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Mars Ascent-Stage Design Utilizing Nuclear Propulsion

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The majority of NASA manned Mars exploration studies undertaken over the past three decades have advocated high-performance nuclear thermal propulsion systems as a means of reducing spacecraft mass and ameliorating the high Earth-to-orbit launch costs associated with these missions. In these studies the use of nuclear propulsion has been uniformly limited to the Earth–Mars interplanetary transfer maneuvers, however. Recent interest in smaller, higher-thrust-to-weight engines coupled with advances in nuclear fuel material capability have prompted consideration of these engines for other applications. This investigation presents an evaluation of the utilization of a small nuclear propulsion system for the Mars ascent burn. The performance of a nuclear ascent-stage concept is compared with several chemical propulsion concepts, including those benefiting from Mars atmospheric in situ propellant production. Propellant requirements, boiloff losses, mission delta velocities, and other parameters relevant to a piloted Mars mission are discussed. Also, a unique low-initial-mass mission trajectory suited for Mars base crew rotation, especially benefited by ascent systems utilizing high-specific-impulse nuclear systems or in situ propellant production, is described.

Nomenclature

I_{sp} = specific impulse, s
T/W = thrust-to-weight ratio
V_e = Mars escape velocity, m/s
V_{hp} = hyperbolic velocity at swingby, m/s
$V_{surface}$ = rotational velocity at liftoff site, m/s
ΔV = delta velocity, m/s

Introduction

NUCLEAR thermal propulsion (NTP) has been selected in many Mars mission studies as the propulsion system of choice for the Earth–Mars interplanetary transfer vehicle because of its high specific impulse (I_{sp}), which is double that of the best chemical propulsion systems. Large, high-thrust engines, designed to operate for several hours, have been envisioned. Engines in the 333- to 1000-kN (75- to 225-klbf) thrust class were developed and tested in the 1960s as part of the DOE- and NASA-sponsored Nuclear Engine for Rocket Vehicle Application (NERVA) test program.[1] Interest in engines in the smaller, 90- to 130-kN (20- to 30-klbf) thrust class and recent development work on high-thrust-to-weight nuclear engine concepts under the United States Air Force Space Nuclear Thermal Propulsion (SNTP) program[2] have prompted consideration of nuclear propulsion systems beyond their usual conceptual application to Mars interplanetary transfer. This investigation assesses the suitability of a small nuclear system for the Mars ascent stage. The performance of this stage is compared with several chemical propulsion concepts, including those benefiting from in situ atmospheric resources at Mars.

The utilization of in situ propellant production (ISPP) for reducing the total payload mass required in Earth orbit has been advocated for Mars exploration missions since the 1970s,[3] and has recently been proposed in a NASA Mars mission study.[4] In that ISPP scenario, liquid oxygen (O_2) propellant processed out of Martian atmospheric carbon dioxide (CO_2), and liquid methane (CH_4) propellant formed by combining the carbon extracted from the atmospheric CO_2 separation process with H_2 carried with the vehicle, would be supplied to an ascent stage on the surface. This vehicle would be deployed a year or two previous to the arrival of a crew on a separate lander stage. Once fueled autonomously on the surface, the vehicle is to be used to return the crew after their surface stay to Mars orbit for rendezvous with a Mars–Earth transfer stage. Since no ascent-stage propellant need be transported from the Earth, the total payload required of the Earth-to-orbit (ETO) and Earth–Mars transfer systems can be reduced. Other variations on this scheme have been considered, including those in which only O_2 is produced at Mars as oxidizer to supply vehicles utilizing Earth-supplied hydrogen (H_2), monomethyl hydrazine (MMH), or other fuel. The two factors most influential in determining propellant requirements and thus vehicle mass estimates are the vehicle's trajectory ΔV requirements and its propulsion system efficiency.

Ascent Delta Velocity

Missions to Mars are very energy-intensive. Maneuvers with large delta velocity (ΔV) are required of the spacecraft's propulsion systems. Listed in Fig. 1 is a ΔV set (in meters per second) required for a typical short-stay-time (30 days on the surface) Mars mission trajectory. Reductions in vehicle initial mass in low earth orbit (IMLEO) can be achieved by reducing the mission ΔV requirement, which is primarily a function of opportunity year, trip time, and stay time, or by increasing the propulsion-system I_{sp}, or by using combinations of aerocapture and propulsion. Elimination of the Earth return capture ΔV (with expenditure of the transfer vehicle, and crew return via a small re-entry vehicle) is typically the largest IMLEO reduction that can be made. Low-energy, long-stay-time trajectories (2 years on the surface) can be utilized to minimize the ΔV of the remaining planetary transfer burns (Earth departure, Mars capture, and Mars departure). Lengthening trip times to minimum-energy Hohmann transfers represents about the most that can be done to reduce the transfer ΔV, though crew concerns may limit trip times to some duration less than that which corresponds to the lowest energy. After minimization of mission ΔV, further mass savings can be achieved by the application of advanced (high I_{sp}) propulsion; NTP is typically chosen for the planetary transfer ΔV. Mars orbit aerocapture is also often cited as an effective means of reducing vehicle mass by eliminating the propellant load necessary for propulsive capture. [However, given that adequate propulsive capability for orbit capture exists on the vehicle already, assessing the actual cost-effectiveness of incorporating another major system (planetary capture aerobrake) into the transfer vehicle flying a low-energy trajectory would require the evaluation of several factors, including the extent of the mass saving, the packageability of the propulsion and habitat systems into the aerobrake, and on-orbit aerobrake assembly complexity.]

