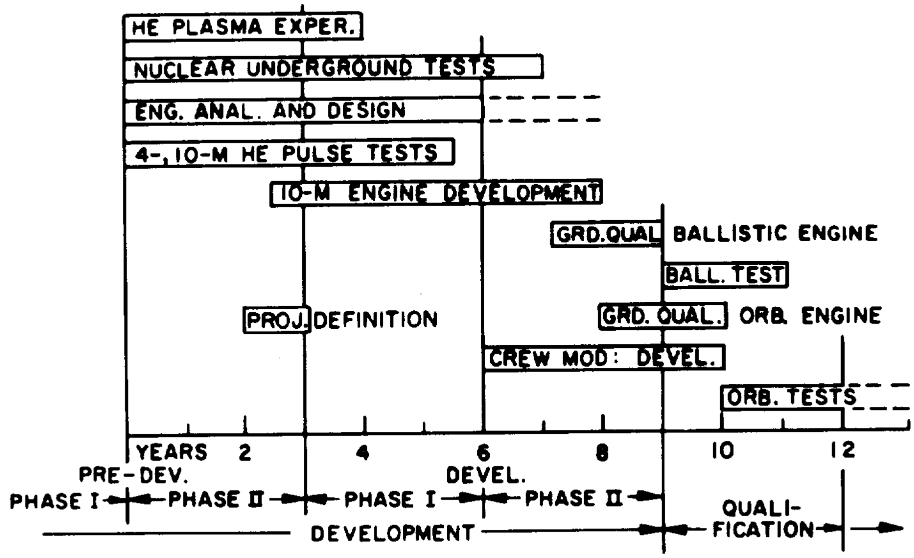

Fig. 6.3--Condensed development schedule

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:

	$ Million
Predevelopment Phase II	63
Development Phase I	152
Development Phase II	239
Qualification program	1,037
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).

[redacted]

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 3.6×10^6 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 Ehricke, as part of an interdivisional assistance effort by General Dynamics/Astronautics. Mr. Ehricke 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. Ehricke's work that includes classified performance data on the nuclear-pulse systems.

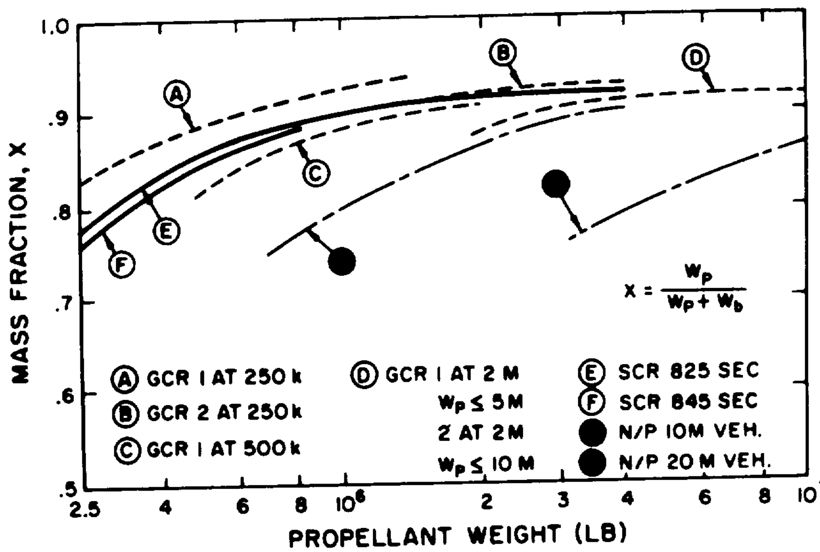

Fig. 7.1--Propellant mass fractions of nuclear propulsion systems vs propellant weight

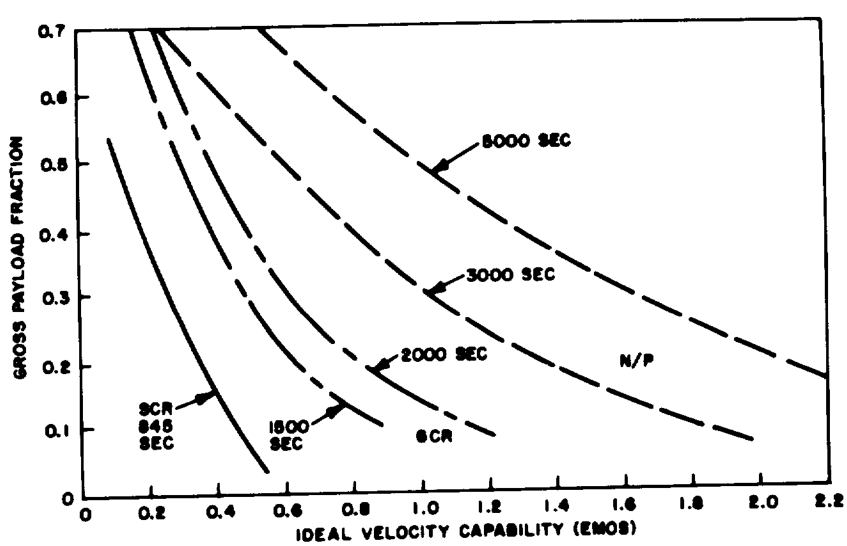

Fig. 7.2--Payload fractions on nuclear interorbital vehicles

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.

[redacted]

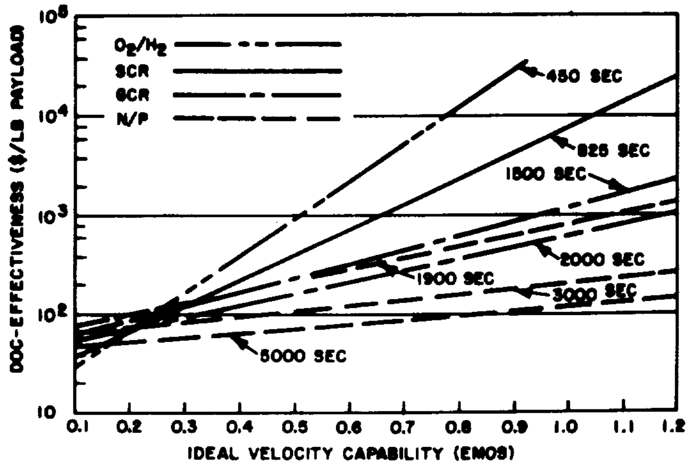

Fig. 7.3--Direct operating cost-effectiveness vs ideal velocities for various propulsion systems

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.

[redacted]

Table 7. 1

MISSION VERSATILITY OF VARIOUS PROPULSIONS 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	Solar System Inner
Solid-core reactor	**	---	**	*	---
Gas-core reactor	**	*	**	***	*
Nuclear pulse	***	***	***	***	***
Nuclear electric; controlled thermonuclear reactor	---	---	*	***	***

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