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Manned Planetary Exploration Capability Using Nuclear Pulse Propulsion

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Summary

This paper discusses the capability of nuclear pulse propulsion in performing manned missions to Mars and other planets using a single-stage vehicle. Recently declassified descriptive data on a 10-meter-diam nuclear pulse propulsion module design (Saturn V compatible) is summarized and a typical complete exploration vehicle employing the module is described. The specific impulse of the propulsion module is 2,500 sec, its dry weight 200,000 lb.

The mission versatility of this single vehicle design is emphasized. The same propulsion module, and essentially the same overall vehicle design, are shown capable of performing single-stage missions requiring velocity increments ranging from under 40,000 ft/sec to over 114,000 ft/sec. Total payloads ranging from 100,000 to 500,000 lb are considered.

For a minimal Mars landing mission, roughly comparable to those proposed for less-capable, multi-staged propulsion systems, an earth-orbit departure weight of 640,000 lb is indicated. A higher payload and propellant loading of the same vehicle, however, is considered preferable. Performing a complete Mars surface excursion mission in 200 to 250 days is shown as one optional way of exploiting the vehicle's capability. Carrying a considerably larger personnel complement, including scientists as well as astronauts, is another. Consistent with the latter idea, two reference design missions, presented in detail, employ retrothrust to return the vehicle to an elliptical earth orbit, avoiding the typically necessary atmospheric reentry maneuver as the final task of a long mission.

A Mars mission capability supported by a single launch, using a Saturn first stage, is also described. In this instance nuclear pulse propulsion is begun suborbitally, starting at an altitude greater than 50 nautical miles.

The major system advantages and systems problems are outlined and briefly discussed. Finally, a series of artist's conceptions of the major operational steps is presented.

I. Introduction

This paper is concerned primarily with the space exploration capability that could be made possible by nuclear pulse propulsion. The major objectives are to better acquaint the space-interested scientific community with (1) the single-stage capability of such a space transportation system, and (2) the broad operational flexibility inherent in one selected propulsion module design.

Only recently have selected nuclear pulse performance data been declassified and made available in the open literature.[1] The concept of employing nuclear explosives to propel space vehicles, however, was initially explored in the late 1940's to middle 1950's. The externally exploded, noncontained-explosion system to be considered here is called ORION; it has been under continuous analytical and experimental study for over seven years. Research on ORION has been sponsored by the Advanced Research Projects Agency, the U. S. Air Force, the National Aeronautics and Space Administration, and General Dynamics. Much progress has been made in understanding the fundamental process, the vehicle-plasma interactions, and in the design and testing of basic hardware elements.

Nuclear pulse propulsion, by its fundamental nature, offers an impressive potential for the peaceful use of nuclear energy. There is no more compact energy source known to man than that provided by nuclear explosives. The propulsion module described below is inefficient and relatively crude compared to eventual pulse propulsion systems now envisioned; yet its performance capability in the realm of space exploration will be seen to be impressive.

II. The Nuclear Pulse Propulsion Module

The principle of operation is summarized here for those not familiar with the concept. Following that a specific propulsion module is described; it furnishes the basis for the performance data to follow.

Principle of Operation

Briefly, the external-explosion pulse propulsion system, or ORION, operates as follows: A large number of small nuclear explosive systems called pulse units are stored within the overall vehicle. These pulse units are sequentially ejected and detonated external to and some distance behind the vehicle. Some of the expanding debris of each explosion, in the form of a high-velocity, high-density plasma, is intercepted by the base of the vehicle, called a pusher plate. The momentum of the intercepted debris is thereby rapidly transferred to the pusher plate, resulting in a high acceleration of the pusher. These accelerations are smoothed out by shock absorbing devices to levels of a few g's in the upper vehicle—well within human tolerance. After compressing the shock absorbers, the pusher returns to its neutral position and is ready to accept the subsequent impulse. The desired total vehicle velocity increment is acquired by varying the number of pulse units expended in a given vehicle maneuver.

The Reference 10-meter Propulsion Module

Conceptual designs of nuclear pulse propulsion modules, or engines, have been prepared in a number of fixed sizes, in order to obtain self-consistent data to support the technology research program. One such reference design, whose gross configuration arrangement and performance data have recently been declassified, is shown in Fig. 1. The module maximum diameter is approximately 33 ft (10 m), selected in part to be compatible with the Saturn V earth launch vehicle. The gross performance data used for the mission performance discussed in this paper is as listed in the figure: specific impulse, 2,500 sec; dry weight of the basic module, 200,000 lb; effective thrust, 780,000 lb.

The propulsion module represents something more than an "engine," in the usual chemical engine terminology. The module includes all pulse unit handling and delivery apparatus, engine controls, auxiliary subsystem, and (in the design pictured) some internal capacity for propellant (pulse units). Additional propellant is carried in propellant magazines attached external to the basic module as shown. The magazines are jettisoned, during coast phases, after being emptied.

The dense, highly storable nature of the nuclear pulse propellant, combined with the external magazine storage arrangement, permits a single design of a propulsion module to be usable over a wide range of mission velocities. This point will be made clear in the performance data to follow, all of which is based on the propulsion module just described.

The basic propulsion module weight includes neither the payload spine shown nor the external structure to support the propellant magazines or payload canisters. Nor does it include, of course, the weight of the empty propellant magazines or any propellant. These are all variables depending on a given mission's requirements. The payload spine and payload support structure are considered part of the overall payload carried by the propulsion module. For computational convenience in generating parametric performance data, downgrading the propulsion module's net specific impulse by a fixed percentage has been found to adequately substitute for the more rigorous accounting of magazine weight, guidance propellant, etc. For the propulsion module described here, having some internal propellant storage capacity, a 4 percent degradation of specific impulse was found to account for magazine weight, magazine support structure, and include an allowance for chemical rocket attitude control during propulsion periods. Thus the corrected specific impulse for the parametric performance calculations is 2,500/1.04 or 2,405 sec.

III. Planetary Mission Performance Capability

Nuclear pulse propulsion has the capability of performing the space transportation function for a wide variety of potential planetary missions. Table 1 lists a number of currently considered manned missions to the planets and the moon. The performance data to follow will indicate that all of them—with the exception of the more ambitious Jupiter missions which require a larger or higher-performance version—can be accomplished in a one-stage vehicle using the reference propulsion module just described.

