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Advanced Propulsion for Future Planetary Spacecraft

P.W. Garrison*
Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California

Advanced propulsion concepts are evaluated for unmanned exploration class missions to the outer planets. Spacecraft propulsion requirements for these missions are compared with those for previous missions. A major improvement in performance above that offered by current systems is needed to deliver the payloads required by these missions in an acceptable trip time. Advanced pump-fed space-storable and cryogenic propulsion systems are evaluated. Nuclear propulsion options considered include the solid core, particle bed, gaseous core, nuclear pulse, and nuclear electric concepts. The results of this study reaffirm the superiority of nuclear electric propulsion for this mission category.

Introduction

TABLE 1 presents a summary of the spacecraft propulsion requirements for representative planetary missions conducted to date and those planned for the present decade. Spacecraft velocity requirements for these missions range between 0.05 and 5 km/s and flight times range between 0.3 and 4 yr. Primary propulsion requirements for the low velocity missions have been met by monopropellant systems. Pressure-fed Earth-storable bipropellant propulsion systems have been used and are proposed for the higher velocity missions. The Viking N_2O_4/MMH orbiter propulsion system is typical of this category. The Viking orbiter engine produced a thrust of 1334 N at a specific impulse of 291 s. The attitude control system for this spacecraft was a cold gas nitrogen system. The Viking orbiter propellant was depleted in July of 1980, approximately 2 yr beyond the completion of the design mission life of 3 yr.

The Galileo mission planned for a 1985 launch will place an 850-kg spacecraft payload in orbit about Jupiter after deploying a 340-kg atmospheric probe. A pressure-fed N_2O_4/MMH propulsion system consisting of a 400-N main engine and 10-N attitude control engines will be employed.

The VOIR mission originally scheduled for launch in 1988 will place a 1200-1400 kg payload in orbit about Venus. The relatively large velocity increment (4.3 km/s) required by this mission can be achieved by a combination of aerobraking and solid and liquid Earth-storable propulsion systems.

Unmanned spacecraft can be divided into three classes; reconnaissance, exploration, and laboratory spacecraft. The Voyager spacecraft is a reconnaissance class (200-800 kg) spacecraft characterized by a flyby encounter and limited science. Galileo is a exploration class (750-1500 kg) spacecraft consisting of an orbiter and an atmospheric probe. The Viking mission is representative of the laboratory class (>1500 kg) mission providing both orbiter and lander spacecraft. Future laboratory class spacecraft will feature orbiters with high power radar, laser and radio systems, and active refrigeration systems for sensor cooling and propellant thermal control. Mobile landers with sophisticated laboratories and sample return vehicles will also be featured.

A number of trajectory options are available for planetary missions. Figure 1 illustrates several ballistic (high thrust-to-weight) options. The direct trajectory class includes the minimum energy (Hohmann) trajectory. The propulsion energy requirement and/or flight time can frequently be reduced by flying the Earth (EGA) or Jupiter (JGA) gravity assist trajectories.

Figure 2 presents the spacecraft ballistic velocity requirement as a function of flight time for the Neptune orbiter mission for a JGA trajectory and a combination EGA/JGA trajectory. This figure is based on an initial 270-km circular orbit about the Earth and a final 3×300 r_n orbit about Neptune. The propulsion requirements for this mission are representative of the requirements for the Mars, asteroid and comet sample return, and outer planet cruiser missions. In the past, the injection velocity has been provided by the booster or a booster/kick stage combination. Future spacecraft using the Space Transportation System (STS) will rely on the Centaur, an advanced orbit transfer vehicle (OTV), or the spacecraft propulsion system to meet this requirement. It should be noted that the orbit insertion velocity requirement is a major fraction of the total mission requirement.

The performance of a number of chemical propulsion systems for the Neptune orbiter mission is shown in Fig. 3. This analysis is based on injection by the wide-body Centaur, the performance of which is shown in Fig. 4, and an EGA/JGA trajectory ($V_\infty^2 = 30$ km^2/s^2). Both pump-fed and pressure-fed propulsion systems are considered. Only pressure-fed spacecraft propulsion systems have been flown to date. The pump-fed system operates in a blowdown mode at a lower propellant tank pressure than the pressure-fed system. This permits the use of light-weight, thin-wall tanks and eliminates the requirement for a pressurization system. The associated mass savings offset the requirement for a battery and motor-pump assembly in pump-fed systems designed for large velocity increments. Calculations[1] have shown that pump-fed systems provide a greater payload than pressure-fed systems for missions requiring more than 450 kg of propellant.