Once in Mars orbit, the crew and surface payload is emplaced on the surface by one or more landing craft. All but about 1000 m/s of the descent maneuver ΔV can be taken out aerodynamically by the

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Table 1 Engine performance characteristics

Propellant	Chamber Max. temp, K	Chamber Pressure, atm	Nozzle, expansion ratio	Propellant mol wt, amu	Efficiency η	Specific impulse, s
O_2–MMH	3419	68	200	64	0.94	372
O_2–CH_4	3560	68	200	48	0.94	371
O_2–H_2	3250	68	200	34	0.98	470
Nuclear, H_2	3100	68	200	2	0.98	1000

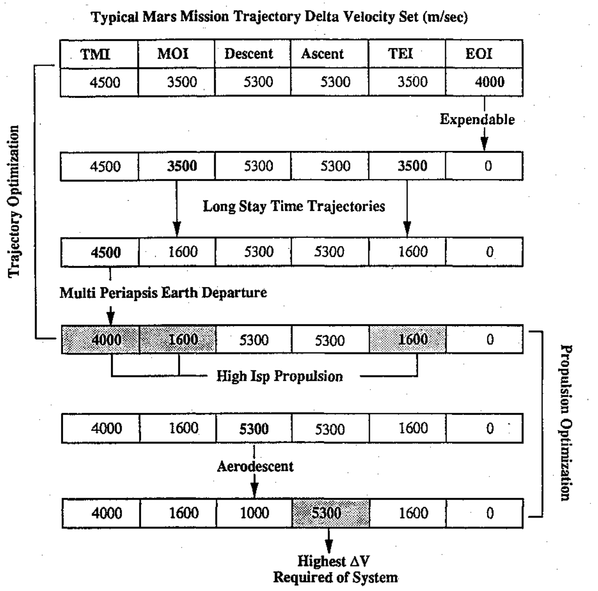

Fig. 1 Mars mission delta velocities.

landing aerobrake. The only large-ΔV maneuver remaining to be considered is the Mars ascent ΔV; it is the most energy-intensive of all the maneuvers, requiring 5300 m/s from the ascent-stage propulsion system to ascend to a 250-km periapsis by 24-h-period parking orbit necessary for rendezvous with a orbiting transfer stage. [This ΔV value exceeds the ideal escape velocity (5000 m/s) for Mars, because of the excess velocity needed to offset gravitational and atmospheric drag losses.] Such a high energy requirement invites the application of efficient propulsion systems.

Rocket Propulsion Performance

Rocket propulsion systems are thermally driven gasdynamic devices whose I_{sp} is inversely proportional to the square root of the molecular weight of the exhaust gases and directly proportional to the engine chamber temperature. NTP systems, which can operate with low-molecular-weight exhaust gases (such as hydrogen, which has a molecular weight of 2), have an inherently high I_{sp}. Chemical propulsion is limited to combustion products of inherently higher molecular weight and thus lower I_{sp} (see Table 1 for chemical-system I_{sp} values, which are taken from Ref. 5). NTP systems are similar to conventional liquid propulsion systems in several ways. An important difference is that the nuclear reactor, which acts as a heat source, replaces the combustion chamber of the chemical engine, and the fuel–oxidizer combination is replaced with a single fluid. Ignition and combustion of a bipropellant mixture is unnecessary, and the need for dual tankage systems and associated plumbing is eliminated.

In a NTP engine, the reactor heats the propellant directly as the propellant flows through cooling passages in the core. Fuel elements transfer heat directly to flowing hydrogen, which is expanded out of the engine through a nozzle to produce thrust. The engine I_{sp} is limited by the maximum temperature that these elements can withstand. Temperature endurance levels in excess of those attained in the late 1960s during the NERVA test programs have been attained in recent tests. These gains have come primarily through the development of high-temperature-resistant combinations of uranium, zirconium, tantalum, and carbide materials. In the NERVA tests, maximum operating temperatures of 2750 K were achieved in 1968.[6] In the 1980s, reactor exhaust gas temperatures of 3100 K were demonstrated for over one hour of operational time in Russian test reactors stationed near Semipalatinsk, Kazakhstan.[7,8] At the Lutch Research Institute, in Podolsk, Russia, in 1993 and 1994, fuel elements were exposed to flowing hydrogen in excess of 3100 K in several nonnuclear hot-flow tests,[9] including operation[10] at 3300 K for 2 h. These results substantiate independent projections made by the Space Nuclear Thermal Propulsion project team[11] and the NASA Nuclear Propulsion Technology Interagency Panel.[12] At an exhaust gas temperature of 3100 K, a chamber pressure of 68 atm (1000 psia), and a nozzle expansion ratio of 200:1, a nuclear engine would produce[13] an I_{sp} slightly above 1000 s.