Performance data has been generated for nearly all of the missions listed. This paper, however, will concentrate on a wide variety of optional ways of performing one mission from the middle of the matrix: manned excursions to the Mars surface, as circled on the table.

Mars Mission Velocity Requirements

Much work on planetary mission velocity requirements has recently been accomplished and more is currently being done.[2,3] Today's data, however, for a specific application study, is typically superceded "tomorrow" as the interorbital transfer is modified by such factors as perihelion braking, Venus swing-by, atmospheric braking, varied perihelion distance, etc.

For purposes of this paper, representative mission velocity information for a 1982 departure to Mars was selected, assuming all required maneuvers are performed by propulsion (no drag deceleration). The data were then simplified by combining the various maneuver requirements into two velocity increments: Earth-to-Mars (ΔV_{out}) and Mars-to-Earth (ΔV_{back}). This is a legitimate simplification when a single-stage vehicle provides all of the propulsion requirements; the separation of "out" and "back" velocity increments is needed only because a significant amount of payload is typically consumed or left at the destination planet.

The velocity increment data used for the basic performance curves to follow assumed an earth return to an elliptical earth orbit (earth approach velocity approximately 35,000 ft/sec). The effect of an alternate approach velocity of 50,000 ft/sec or return to a circular orbit is separately shown. Midcourse corrections totaling 1,000 ft/sec outbound and 1,500 ft/sec on return were allowed in the following data, and a 3 percent performance reserve was applied to the total velocity requirements.

A factor of increasing concern in the consideration of manned planetary explorations is total trip duration. For this reason faster missions than the minimum ΔV 450- to 500-day trips are explored. A 40-day Mars orbit capture period is included in the trip durations. The representative velocity increments, rounded off, are then:

Mission Duration	ΔV_{out}	ΔV_{back}	ΔV_{total}
450-day	30,000 ft/sec	41,600 ft/sec	71,600 ft/sec
350-day	45,000	48,000	93,000
250-day	59,000	55,000	114,000

It should be stated that the above data are known to be conservative. Recent work on faster transit missions by NASA's Ames Research Center, the Space Technology Laboratories, and General Dynamics/Astronautics, as yet unpublished, promises a trip time reduction on the order of 50 days for the velocity increments shown above (by the use of swing-by maneuvers and other fast mission optimizations). For this reason the above mission durations, on the figures to follow, are considered as 400- to 450-day, 300- to 350-day, and 200- to 250-day missions, respectively.

NASA/Ames and STL have also used the Venus swing-by maneuver to minimize both the Mars mission velocity requirements and the velocity variation due to departure year. These results have been at least partially reported.[4,5] In general, it is concluded that allowing 450 to 500 days total trip duration and an earth approach velocity of 45,000 ft/sec, will permit a Mars mission, independent of departure year, for a total ΔV requirement of approximately 40,000 ft/sec. This includes the three maneuvers: earth orbit departure, Mars orbit capture, and Mars orbit departure. Earth arrival makes use of drag deceleration, an extension of Project Mercury capability. For a representative minimum ΔV Mars mission, therefore, the following velocity requirements were used:

Mission Duration	ΔV_{out}	ΔV_{back}	ΔV_{total}
450- to 500-day	25,000 ft/sec	15,000 ft/sec	40,000 ft/sec

The range of Mars mission velocity requirements related above vary from a total of 40,000 ft/sec to 114,000 ft/sec. With the exception of the round-trip Jupiter-moon capture missions (which require upward of 200,000 ft/sec and merit a larger or more advanced propulsion module) this range of velocities covers the complete spectrum of planetary missions tabulated. This range of velocities can also be nicely handled by different propellant loadings of the same nuclear pulse propulsion module, as will now be seen. The implications of this statement for space propulsion system versatility, flexibility in mission planning, and space system economy should not be overlooked.

Mars Mission Payload and Duration Options

Parametric performance data, using the mission velocity requirements just related and a 5-fold variation in mission payloads, are plotted in Fig. 2. The plot shows earth orbit departure weight versus total payload. The total payload is divided 50-50 into "round trip" payload, assumed carried both ways, and "destination" payload consumed or left at the planet. This 50-50 split has been found to be reasonably well approximated by a number of carefully planned Mars mission studies involving surface excursions; an example weight statement will be given later.

Note that the right-hand ordinate divides the orbit departure weight into the number of successful Saturn V deliveries required. All elements of the vehicle and its payload are deliverable to orbit by the 2-stage Saturn V. For missions in the early 1980's it is assumed that the Saturn will be uprated (higher thrust and larger first stage propellant capacity, perhaps) to a delivery capability of 280,000 lb per launch.

The plot of Fig. 2, entirely within the capability of the one propulsion module design being considered here, is seen to cover a wide spectrum of mission options. Orbit departure weights vary from 0.5 to 2.5 million lb. Two reference design points are flagged, based on specific payload breakdowns totaling about 320,000 lb. For the 450-day mission duration the reference design point has an orbit departure weight of 1.15×10^6 lb, which is barely over the specified earth launch capability of four Saturn V's. The referenced 250-day design point has a departure weight of 1.85×10^6 lb, in between the delivery capability of six and seven Saturns.

It should be remembered that the solid-line data of Fig. 2 represents missions requiring no new earth reentry technology nor the carrying of a reentry vehicle throughout the mission; the basic vehicle decelerates to an elliptical earth orbit on return and can be met by an earth-based or space-station-based pickup vehicle.

Observe also that much-reduced mission durations are possible without requiring exorbitant departure weights. Half-year Mars missions, with this propulsion capability, are apparently not at all unreasonable.

If one wishes to compare the nuclear pulse vehicle performance with that of better-understood but less capable multi-staged vehicle systems, the lower, minimum ΔV line should be used. Here too, for a more direct comparison with some mission concepts, a "bare minimum" landing mission payload is flagged. Such payload minimization implies a high degree of expensive subsystem development and optimization and a minimum number of multi-function crewmen—not considered a good approach to either system economy or reliability. The departure weight, however, is only 0.64×10^6 lb and three Saturn deliveries will more than handle it.

Weight statements for the reference design 400- to 450-day and 200-to 250-day missions are shown in Table 2. It is seen that for the 400- to 450-day mission the round-trip and destination payloads do indeed come out as a 50-50 split (with only minimum juggling of the raw data while rounding off). For the shorter 200- to 250-day mission, the round-trip payload is reduced, primarily due to the thereby lowered life support requirements of the mission personnel.