The space storable F_2/N_2H_4 propellant combination provides a specific impulse of 370 s. F_2/H_2 provides a specific impulse of 460 s but requires active refrigeration for fuel thermal control. Figure 3 indicates that the advanced systems provide a significant increase in on-orbit mass or a reduction in trip time in comparison to that provided by the Viking class pressure-fed Earth-storable propulsion system. Nevertheless, the best performing chemical system requires a trip time of 15-16 yr to deliver a 1500-kg payload to an orbit about Neptune. Such trip times are well in excess of demonstrated spacecraft lifetimes and translate into large mission support costs. Clearly, improved performance in terms of reduced trip time is highly desirable for a viable far outer planet exploration program.

Advanced Propulsion Options

The specific energies for candidate energy sources for advanced propulsion applications are given in Fig. 5. Aerocapture is not a propulsion option in the conventional

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*Member Technical Staff, Advanced Technology and Applications. Member AIAA.

Table 1 Planetary spacecraft propulsion requirements

Spacecraft[a]	Propulsion system	Payload mass,[b] kg	Spacecraft ΔV, km/s	Flight time, yr
MM '69, F/B	N_2H_4	360	0.05	0.4
MM '71, o	N_2O_4/MMH-GN_2	400	1.70	0.5
MVM '73, F/B	N_2H_4	460	0.13	0.4
Viking, O + L	N_2O_4/MMH-GN_2	700/1850[c]	1.50	0.8
Voyager, F/B	N_2H_4	690	0.15	4.0
PV '78, 0	Solid-N_2H_4	320	1.05	0.5
PV '79, F/B + 4P	N_2H_4	840[c]	0.02	0.3
Galileo, O + P	N_2O_4/MMH	850/1190[c]	1.70	2.3
VOIR, O	Solid-aerobrake-Earth storable	1200-1400	4.30	0.3

[a] F/B—Flyby, O—Orbiter, P—Probe, L—Lander. [b] Spacecraft burnout mass less propulsion system mass.
[c] Probe or Lander mass included.

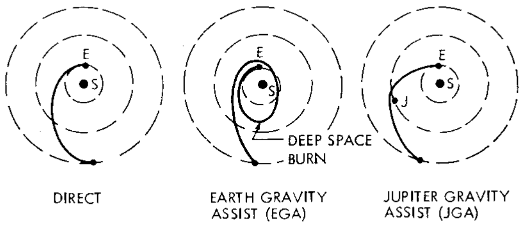

Fig. 1 Ballistic transfer trajectories.

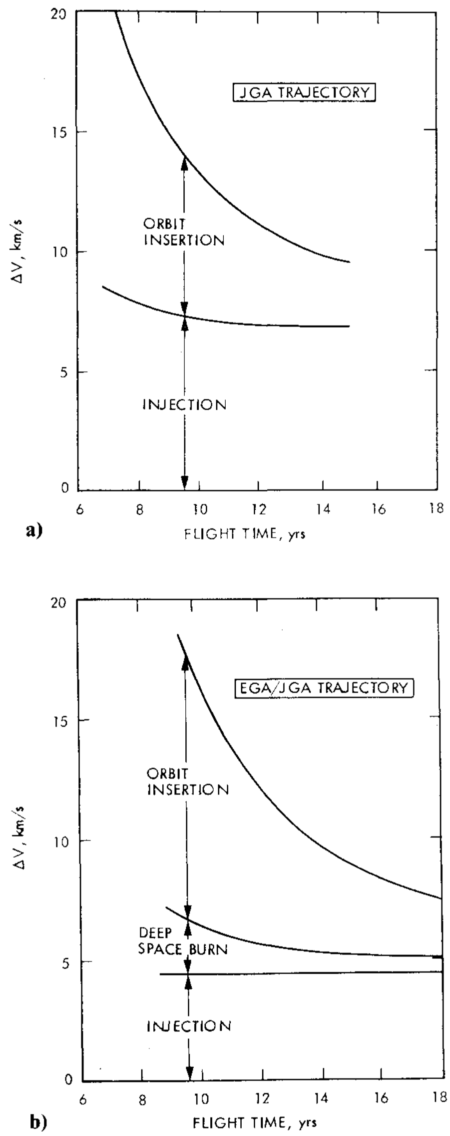

Fig. 2 Ballistic velocity requirement for Neptune orbiter mission.