NTP fuel-element temperatures have in the past been limited to levels commensurate with the burn time (several hours) expected of Earth–Mars transfer-stage engines. This is necessary because element longevity is dependent on both the operational temperature and the duration of operation at that temperature.[14] After long periods at high temperatures, the fuel elements begin to degrade by corrosion in the presence of the hot flowing gaseous hydrogen, with a consequential loss in performance. The duration of a Mars ascent burn, however, would be less than 15 min; for this period a higher operating temperature could be sustained then would be possible for the burn time required for the Earth departure maneuver of the interplanetary vehicle, for example, because the fuel-element degradation process would have little time to develop or progress. Since 3300 K for 2 h has been demonstrated in fuel-element tests, operation 200 K cooler at 3100 K should represent sufficient conservation in a flight engine designed for a short 15-min burn. For this reason a relatively high I_{sp} of 1000 s was utilized in the analysis.

Another factor affecting vehicle performance is engine T/W ratio. Nuclear engines developed during the NERVA program had low engine T/W ratios compared to conventional liquid propulsion systems. Improvements in T/W beyond the ratios achieved in the 1960s are projected in several NASA and DOD studies; T/W estimates of 30 : 1 have been proposed[15,16] for a 330-kN (75-klbf) engine based on the particle-bed-reactor (PBR) concept[17,18] designed by Brookhaven National Laboratory for the SNTP program. The PBR, designed for high T/W, is an extremely compact and efficient nuclear reactor, maximizing the surface-area-to-volume ratio of the fuel. In its simplest form, the PBR consists of a packed bed of very small spherical fuel particles, through which coolant is pumped. The SNTP program's stated goals were $I_{sp} = 1000$ s and $T/W =$ 25–35:1 (depending on the thrust level) for the flight engine. For small engines in the 90-kN (20-klbf) thrust range (of interest to this study), a specific PBR engine having $T/W = 20$:1 has been proposed.[19] However, a more conservative T/W of 10:1 was utilized in this investigation's nuclear-stage calculations. It should be noted that the magnitude of T/W improvements achievable through PBR technology is still a matter of debate, and a 10:1 estimate would be considered optimistic by some,[20,21] especially those who advocate use of NERVA derivative technology. A PBR engine is illustrated in Fig. 2.

For the conventional liquid-chemical-propulsion ISPP vehicles, estimated production-plant and power-system masses of 500 and 1500 kg were taken from Ref. 22. These elements are carried on the descent stage to the surface for ascent-stage fueling. Pumps acquire CO_2 gas from the Martian atmosphere, liquefy it, and through a series of reactions produce CH_4 and O_2 (Ref. 23). Hydrogen, brought

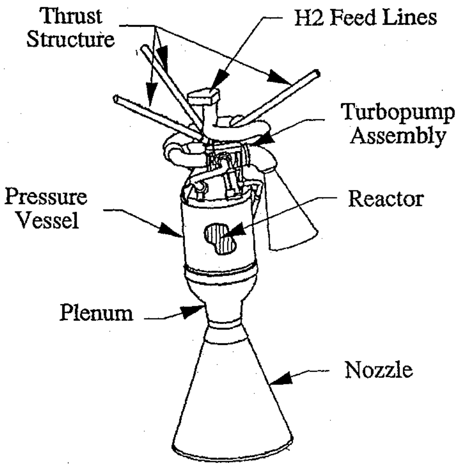

Fig. 2 Nuclear thermal engine.

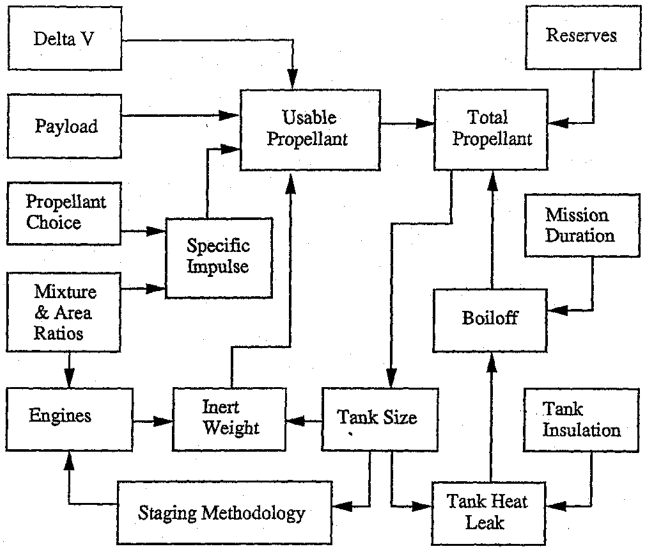

Fig. 3 Vehicle-sizing iteration loop.

from Earth, must be supplied for the reaction process at a ratio of one part H_2 for every four parts of CH_4 produced. For the ISPP O_2–H_2 and O_2–MMH options, only O_2 is processed from the CO_2 atmosphere (the H_2 or MMH fuel is, in these cases, Earth supplied). A descent stage sized for each ascent concept utilizes an O_2–H_2 propulsion system that provides terminal descent and touchdown after aerodynamic braking has reduced the vehicle velocity to 1000 m/s. These propellant options bound the range of plausible Mars ascent concepts.