The radiation shelter listed serves a dual purpose. Mission personnel must be within the shielded "powered flight station" during propulsion periods (they typically last from a few to 20 minutes) to avoid radiation from the nuclear pulse units. The shielded compartment also serves as a "storm cellar" during a solar flare or traverse of a radiation belt. The shielding was designed to allow 50 rem per mission from propulsion, which permits a similar dose from solar radiation for a total of about 100 rem per mission. For the 200- to 250-day mission it was determined that the reduced solar radiation exposure was approximately offset by the added exposure to propulsive radiation; therefore, no change was made in radiation shelter weight.

While discussing radiation it should be mentioned that the nuclear pulse radiation flux does not appreciably activate the vehicle structure. Free access may be had to any part of the vehicle within a short time after propulsion shutdown.

The destination payload listed is composed primarily of two Mars excursion modules (MEM) as recently conceived by the Aeronutronic Division of Philco Corp. in a study for NASA.[6,7] The preferred tail-sitter versions of the MEM design were found to fit on a current design of the 10-meter nuclear pulse exploration vehicle with only minor local modifications.

The Exploration Vehicle Configuration

The conceptual design of the complete nuclear pulse exploration vehicle is shown in Fig. 3. Personnel accommodations for 8 men are shown atop the central payload spine. The payload spine provides ready internal access to the propulsion module, and its lower region provides a repair-bay/spares-storage room 10 ft in diameter by 25 ft long. Two Mars excursion modules are carried on opposite sides of the (locally flattened) payload spine as shown. Additional Mars payload can be carried in the large external canisters adjacent to the excursion modules. The number of propellant magazines shown provides for a mission ΔV between that required of the two reference missions of Table 2. For the faster mission, if the excursion modules are carried as shown, a small increase in the payload spine length would be required to accommodate additional magazines. Otherwise, the configuration would not change.

The personnel accommodations provided are "upside down" to permit artificial gravity by slowly tumbling the vehicle (about 4 rpm) during prolonged coast periods. The coast-phase CG location range indicated provides at least a 50-ft rotation radius for normally occupied personnel areas. The shielded powered flight station/escape vehicle, always occupied during propulsion periods, is oriented upright relative to the direction of travel. Its lower level "bunk room," however, can invert its furnishings for more confortable occupancy, if required, during artificial g periods.

Options in the Personnel Complement

The recent studies of planetary exploration systems have all tended to minimize the number of mission personnel. Using the well-understood rocket propulsion systems places such a premium on low payloads that crew size, crew accommodations, work spaces, and life-support systems must all be the very minimum considered feasible. Weight savings for such systems are so important that it has become logical to consider the system benefits of using only physically small crewmen, living in space suits for much of the trip, breathing 3.5 psia pure oxygen, drug-induced low metabolic rates, and other rather desperate-sounding measures. It is believed worthwhile to also consider the opposite approach: Given a more capable propulsion system, what are the benefits gained from the exploration if more personnel and equipment are taken than the minimum necessary to drive the vehicles?

Figure 4 again shows orbital departure weight and the number of Saturn V deliveries required—this time versus the number of mission personnel. It is clearly seen that with a nuclear pulse interplanetary vehicle it does not cost much in orbital mass to significantly increase the personnel complement. The data shown are based on a rather thorough study of the requirements for both 8-man and 20-man personnel complements,[8] with the variables being such that linear interpolation is indicated.

Of significance here is the fact that, again, the complete gamut of variations shown on Fig. 4 represent variations in loading of the same propulsion module and almost the same overall vehicle. For every two additional persons added, another stateroom is needed, some additional work space and equipment, and an incremental increase in recreation space, the shielded compartment, and, of course, in food and other ecology supplies. These accommodations requirements need not be frozen until a few years, at most, before the departure date. This is quite unlike the multi-stage rocket situation we are used to thinking of for such missions. In the multi-stage situation, the size of tankage and motors of each stage is directly dependent upon the mission payload which must then be decided far in advance; any very significant increase in that payload results in a "no-go" mission.

Another useful way to consider the data of Fig. 4 is the notion of "loading factor." From the data used, each additional person on a 400 to 450-day mission increased the round-trip payload by about 10,000 lb. This resulted in a departure weight increase of about 24,800 lb, or a loading factor of 2.48 to 1. Differently stated, for each additional, say, 100 lb of inert weight—whether it be telescopes, cornflakes, meteoroid protection, or a heavier structure—one needs only add 148 lb of additional propellant to carry it through the journey! No vehicle change is required. For the 200 to 250-day mission the comparative numbers are 8,840 lb per added person (shorter trip, less supplies) resulting in a departure weight increase of 38,000 lb or a loading factor of 4.3 to 1.

The slope of a typical departure weight versus crew size curve for a multi-stage, 850-sec, rocket-propelled Mars system is also shown. This is not directly comparable to the other data of the plot since the multi-staged systems are not being designed for total mission velocities of 71,600 or 114,000 ft/sec, which was the basis for the nuclear pulse curves shown. The slope for the multi-staged system curve is much steeper not only because the mass ratios are higher due to the lower specific impulse, but also because it represents a complete system growth factor rather than a loading factor. The whole system increases in size if another man is added. The implication of the staged system curve slope—in launch facilities, physical vehicle sizes, direct operating cost, etc., as well as in departure weight—is, of course, why there is currently so much thought of minimizing the crew and the mission payload.

Options in Earth Return Conditions

The referenced designs previously discussed were single-stage vehicles that depart from a circular earth orbit and return to an elliptical earth orbit (earth approach velocity approximately 35,000 ft/sec). The returning vehicle can then be met by an earth-based or orbit-based reentry vehicle, and the mission personnel returned to the earth surface when conditions are favorable—using a "fresh" reentry pilot and a recently checked reentry vehicle. The nuclear pulse vehicle then also remains available for restocking and reuse if desired.

It may be desirable, instead, to carry some additional propellant and return to a circular earth orbit; or conversely, to save propellant, reduce departure weight, and approach earth at a higher speed.

By today's mission thinking, return to a circular earth orbit seems sort of an "old man's mission," and almost takes the thrill out of the whole trip! But it might be acceptable to the astronauts, and would seem to be a comforting notion to the scientists we suggest might go along.