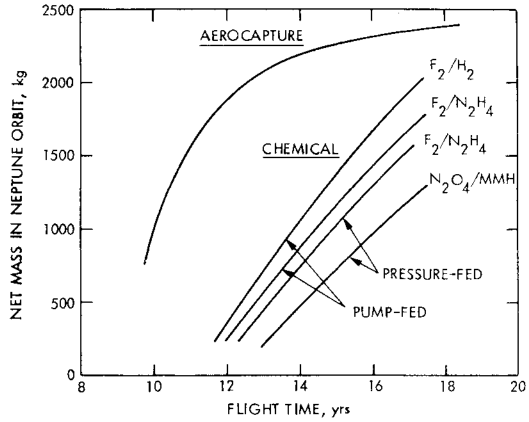

Fig. 3 Propulsion and aerocapture performance for Neptune orbiter mission.

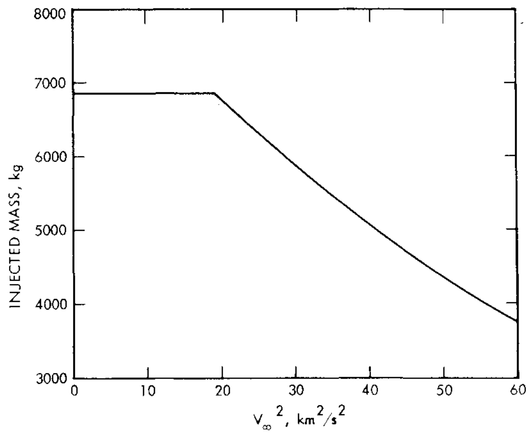

Fig. 4 Shuttle/Centaur performance.

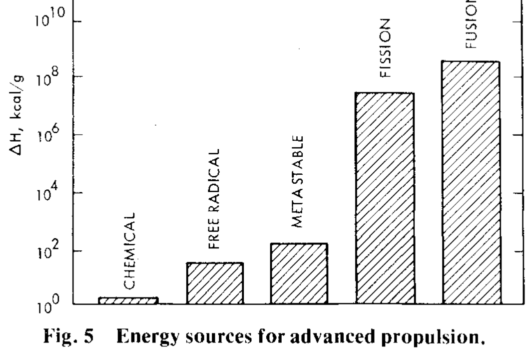

Fig. 5 Energy sources for advanced propulsion.

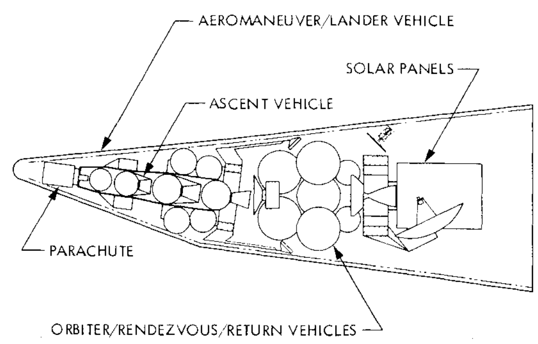

Fig. 6 Aerocapture/sample return spacecraft.