Ascent-Stage Mass Estimation

Boeing's Mars lander spacecraft design model was utilized to estimate ascent- and descent-stage performance and mass as a function of a variety of variables related to the mission, vehicle, and subsystems. System interrelationships and functional dependencies (Fig. 3) too extensive for hand calculation are mathematically modeled and integrated to provide a detailed parametric study tool. Seven different ascent concepts were evaluated, of which four use only Earth-supplied propellants and three use ISPP. Included in the first group are the nuclear H_2 and the chemical O_2–H_2, O_2–CH_4 and O_2–MMH vehicles. In the latter group are the O_2–H_2 and O_2–MMH vehicles, which benefit from ISPP O_2 oxidizer, and the O_2–CH_4 vehicle, which utilizes ISPP oxidizer and fuel. Assumptions used in the weights analysis are listed below.

General Assumptions

1) Mass[24] of six-man crew cabin is 4.3 tons.
2) Ascent-vehicle $T/W = 1.5$ at Mars liftoff.
3) ΔV = 5300-m/s for the ascent burn to a 250-km, 24.6-h Mars parking orbit.
4) Aeroshell mass is set to 15% of the total decelerated mass.
5) Descent engines provide a terminal descent and hover ΔV of 1000 m/s.
6) A 160-day Earth–Mars outbound transfer time and a 500-day surface-stay time are used for propellant boiloff calculations.
7) Mars surface propellant boiloff rates of 1.20, 0.32, and 0.28% per month[25] are used for calculating boiloff for the cryogenic H_2, O_2, and CH_4 propellants, respectively.
8) Weight growth of 15% is used for stage dry-mass calculations.

Nuclear Vehicle Assumptions

1) A single nuclear engine is utilized with a 1.2-ton (1 ton = 1000 kg) dedicated radiation-shield mass.
2) The in-line primary H_2 tank is configured to act as a supplemental shield for radiation attenuation.
3) The Engine is jettisoned before rendezvous with the orbiting transfer vehicle; it remains in a long-lived Mars orbit.

ISPP Vehicle Assumptions

1) A propellant plant and power generation system of 2000-kg mass is delivered with the ascent stage. This mass is counted as part of the descent stage in the weight calculations. Ascent tanks are topped off prior to liftoff to negate boiloff.
2) Ascent- and descent-stage load-bearing structure (frame and landing legs) are sized to support the vehicle when fully fueled on the surface.

Nominal Performance Comparison

Vehicle mass statements for the seven different ascent concepts are listed in Table 2. These data are shown graphically in Fig. 4. In Fig. 5, ascent-stage mass and required thrust at Mars liftoff are plotted against variations in engine T/W. As can be seen in Table 2, the high-I_{sp}, high-T/W nuclear system enables a sizable mass reduction in comparison with the non-ISPP chemical systems. At 17.8 tons it provides a 37% reduction in mass from the chemical O_2–H_2 vehicle (28.3 tons), the concept typically proposed in Mars mission studies.

ISPP also offers significant mass savings with respect to the Earth-supplied options when ascent-stage comparisons are made at the "at Earth departure" mass level (Table 2, column 10). The ISPP oxidizer-supplied O_2–H_2 craft is 16.9 tons lighter than its counterpart with all Earth-supplied propellant (11.4 vs 28.3 tons). Likewise, production of both oxidizer and fuel on the surface would allow the O_2–CH_4 vehicle to be transported to Mars with its propellant tanks empty, at one-fifth the mass of its fully fueled Earth-supplied counterpart. At 10.5 tons, this all-ISPP ascent concept proved to be the lightest of all those evaluated. After fueling on the surface, however, the ISPP O_2–CH_4 craft becomes the heaviest of those evaluated, at 48.9 tons, nearly quadrupling its empty delivered mass (38 tons of this amount is propellant). The load-bearing frame and landing-leg structure of the descent stage needed to support this amount is heavy, even in the reduced gravity environment on Mars. This additional descent-stage structural mass (along with the additional ISPP hardware and H_2 seed propellant that must be carried to the surface) is enough to change the rankings of the weight comparison when the concepts are assessed at the level of the total lander mass (Table 2, last column), which is the most relevant comparison. The descent stage necessary for the ISPP O_2–CH_4 ascent stage is 11 tons heavier than that required for the nuclear stage (26.0 vs 15.1 tons). Consequently, the total lander mass associated with the nuclear ascent system is less than that of the ISPP O_2–CH_4 system, even though its ascent-stage delivered mass is 7 tons larger. At 27.9 tons, the total lander associated with the ascent vehicle with in situ O_2 and Earth-supplied H_2 is the lightest overall.