Approaching earth at higher-than-parabolic speed, like 50,000 ft/sec, means, of course, that the basic vehicle goes by earth while the mission personnel depart it in an earth reentry module that has been carried along for that purpose. The reentry corridor boundaries and 50,000-ft/sec reentry vehicle characteristics are now fairly well understood, so that feat is considered realistic for mission planning purposes.

Figure 5 shows alternate earth departure weights for the two reference missions, when returning at 50,000 ft/sec and when returning to a circular orbit. The reference designs do not carry an earth reentry vehicle for the trip to Mars and back. Hence the 50,000-ft/sec mission was first computed carrying the same payload as for the slower return missions; then the incremental departure weight due to the reentry vehicle was added as shown. The reentry vehicle weight for 8 men and a 50,000-ft/sec approach was 15,400 lb, taken from an earlier study.[8]

IV. Single Launch Mission Capability

All performance data discussed to this point concerns an orbital-load-up mode of operation. The propulsion module is delivered to orbit by one launch of the 2-stage Saturn V; the operational payload (personnel accommodations, remaining vehicle structure, some supplies) and the Mars excursion modules are delivered in another launch, with subsequent launches carrying mixes of propellant and miscellaneous small payloads. This is not, however, the only mode of operation for the nuclear pulse vehicle. It will also perform a respectable Mars mission based on a single earth-launch using Saturn V hardware.

The incentives for a single-launch mission are primarily economics and operational simplicity. Direct operating costs for the orbital load-up Mars missions, as is typically true for space systems, are dominated by the cost of deliveries to earth orbit. Reducing the delivery problem to one launch is obviously desirable from both a cost and operational viewpoint.

Figure 6 pictures the single-launch operational situation and shows the Saturn V Apollo configuration for a size comparison. Again, the nuclear pulse vehicle shown is the same one previously described, but this time fully assembled and loaded to a gross weight of 1.4×10^6 lb. The Saturn S-1C stage is, again, assumed uprated in thrust and propellant capacity and is now also structurally modified as required to carry the nuclear pulse vehicle as its upper stage.

The nuclear pulse vehicle arrives in orbit with its gross weight reduced to somewhat over 1×10^6 lb, having consumed about 350,000 lb of propellant in getting there. Some additional propellant is allowed for shakedown operations in orbit before departing on the Mars mission, so that the earth-orbit departure gross weight is about an even 1×10^6 lb. Referring back to Fig. 2 this departure weight is seen to provide a fair number of mission options; for example, 250,000 lb of total payload for a 400- to 450-day mission returning to an elliptical earth orbit, or some 430,000 lb of payload if satisfied with a minimum ΔV, 450- to 500-day mission.

The single launch, then, can provide earth launch support for the entire Mars mission except for mission personnel. Mission personnel are assumed delivered to orbit separately, since it would appear unnecessary to man-rate the interplanetary nuclear pulse vehicle for the relatively critical self-delivery to earth orbit.

From the nuclear pulse vehicle initial gross weight (1.4×10^6 lb) and 780,000-lb effective thrust previously given, it is seen that its initial thrust-to-weight ratio is 0.55. This will be recognized as quite low compared to a typical launch vehicle second-stage separating at some 8,000-ft/sec actual velocity. Detailed trajectory computations, however, confirm that such a thrust-to-weight ratio maximizes the weight delivered to orbit, when the specific impulse is like 2,500 sec. A quite high gravity-loss is incurred in the process of getting to orbit, and the trajectory is more "lofted" than is usual. The initial indications of the benefits of such lowered thrust-to-weight ratios came from advanced mission personnel of the NASA Marshall Space Flight Center; their performance calculations were subsequently found to essentially confirm the boost-to-orbit performance indicated in Fig. 6.

System economics and launch-vehicle simplicity, if considered alone, would suggest that the single-launch mode of operation be the primary mode considered. Questions concerning surface hazard aspects of the sub-orbital start, however, keep consideration of this mode of operation as an attractive alternative; not the planned mode for early system operations.

V. System Advantages and System Problems

There are several very significant advantages to a space propulsion system having the characteristics of nuclear pulse. The wide mission flexibility—whereby a single design can handle a vast variety of payloads and mission velocities—has already been discussed in considerable detail. Other advantages—predeparture shakedown, system cost savings, and maintenance capability—will be more briefly treated. Some of the major system problems will be mentioned also.

Single Vehicle Operational Advantages

The nuclear pulse space propulsion system being discussed, one needs to remember, operates as a single vehicle once leaving orbit, not as a series of stages. In a thoughtful comparison of operational characteristics, a one-stage, multi-start vehicle such as this has many advantages over any multi-stage system. Such a one-stage vehicle can be exercised on test flights or on a shakedown cruise, by repeated earth-orbit perturbations in its proper space-vacuum environment, until all systems are debugged and operating personnel are familiar with their peculiarities. Incipient or "break-in" failures, so familiar to the reliability-conscious, can be taken care of before departure. Actual operating performance can be verified, the CG can be retrimmed for minimum directional control, extra supplies or unexpectedly high-loss or high-consumption expendables can be taken aboard, and various other prudent actions taken—even to the extent of cancelling the trip should the particular vehicle turn out to be unsatisfactory. Figure 7 is a reminder of some of the benefits of such shakedown operations.

Such test flights, shakedown cruises, or test drives are standard procedures in aircraft, marine, automotive, and other transportation fields. In these fields of transportation, to contemplate a long voyage without such tests would be almost unthinkable. Only the expendable, one-shot, and multi-stage rocket industry has has to cope with operating untried "transportation" hardware. Only the rocket industry, therefore, has grown accustomed to expensive, synthetic means of exercising such vehicles and in predicting mission success or failure on that basis. One should not forget that there is no real substitute for testing the actual vehicle that the mission will use.

Economic Advantages

Only a few of the economic advantages of such a space vehicle will be mentioned. Payload and mission velocity versatility is an obvious one; no need to redesign a whole new series of stages to perform a mission 50 percent "bigger" or "smaller" than the last one. The reduction in the number of simulated flight tests needed to ensure a tolerable reliability is another economic plus, due to the actual-vehicle shakedown capability just discussed.

The dense, highly storable nature of the propellant is an operational cost advantage since it relieves the number of earth launch vehicles and launch facilities required. No need to rush successive launches of propellant to orbit to keep boil-off under control; there is no boil-off. Similarly, the rendezvous-orbit decay is much slower, since the compact vehicle and dense propellant result in a high ballistic coefficient (W/C_dA) of the orbiting vehicle.