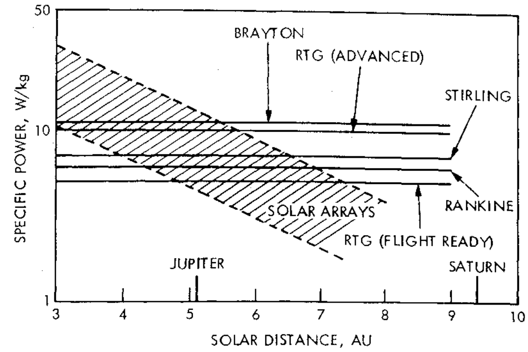

Fig. 7 Electric power supply specific power.

sense, but appears to be a competitive option for orbital insertion requirements for targets with sufficient atmosphere to decelerate the spacecraft. A conceptual design for an aerocapture vehicle is shown in Fig. 6.[2] The lift-to-drag ratio is controlled by varying the attitude of the vehicle with a combination of attitude control system thrusters and aerodynamic control surfaces. The trajectory is designed to maximize aerodynamic braking without exceeding vehicle structural and aerodynamic heating limits. The performance of an aerocapture vehicle for the Neptune orbiter mission is shown in Fig. 3. The potential for a significant gain in performance over that provided by chemical retropropulsion systems is indicated. Major aerocapture issues that remain to be resolved include the thermal control and the flight dynamics and control of the spacecraft during atmospheric entry.

A study was conducted to review and evaluate the performance of advanced chemical and nuclear propulsion concepts for unmanned missions to the outer planets. The characteristics of those concepts reviewed are discussed in the following paragraphs.

Chemical

The performance of state-of-the-art (SOA) liquid propellant rocket engines is near the theoretical limit for known propellant combinations. The energy release for the F_2/H_2 reaction, one of the most energetic propellant combinations known, is 3.1 kcal/g, which corresponds to an ideal specific impulse of 530 s. The O_2/H_2 Space Shuttle main engine (SSME) operates at a specific impulse of 460 s.

A number of chemical reactions provide a large specific energy release but are unacceptable as rocket propellants because a significant fraction of the reaction product is nongaseous. Tripropellant concepts attempt to effectively utilize this energy by introducing hydrogen as a working fluid in addition to the usual fuel and oxidizer. Beryllium-oxygen-hydrogen and lithium-fluorine-hydrogen are attractive tripropellant combinations with an ideal specific impulse of approximately 700 s. Experimental programs conducted by NASA and the Air Force,[3] however, have failed to demonstrate this performance potential.

Free radical and metastable fuels have been identified as energetic propellant candidates. The recombination of atomic hydrogen provides a specific energy release of 52 kcal/g and an ideal specific impulse of 2130 s. The decay of triplet helium to the ground state is characterized by an energy release of 114 kcal/g and an ideal specific impulse of 3150 s. Low concentrations of these species have been produced in the laboratory, but major problems remain in the production of high concentrations and in the storage and controlled combustion of these propellants. The short radiative life of triplet helium (2.3 h) appears to be a major obstacle to the storage and use of this fuel.[4] The low temperatures (0.2 K) and high magnetic fields (30 kG) required for H storage in a solid H_2 matrix will likely limit the H molar concentration to 25%, which corresponds to an ideal specific impulse of 830 s. Major breakthroughs in magnet and refrigeration system design will be required to capitalize on this potential. Control of the combustion or recombination of the free radicals also presents major problems. Direct burning of the solid H-H_2 matrix is a possible solution to this problem.

Laser Thermal

Laser propulsion concepts based on transmitters located in Earth or 1-a.u. solar orbit can provide an ideal specific impulse of approximately 4600 s and a high stage mass fraction.[5] Beam spreading and beam pointing requirements make this concept noncompetitive for the transmission range (30-50 a.u.) required for far outer planet orbital operations.[6]

Solar Thermal/Electric

Collector requirements dictated by the solar flux intensity will restrict solar electric or solar thermal operation to heliocentric radii less than about 6 a.u. This is illustrated for small power supplies (5 kW) in Fig 7, which compares the specific power of various radioisotope power supplies with that for solar photovoltaic arrays as a function of solar distance.[7]

Fission

The structural integrity of the nuclear fuel limits the propellant temperature of the solid core nuclear rocket[8] to less than about 2700 K and specific impulse to 900 s. This performance represents only a small fraction of the potential fission yield of 10^7 kcal/g, which corresponds to an ideal specific impulse of 10^6 s. Advanced solid core concepts such as the particle bed reactor[9] provide somewhat higher performance; significantly higher performance will require concepts which permit a relaxation of the nuclear fuel temperature constraint. The gas core nuclear rocket[10,11,12] with a specific impulse between 1500 and 6000 s and the nuclear pulse rocket[13] (Orion) with a specific impulse between 1500 and 2500 s are examples of such concepts.