Minimum mass, however, cannot be treated as the only criterion for evaluation; operational differences must also be assessed. In this regard there exists a notable difference between the Earth-supplied and the ISPP vehicle.[5] The former do not require the additional complexity associated with autonomous vehicle predeployment and operation on the Martian surface previous to crew arrival. Also, the non-ISPP landers have the capability to effect an abort during descent if necessary. In such an event the ascent stage can separate

Table 2 Mars ascent-stage and descent-stage weight statements[a]

	Ascent stage										Descent stage							
	Dry mass			Propellant					Total		Dry mass				Propellant			
				Earth-supplied			Mars supplied								Earth-supplied			
	Crew cabin	Propulsion	Tankage, structure, and RCS	O_2	Fuel	Boiloff	O_2	Fuel	At Earth departure	At Mars liftoff	Tankage, structure, RCS and landing legs[b]	Propulsion	In situ plant and power	Aerobrake	O_2	H_2	Total	Total lander mass
O_2–MMH	4.3	0.8	4.8	21.6	13.5	1.2	n/a	n/a	46.2	45.1	8.4	2.2	0.0	12.6	12.2	2.1	37.5	83.7
In situ O_2–MMH	4.3	0.8	4.8	n/a	13.5	0.0	21.6	n/a	23.4	45.1	7.9[b]	1.3	2.0	7.6	7.4	1.3	27.5	50.9
O_2/H_2	4.3	0.5	3.4	16.0	2.6	1.5	n/a	n/a	28.3	26.8	5.1	1.3	n/a	7.6	7.4	1.3	22.7	51.0
In situ O_2–H_2	4.3	0.5	3.4	n/a	2.6	0.6	16.0	n/a	11.4	26.8	4.8[b]	0.7	2.0	4.2	4.1	0.7	16.5	27.9
O_2–CH_4	4.3	0.8	5.5	29.8	8.5	2.1	n/a	n/a	51.0	48.9	9.2	2.4	n/a	13.8	13.4	2.3	41.1	92.1
In situ O_2–CH_4	4.3	0.8	5.5	n/a	n/a 2.2[c]	n/a	29.8	8.5	10.6	48.9	9.2[b]	0.9	2.0	5.5	5.3	0.9	26.0	36.6
Nuclear, H_2	4.3	2.2[d]	3.0	n/a	6.6	1.7	n/a	n/a	17.8	16.1	3.8	0.8	n/a	4.9	4.8	0.8	15.1	32.9

[a]All masses are in kilograms.
[b]Descent-stage structure and landing-leg mass is a function of the load supported, which includes the ascent stage when it is fully fueled.
[c]Seed H_2 taken for CH_4 production counted as descent payload.
[d]With shield.

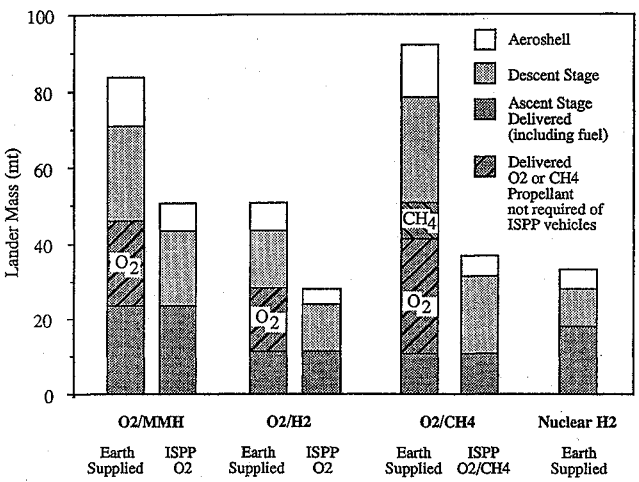

Fig. 4 Mars ascent- and descent-stage mass comparisons.

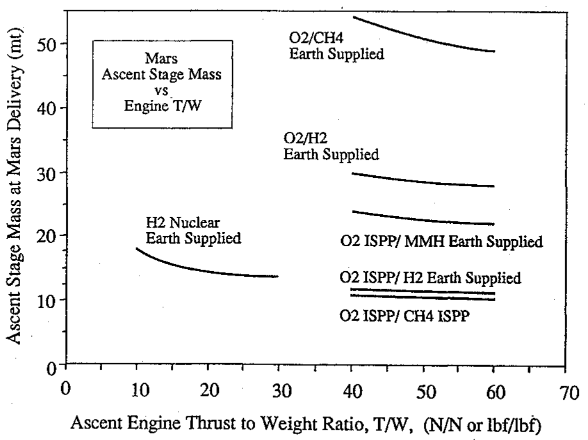

Fig. 5 Ascent-stage mass vs engine thrust-to-weight ratio.

from the descent stage and burn its propellant to return to Mars orbit. The ISPP vehicles are delivered to Mars with their ascent tanks empty and inherently have not this capability. However, the ISPP strategy does offer a secondary benefit for scenarios in which extensive use is made of surface rovers. If an ISPP system is emplaced and operational, additional propellant beyond that required for the ascent craft can be used for powering rovers and other surface base vehicles. Overall, the nuclear concept both is less massive and presents fewer credible failure modes than the O_2–CH_4 ISSP concept recently advocated.