The nuclear pulse propellant, however, by its nature is rather expensive, if compared in cost-per-pound, for example, to hydrocarbon fuels. While details of propellant costs are classified data due to the nuclear explosive device involved, the overall propellant costs on a typical Mars mission are a noticeable fraction of the total direct operating cost. This is rarely the case for space systems burning rocket propellants.

Perhaps the most significant economic advantage of such a high-capability space vehicle lies in the relieved requirements for mission subsystems. Were a propulsion system such as nuclear pulse assured and under development, there would be no pressing need for a nearly-closed ecology system, less need for the ultimate in strength/weight structural design, more permissible margin-for-safety against the unknowns of space, and no necessity for faster earth reentry systems nor drag deceleration at the planets. When an additional 100 lb can be carried by simply loading with it an extra 148 lb of propellant, many subsystem problems become easier to solve.

All this does not mean to imply that system and subsystem design and reliability will not remain significant problems for several-hundred-day missions; even if weight bogies are relieved a bit and the entire system can be exercised before departure. It does mean that such capability should significantly reduce the total system development time and cost, in addition to making the mission less of an operational gamble.

Enroute Maintenance Capability

The nuclear pulse vehicle appears to be more maintainable enroute than are most conceptual space vehicles, especially other nuclear ones. Figure 8 illustrates some of the coast period maintenance concepts of the current vehicle designs.

The nuclear pulse vehicle has a low residual radioactivity, even after performing a large ΔV maneuver. The shielded powered-flight-station may be departed immediately on propulsion shutdown, and within a short time activation levels should not preclude manned access to any portion of the vehicle. With the propellant packaged in discrete, dense containers, access to all vehicle components can be made available whether inside of the propulsion module or outside.

There are no cryogenics in the propulsion system, and operating temperatures never exceed a few hundred degrees Fahrenheit. The space vacuum appears to be the most unique operational environment. Thus, structural components are currently designed in the familiar steels, aluminum alloys, and occasionally titanium. There are relatively few stored fluids aboard and containers for these are within the structural shell and are accessible. The components, therefore, are considered reasonably receptive to coast period maintenance and repair. To facilitate this, the vehicle conceptual designs provide space and weight allowances for a relatively large and well-equipped repair-bay and spares-storage area.

Developmental Problems

Discussion of nuclear pulse propulsion, up to this point, has largely concerned the positive features of the project. There are also some problems and scientific uncertainties. In the interest of a balanced presentation, unfortunately, the problems cannot be discussed here in the detail given to the performance and operational advantages of nuclear pulse systems. It is precisely these important problems and uncertainties (at least, so it is hoped) that have received attention in the research and experimental efforts of the past seven years. The major technical problems and some of the experimental research techniques have recently been summarized in a brief technical paper,[1] but details of the technical status and development approach involve technology related to that of nuclear weapons, and must be limited to classified communications.[9,10]

Problems in the understanding and potential development of nuclear pulse propulsion can be grouped into three classes: technical, programmatic, and political. The first two are ever-present in any new system development effort; the third. in the international, nonpartisan sense used here, usually is not a problem.

The technical problems, naturally, have been the subject of the past research and experimental efforts. The basic problems; ablation, explosive debris—pusher plate interactions, and impulsive loading of structures; are, now, not only well defined but progress has been made in their practical solution. Ground-based and nonnuclear development techniques have evolved. Much use has been made of relatively new experimental tools: high explosive plasma generators, sheet-high-explosive impulse generators, and the like.[1]

The programmatic problems are quite like those of other advanced R-and-D projects. They reflect the very real environment in which R-and-D programmatic decisions are made: limited budgets, certain defined requirements for advanced programs. It is perhaps enough to say that currently there is no immediate "requirement" for the degree of space propulsion capability discussed in this paper.

The political problem, rather obviously, stems from the fact that nuclear pulse propulsion uses in small scale the same energy source used for nuclear weapons. The recent nuclear weapons test ban treaty, seeking ultimately "the discontinuance of all test explosions of nuclear weapons for all time," by its language also excludes other nuclear explosions except for underground tests. The treaty as it stands prohibits, then, the operation of nuclear pulse vehicles as well as the eventually necessary final developmental and qualification testing. The treaty, however, provides procedures for its own amendment—and the spirit of the treaty is clearly not to prevent the development of advanced space propulsion nor to hinder the scientific exploration of space.

The above problems, in total, sound rather formidable. Indeed, they appear so at this writing. The space system advantages that have been discussed, however, appear to be real and valid ones—unless the performance basis is found to be grossly optimistic. Furthermore, the technical problems appear to be well understood, means for their solution are planned in reasonable detail, and a series of logical development steps leading to operational capability can be described. Estimated development costs and schedule times are of the same order as for most other advanced vehicle developments. In fact, if today's understanding of the development task is reasonably accurate, and the performance potential not grossly in error, the payoff/risk ratio must be one of the best in aerospace history. This is particularly true since the answers as to practicality and performance become available very early in the program.

VI. Mission Operational Scenes

The following illustrations depict, for the most part, artist's concepts of operational scenes on a planned Mars exploration mission sometime in the future. For a bit of comparison first, Fig. 9 is a flight test operational scene from the past. Here is shown a 1-meter, high-explosive pulse-propelled research vehicle that was flown repeatedly in late 1959 and early 1960. This was a relatively simple vehicle, intended primarily to demonstrate its stability characteristics and to acquire experience in the problem of repeated explosive charge ejection.

Figure 10 is a launch "scene" showing the complete earth launch requirements to support a Mars exploration. The central 2-stage Saturn V carries the nearly-dry propulsion module; the Saturn in the left foreground carries the operational payload structure and the Mars excursion modules; while the other Saturns carry primarily magazines of dense propellant. Artistic license has drawn the four launch sites rather close together for illustrative purposes.

Figure 11 shows the Mars exploration vehicle coupled together and fully loaded for the trip. It is performing a final shakedown operation prior to departing earth orbit. In the background is a manned orbiting station that could serve as a base for the assembly-and-loading crew and as a staging point for the Mars mission personnel.

Figure 12 depicts the exploration vehicle enroute some two days out from earth. Two crewmen are examining the vehicle's primary shock absorbers at the same time several empty propellant magazines are being ejected.

Figure 13 is an operational scene in Mars orbit. One Mars excursion module is making a descent to the surface while the second is being checked out to stand by. The nuclear pulse vehicle, nearby, continues to be the base of operations.