The nuclear electric rocket,[7] based on thermoelectric or dynamic thermal-to-electric conversion techniques, provides a specific impulse between 3000 and 6000 s and a specific mass between 10 and 50 kg/kW.

Criticality requirements for nuclear fission ultimately limit the minimum size at which a nuclear rocket engine can operate efficiently. The power density of small solid core nuclear rockets is significantly less than that for higher power nuclear rockets. Criticality requirements also dictate the minimum bomblet size for the nuclear pulse engine, and along with the low density of the gaseous fuel, strongly influence the minimum size of the gas core reactor. In both cases large engine masses are required.

Table 2 Advanced spacecraft propulsion system characteristics

				Stage dry mass, kg $= A + B^* M_p$				
				Stage 1		Stage N		
Class/concept	I_s,s	T, kN	P_j, mW	A	B	A	B	Reference
Chemical								
F_2/N_2H_4	370	4.5	8	160	0.14	160	0.14	c
F_2/H_2	460	4.5	10	200	0.13	200	0.13	c
Nuclear, fission								
Solid core	860	72.6	300	2.6×10^3	0.25	$300^{a,b}$	0.25	8
Particle bed	1000	230	1×10^3	3.2×10^3	0.25	$340^{a,b}$	0.25	9
Gas core—A	2000	445	4×10^3	5.7×10^4	0.25	125^a	0.25	10
Gas core—B	1500	43	450	1.5×10^4	0.25	125^a	0.25	11
Pulse (Orion)	2600	3.5×10^3	4×10^4	9.4×10^4	0.05	Not Applicable		13
Electric	6100	$< 10^{-2}$	0.05-0.4-0	$\alpha = 50\text{-}30$ kg/kW		Not applicable		c
Nuclear, fusion								
Pulse	6300	970	3×10^4	2×10^5	0.20	Not applicable		13

[a] Only tank modules staged. [b] Cooldown propellant included. [c] Based on results of this study.

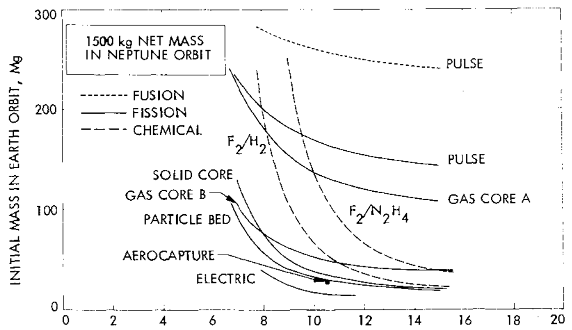

Fig. 8 Advanced propulsion system performance for Neptune orbiter mission.

Fusion

A typical nuclear fusion reaction yields approximately 10^8 kcal/g, which corresponds to an ideal specific impulse of 3×10^6 s. Studies[13] indicate that a nuclear pulse application of this technology can provide a specific impulse between 10^5 and 10^6 s. The size of the fusion engine is not constrained by bomblet critical mass requirements, but studies indicate that the mass of the magnet and fusion trigger systems will limit the application of this technology to large vehicles.

Mission Performance

A first-order assessment of the performance of these advanced concepts for the Neptune orbiter mission was performed. Propulsion system performance was modeled based on the assumption of a linear relationship between the dry mass of the stage M_s and the propellant M_p mass.

$$M_s = a + bM_p \tag{1}$$

The constant term in this relationship was selected to reflect the mass of the rocket engine and other hardware, which was judged to be independent of propellant mass. The coefficient of the linear term was selected to model the mass of the propellant tankage and thermal control systems. Table 2 presents a tabulation of the stage characteristics used in this analysis. Two SOA or near-SOA chemical spacecraft propulsion systems are included for comparison. These are the same pump-fed propulsion systems addressed in Fig. 3. Performance and engine mass estimates for the nuclear rockets are based on data from the indicated references. Two gas core fission engines are considered. Both are closed cycle engines, but the second engine[11] is a small engine based on a low temperature beryllium reflector design. The tankage factor [b in Eq. (1)] for systems using a hydrogen propellant corresponds approximately to a tank mass factor of 16 kg/m^3, which is defined as the tank mass per unit propellant volume. This factor is consistent with data presented in Ref. 14 for the Saturn O_2/RP-1 and O_2/H_2 stages and for the STS external tank. A nuclear electric propulsion stage based on 30-cm mercury ion engine performance and a reactor/thermoelectric conversion system is also shown. Significant improvements in performance can be achieved with dynamic conversion systems for large nuclear electric vehicles. The fusion pulse propulsion system is based on a high thrust-to-weight ratio designed by Winterberg presented in Ref. 13 in which a hydrogen propellant is used to increase thrust at the expense of reduced specific impulse.