The Nuclear Engine

The nuclear engine is basically a heat exchanger and not a combustion device. Mixing of oxidizer and fuel in an injector is unnecessary, and neither ignition nor combustion takes place within the reactor. Typically, propulsion-system reliability is achieved though engine redundancy; contemporary man-rated vehicle concepts are in most cases designed with an engine-out margin. The relative operational simplicity of the nuclear engine allows for an exception in this case, however. Though not a nuclear stage, the Apollo Lunar Module ascent stage can be cited as an example of a single-engine manned ascent stage. High confidence in this propulsion system was achieved through simplicity: no igniter was used (hypergolic, or self-igniting, fuels were used), and no engine throttling, gimbaling, or restart was required. These simplifications reduced the number of credible failure modes and justified the choice of a single engine. Likewise, a single-engine nuclear system would require no igniters, nor would it have a throttling, gimbaling, or restart requirement. Unlike the Apollo system, however, the nuclear system would require a hydrogen turbopump, which represents an additional failure mode. The utilization of dual turbopumps, however, would allow continued operation in the event of a single pump failure. Rocketdyne designed, built, and tested a dual turbopump system for the NERVA series of nuclear engines in 1967.[26]

Ascent-Stage Design Considerations

A single engine of 90-kN (20-klbf) thrust is utilized for the nuclear concept. Its total propellant volume is 115 m^3, which is large in comparison with the chemical O_2–CH_4 system because of the low density of the H_2. The vehicle is illustrated in Fig. 6, and major subsystems are labeled from 1 to 7 on the cutaway view. These elements are 1) the ascent crew cabin, 2) the primary ascent-stage H_2 propellant tank, 3) the ascent-stage structure, 4) the descent-stage propellant tanks, 5) the descent-stage engines, 6) the ascent-stage secondary (conformal) tank, and 7) the ascent-stage nuclear engine.

Vehicle design analysis included consideration of the unique issues associated with the use of nuclear propulsion, including radiation shielding and engine disposal. A NTP system emits about 1% of its energy in gamma and neutron radiation through the reactor pressure vessel.[27] This high-radiation environment in the vicinity of the reactor may produce potentially lethal radiation doses in the vehicle crew compartment, may possibly damage sensitive components, and heats the surrounding structure and propellant. The addition of a radiation shield between the reactor and the tank bottom attenuates the energy disposition into the vehicle. The thickness of the shield for a given energy attenuation is an function of reactor power and also an inverse function of separation distance. The cumulative shielding capability provided by the dedicated external shield, the internal engine forward support plate, the propellant, and other spacecraft and crew cabin hardware mass serves to keep the crew from receiving radiation exposure beyond acceptable limits. Because liquid H_2 has

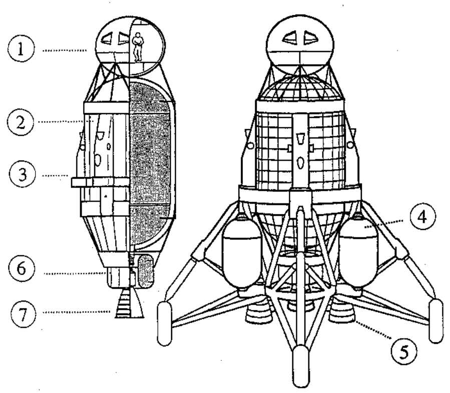

Fig. 6 Nuclear ascent-stage and descent-stage.

good neutron absorption capabilities, the design takes advantage of positioning the 6.6 tons of propellant to supplement the direct-line shielding ability of the radiation shield. Since stage mass is an important figure of merit, an overdesigned shield is not acceptable. Its design must take into account the supplemental shielding ability of the other attenuating elements and the overall geometries of the system, including ducts, voids, and other features.

Radiation-shield effectiveness calculations are quite complicated, and require an extensive accounting of a variety of geometrical elements that can contribute to attenuation or exacerbate the problem through the generation of secondary sources of radiation.[28] A radiation assessment was not done for this study; the estimate of 1.2 tons of shield mass used in the analysis was scaled (based on power level) from a shielding study[29,30] done for a larger class of engines. Previous to the ascent burn, the engine contains no radioactive fission products. Over the course of the 13-min burn time about 5 g of fission products would have accumulated in the reactor; this is roughly $\frac{1}{100000}$ of the amount generated by a typical 3000-MW terrestrial nuclear power plant in one year.

For the short duration the crew would spend ascending through the tenuous Mars atmosphere, which has a surface pressure $\frac{1}{100}$ that of Earth, scattering of radiation is a concern, and a secondary H_2 propellant tank positioned around the engine is used to inhibit atmospheric backscatter radiation to the crew. The conformal tank is fitted around the engine so that the reactor is enclosed in all except the aft (nozzle) direction. During the brief low-altitude atmospheric portion of flight, H_2 is provided to the engine from the larger primary tank while the conformal tank remains full; once above the atmosphere, the smaller tank is utilized until emptied, as H_2 supply is again drawn from the main, in-line tank. Actual atmospheric radiation backscatter fluxes were not calculated. Near the point when the main tank is emptied, the engine is jettisoned and rendezvous with the orbiting transfer stage is done with bipropellant reaction control-system thrusters. Engine cooldown propellant is not required.

The total propellant volumes required for the O_2–H_2, O_2–CH_4, and O_2–MMH ascent systems are 62, 41, and 31 m^3, respectively. Propellant tank fractions (tank mass divided by total tank and propellant mass) for the O_2 (4%), MMH (4%), and CH_4 (6%) tanks are significantly less than for the large primary (15%) and the smaller conformal (20%) H_2 tanks on the nuclear stage. The chemical vehicles utilize two engines to provide an engine-out margin, and each gimbals to track the vehicle center of gravity. The descent stage utilized with all concepts uses O_2–H_2 engines positioned under side-mounted descent tanks. This and other lander configuration types are discussed in detail in Ref. 31.