The final Fig. 14 pictures the nuclear pulse vehicle having again returned to earth orbit. It is intact and complete except down now to its reserve propellant supply and minus the excursion vehicles left at Mars. An earth reentry vehicle from the manned orbiting station has coupled to the nuclear pulse vehicle to pick up personnel for return to the surface.

References

1. Nance, J. C., "Nuclear Pulse Propulsion," Proceedings of the Eleventh Nuclear Science Symposium of the Institute of Electrical and Electronics Engineers (IEEE) (to be published).

2. Ehricke, K. A., "Study of Interplanetary Mission to Mercury Through Saturn with Emphasis on Manned Missions to Venus and Mars 1973/82 Involving Capture," General Dynamics/Astronautics report GD/A 63-0916, September 30, 1963.

3. Planetary Flight Handbook, Space Flight Handbooks Vol. 3, Office of Scientific and Technical Information, National Aeronautics and Space Administration, 1963.

4. Hornby, H., "Ames Research Center Mission Studies Summary," Proceedings of the Symposium on Manned Planetary Missions 1963/1964 Status, NASA Technical Memorandum NASA TMX-53049, June 12, 1964.

5. Sohn, R. L., "Summary of Manned Mars Mission Study," Proceedings of the Symposium on Manned Planetary Missions 1963/1964 Status, NASA Technical Memorandum NASA TMX-53049, June 12, 1964.

6. Dixon, F. P., "Study of Manned Mars Excursion Module," Proceedings of the Symposium on Manned Planetary Missions 1963/1964 Status, NASA Technical Memorandum NASA TMX-53049, June 12, 1964.

7. "Preliminary Studies Completed on 3-Man Mars Excursion Module," Aviation Week—Space Technology, Vol. 81, No. 20, November 16, 1964, p. 53.

8. "Nuclear Pulse Space Vehicle Study," General Atomic report GA-5009, Vol. I–IV, 1964, NASA Contract NAS 8-11053 (Secret report).

9. Day, E. A., and J. C. Nance, "Nuclear Pulse Propulsion (ORION) Technical Status Summary and Ground-oriented Development Plan," paper to be presented at the AIAA Propulsion Joint Specialists Conference in Colorado Springs, Colorado, June, 1965, and to be published in the Proceedings (Secret report).

10. David, C. V., "Nuclear Pulse Propulsion (Project ORION)—1965 Engineering Status," paper to be presented at the AIAA Propulsion Joint Specialists Conference in Colorado Springs, Colorado, June, 1965, and to be published in the Proceedings (Secret report).

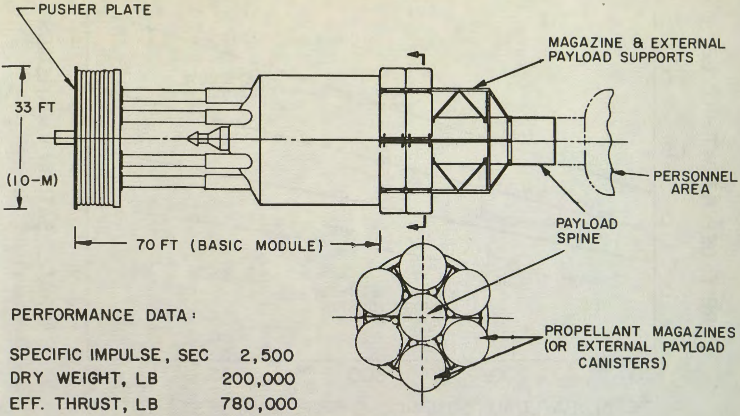

Fig. 1—The 10-meter nuclear pulse propulsion module

LUNAR:	FERRY	BASE SUPPORT	LOGISTICS
VENUS:	ORBITAL EXPLOR.	ORBITAL RECON. STATION	
MARS:	ORBITAL CAPTURE	SURFACE EXCURSION	SYNODIC SURFACE BASE
MERCURY:	ORBITAL CAPTURE	ORBITAL RECON. STA.	SURFACE EXCURSION
JUPITER:	FLY-BY	MOON ORBIT	MOON SURFACE EXCURSION

PLUS ADVANCED PROBES

PLUS FAST RESCUE MISSIONS

Table 1—Potential planetary applications for nuclear pulse propulsion

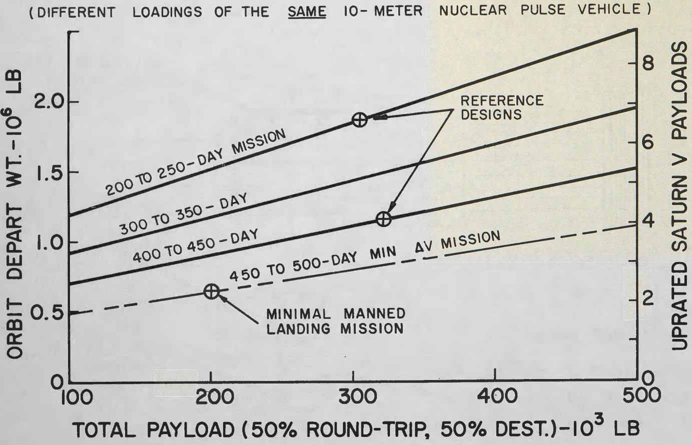

Fig. 2—Mars mission options using a 10-meter nuclear pulse vehicle

	200 TO 250-DAY		400 TO 450-DAY	
ROUND-TRIP PAYLOAD, LB		144,000		160,000
STRUCTURE, FURNISHINGS	50,000		52,000	
RADIATION SHELTER	40,000		40,000	
SPARES, MAINT. EQUIP., ETC	10,000		12,000	
FOOD & ECOLOGY SUPPLIES	21,000		28,000	
ABORT & SPIN PROPELLANT	14,000		18,000	
CONTINGENCY	7,400		8,400	
PERSONNEL (8)	1,600		1,600	
DESTINATION (MARS) PAYLOAD		160,000		160,000
MARS EXCURSION MODULES (2)	130,000		130,000	
UNMANNED RESEARCH VEHICLES	12,000		12,000	
SCIENTIFIC EQUIP & PROBES	10,000		10,000	
MISCELLANEOUS	8,000		8,000	
NU PULSE PROPULSION MODULE		200,000		200,000
PROPELLANT AND MAGAZINES		1,340,000		625,000
ORBIT DEPARTURE WEIGHT (LB)		1,844,000		1,145,000

Table 2—Summary weight statements for two reference-design Mars missions

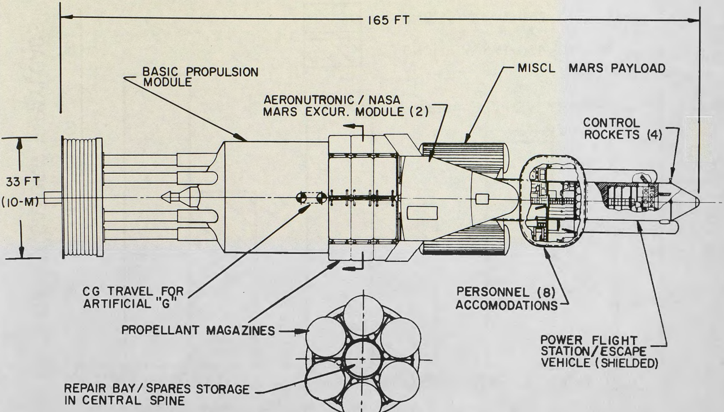

Fig. 3—A 10-meter nuclear pulse vehicle for an exploration trip to Mars

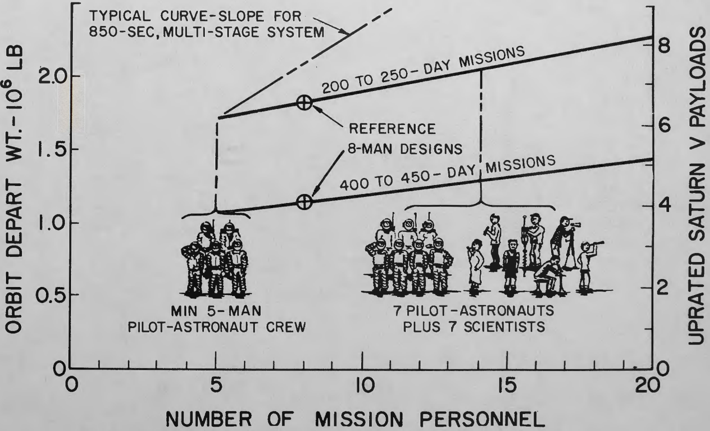

Fig. 4—Effect of personnel complement on departure weight of the vehicle

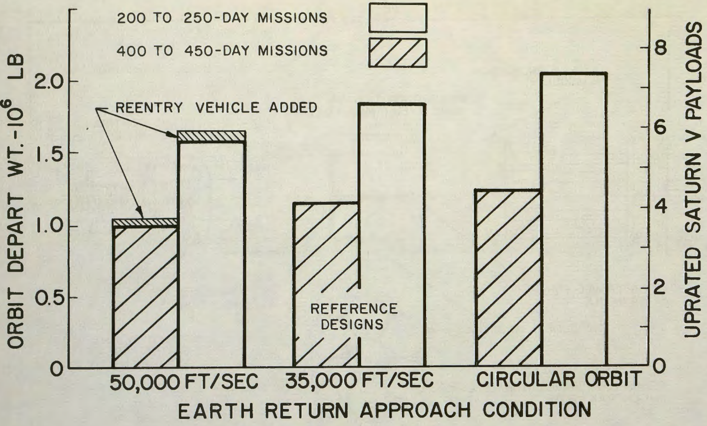

Fig. 5—Vehicle departure weight sensitivity to earth return conditions

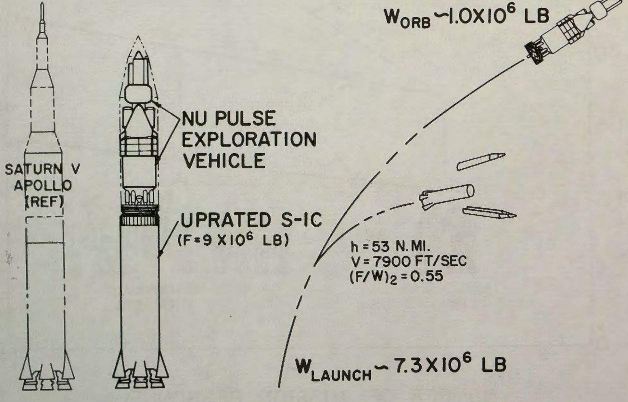

Fig. 6—Single-Saturn launch capability of the Mars exploration vehicle

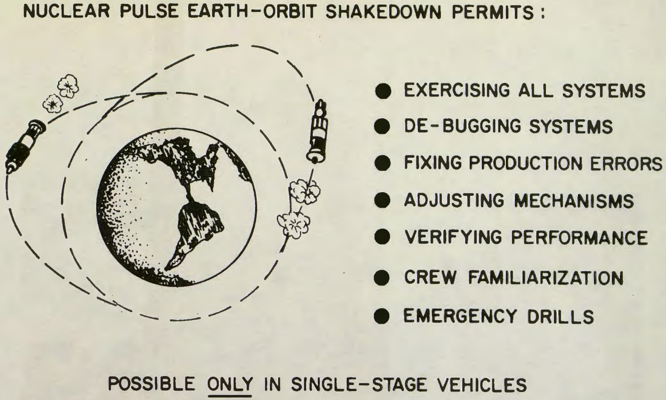

Fig. 7—Operational benefits of a pre-departure earth-orbit shakedown

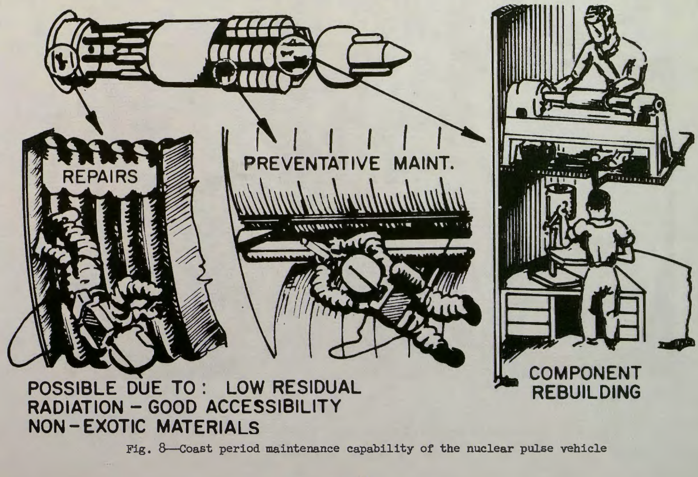

Fig. 8—Coast period maintenance capability of the nuclear pulse vehicle