Mars-Base Crew Rotation Scenarios

Low-cost access is the key to a continual human presence at Mars. A special Earth–Mars–Earth trajectory called the dash–flyby mode offers a unique advantage for short stay time, Mars-base resupply, and crew rotation scenarios. The dash–flyby mission provides the lowest IMLEO of any Mars mission type for any given transfer-vehicle propulsion-system I_{sp} and payload,[32–34] although it is suitable for short surface stay times only. The trajectory is illustrated in Fig. 7 and is briefly described as follows: After Earth departure and near the end of the outbound Earth–Mars journey, as the transfer-stage–lander-stage combination approaches Mars, the landing craft separates from the transfer-stage and executes a short "dash" or acceleration, sufficient for an early arrival at Mars, a few weeks before the transfer stage is to reach its Mars swingby point on its fixed Earth return trajectory. This period would be long enough for a short surface exploration mission, or in the case of a crew rotation mission, allow the newly arrived crew to become accustomed to the base habitat system before the departure of the outgoing crew. The required ΔV for the lander to accelerate for early arrival depends upon the time of separation and the lead time that is desired. It is typically less than 500 m/s for stays of 20 days or less. An alternative to lander-stage acceleration, which achieves the same end, is transfer-stage deceleration. In this case, after vehicle separation, the transfer-stage engines provide thrust for deceleration in order to effect a later Mars arrival and swingby (Fig. 8). Returning crew board the ascent stage for departure; its final velocity must exceed

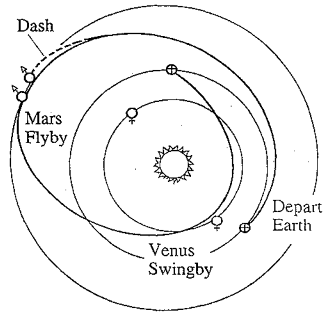

Fig. 7 Dash–flyby mission trajectory profile.

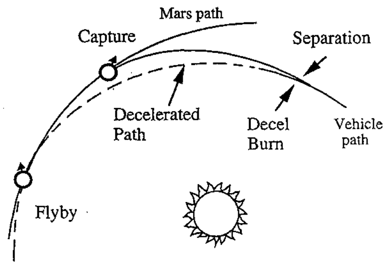

Fig. 8 Dash–flyby mission transfer-stage deceleration option.

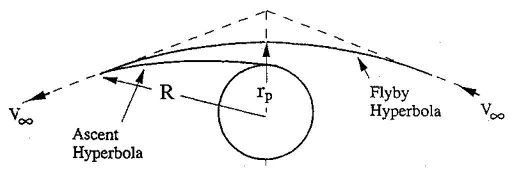

Fig. 9 Dash–flyby ascent to rendezvous.

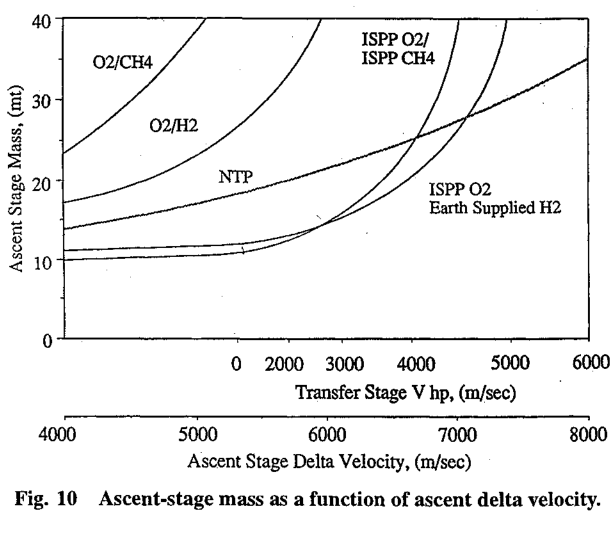

Fig. 10 Ascent-stage mass as a function of ascent delta velocity.

Table 3 Mass statement: Mars crew rotation vehicle for 2020 mission

	Transfer propellant and tankage			Core transfer vehicle					
	Trans-Mars injection	Mars early arrival	Powered Mars swingby	Propulsion & shielding	Structure & RCS	Transfer crew habitat	Crew return capsule	Lander (direct entry at Mars)	Total
ΔV, m/s	3174	278	1612						
Mass, tons	120.6	4.5	23.3	4.4	13.4	56.0	5.8	42.0	270.0

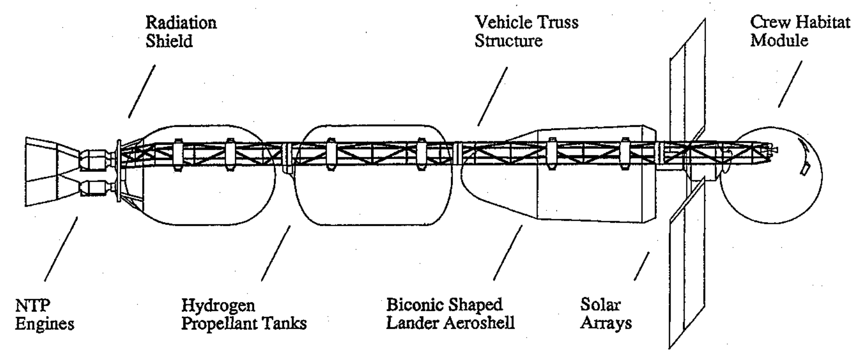

Fig. 11 Transfer vehicle for Mars dash–flyby crew rotation mission.