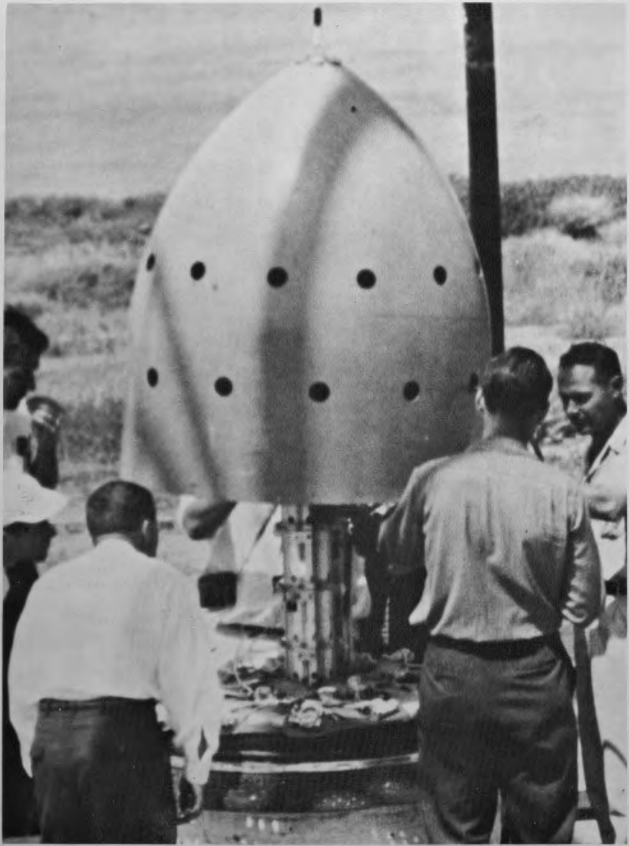

Fig. 9—High-explosive-propelled pulse vehicle model first flown in October 1959

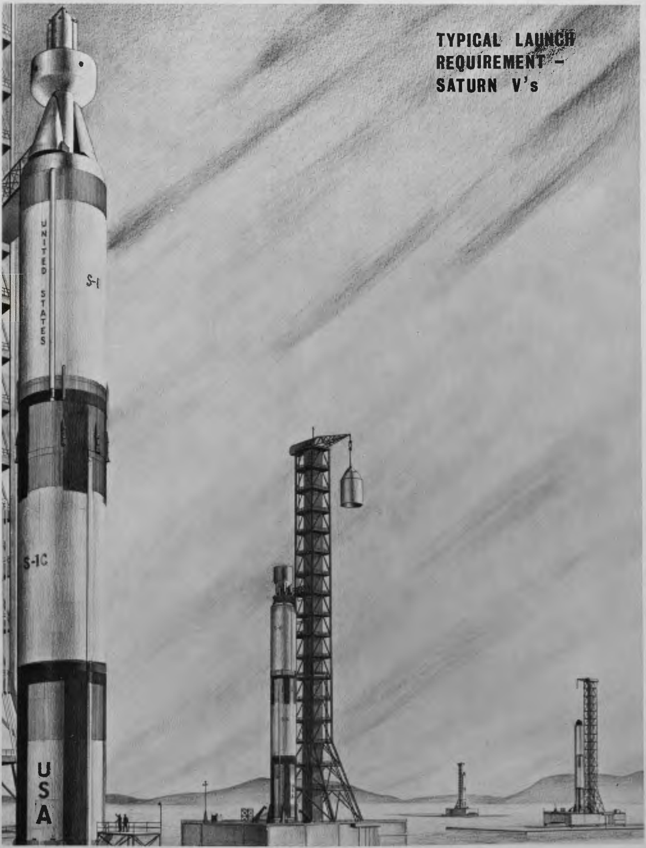

Fig. 10—Launch requirement for typical Mars surface excursion mission using the Saturn V earth launch vehicle

Fig. 11—Fully loaded nuclear pulse vehicle in earth orbit shakedown cruise prior to departure

Fig. 12—Enroute maintenance and ejection of empty propellant magazines two days after earth departure

Fig. 13—Mars excursion module final checkout and operations while in Mars orbit

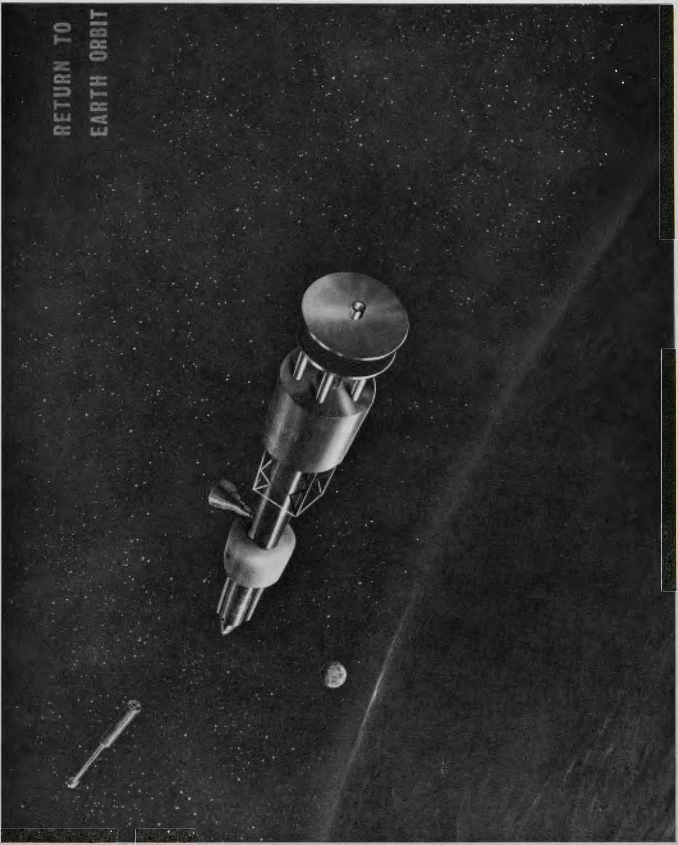

Fig. 14—Return to earth orbit and rendezvous with reentry vehicle at conclusion of Mars trip