Mars escape velocity as the vehicle effects a hyperbolic departure for rendezvous with the transfer stage at a point several Mars radii past the swingby periapsis encounter point. Rendezvous occurs as both the transfer and ascent stages recede from the planet. Transfer and ascent flight paths at rendezvous are illustrated in Fig. 9.

Because the transfer stage does not capture into Mars orbit, its total IMLEO can be much less than that of a equivalent Mars orbit capture mission vehicle delivering the same payload. In the dash–flyby mode only one major burn is required of the transfer vehicle (Earth departure), after which the vehicle follows a prescribed Mars flyby path, taking advantage of a gravity-assisted turn at Mars to return to Earth with the aid of an additional propulsive ΔV of 800 to 2000 m/s to supplement the gravity-assisted swingby in most opportunity years. An unpowered Venus swingby is utilized on return. The large momentum changes (vehicle mass times velocity change) usually required of the transfer system for Mars capture and departure are eliminated, and reductions in IMLEOs of 40% are possible in comparison with standard stopover mission vehicles.

The principle disadvantage of the dash–flyby mission is the increased ascent ΔV necessary for a hyperbolic rendezvous with the transfer stage as it swings by Mars. Ascent ΔV values (which depend upon transfer-stage hyperbolic velocity V_{hp} at Mars) on the order of 6000 to 8000 m/s are necessary, as compared to the 5300 m/s normally required for a subparabolic ascent to orbit. The required ascent ΔV can be calculated as follows:

$$\Delta V = \left(V_e^2 + V_{hp}^2\right)^{0.5} - V_{\mathrm{surface}} \tag{1}$$

For an optimized 2020 dash–flyby trajectory, the transfer-stage V_{hp} at Mars swingby is 3300 m/s; the ascent ΔV required for escape and rendezvous would be 6300 m/s, or about 1000 m/s beyond that required for ascent to orbit. With the 1000-s I_{sp} available from a nuclear engine, this added ΔV would translate into an increase of only 6 tons (to 22.5 tons) over the 17.8-ton subparabolic ascent stage (lander mass increases from 32.9 to 42.0 tons). For Earth-supplied chemical propulsion systems, the ascent mass increases more rapidly with ΔV than for the nuclear stage on account of the I_{sp} difference, and therefore these systems are much less efficient at matching transfer-stage swingby velocities. Ascent-stage mass is plotted against transfer-stage swingby V_{hp} in Fig. 10. In exchange for the ascent-stage mass increase, the dash–flyby mode eliminates the Mars capture and departure maneuvers, which greatly reduces the transfer-stage propellant requirement. Total Mars crew rotation mission IMLEOs of nearly 300 tons are available for most opportunities. For an optimized 2020 mission, a IMLEO of 270 tons would be possible utilizing a NTP Mars transfer vehicle with $I_{sp} = 925$ s.

A transfer-stage design is illustrated in Fig. 11, in which the ascent stage is shown packaged inside a conically shaped aeroshell. A corresponding mass statement is given in Table 3 along with ΔV requirements for the trans-Mars injection burn, the lander early Mars arrival maneuver, and the powered swingby burn at Mars.

Conclusions

A nuclear propulsion stage has been applied to the Mars ascent burn. Recent work in this area indicates that one can obtain high engine performance (both T/W and I_{sp}) from a small, 90-kN-thrust-class engine for the short burn time required of a Mars ascent maneuver. For a vehicle capable of ascending to a 250-km, 24-h-period orbit, the 17.8-ton nuclear ascent stage provides a sizable reduction in mass from that of a chemical O_2–H_2 vehicle of the type typically utilized in Mars mission studies. An all-ISPP O_2–CH_4 ascent stage provides the lowest delivered mass of the concepts considered. However, other factors raise its descent-stage mass to a level such that the total lander mass associated with the ISPP O_2–CH_4 ascent stage is heavier then the lander mass associated with the non-ISPP nuclear ascent system. Furthermore, the nuclear stage does not require autonomous predeployment and fueling at Mars previous to crew arrival, and does not sacrifice descent abort capability for the crew, as do all ISPP crew delivery concepts. This single-tank, single-propellant system requires no ignition or combustion, and without a throttling, gimbaling, or engine restart requirement could be made highly reliable. The complication imposed by its radiation source can be countered by shielding, which involves the addition of no moving parts. The nuclear concept is both less massive and presents fewer credible failure modes than the O_2–CH_4 ISPP system.

The dash–flyby strategy would enable mission IMLEO savings for Mars base crew rotation of up to 40% in comparison with that of the standard stopover trajectories. Risks associated with this mission primarily involve the ascent maneuver. The ascent stage must exceed Mars escape velocity to rendezvous with the transfer vehicle as it recedes from the planet. The high I_{sp} of the nuclear stage makes it optimal for this unique mission mode.

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