The performance of the advanced concepts was evaluated based on the total impulsive velocity requirement for the Neptune (JGA) mission shown in Fig. 2. Staging (up to a maximum of five stages) was assumed when the required stage velocity increment approached the effective exhaust velocity of the propulsion system. Stage performance was estimated based on a form of the impulsive rocket equation in which Eq. (1) had been substituted.

$$M_0 = \frac{a + M_{pl}}{(b+1)\exp(-\Delta V/gI_s) - b} \tag{2}$$

In this equation M_0 represents the initial mass of the vehicle, M_{pl} represents the mass of the payload, I_s represents the specific impulse, and ΔV represents the required velocity increment. The performance of the nuclear electric stage was based on the results of an integrated low thrust trajectory analysis.

The results of these calculations are shown in Fig. 8 for a 1500-kg payload in orbit about Neptune. The vehicle mass (wet) required in Earth orbit was selected as a figure-of-merit since this parameter is directly related to the number of STS launches required to support the mission and less directly to the cost of the mission. The superiority of the nuclear electric propulsion system for this relatively high velocity mission is apparent. As the stage velocity requirement approaches the exhaust velocity of the propulsion system, a large increase in the stage mass ratio $[M_0/(M_0 - M_p)]$ is required to provide additional stage velocity. A significant reduction in the trip time previously quoted is possible with chemical propulsion, but at the expense of large numbers of STS launches and the necessity of on-orbit assembly operations. The solid core nuclear rocket options (including the particle bed reactor) and the small gas core rocket provide superior performance to chemical propulsion but are not competitive with the nuclear electric system. The Orion and large gas core concepts require a very large vehicle mass in Earth orbit (greater than 200 Mg) to be competitive with the chemical or solid core nuclear systems. The fusion propulsion system provides high performance but requires an initial mass greater than 400 Mg to be competitive.

It should be emphasized that this analysis is first-order. It does, however, serve to illustrate the trends and relative performance of the advanced systems for the unmanned outer planet orbiter missions. In this analysis the mass of the powerplant, and hence the power output, has been fixed while allowing the initial mass of the vehicle to vary. The impulsive rocket equation provides only an approximation of the true propulsion requirement for the low acceleration trajectory that results for large values of initial mass. The scope of this study was limited to a consideration of the propulsion requirements for the planetary orbiter mission. Missions which require soft lander and/or sample return operations will require high thrust-to-weight ratio propulsion systems for these functions.

Summary and Conclusions

1) Pump-fed F_2/H_2 and F_2/N_2H_4 spacecraft propulsion systems can deliver to Neptune a significantly greater payload than that delivered by SOA Earth-storable propulsion systems.

2) Aerocapture, advanced propulsion, and/or on-orbit assembly of high performance chemical propulsion systems will be required to significantly reduce the trip time required for this mission to less than 15 yr.

3) A nuclear electric stage launched by a single Shuttle (30 Mg) can deliver the 1500-kg payload to Neptune orbit in approximately 10 yr.

4) Nuclear rocket concepts (fission and fusion) are not competitive with the nuclear electric propulsion stage for this mission class.

Nuclear electric propulsion in combination with chemical propulsion for high acceleration functions is the most attractive advanced propulsion concept for the unmanned exploration of the far outer planets. The 100 kW_e thermoelectric system considered in this analysis is a first generation design. Significantly higher performance is possible with larger nuclear electric systems using dynamic conversion techniques. Aerocapture appears to be a highly competitive option for orbital insertion requirements for solar system targets with atmospheres. Other advanced concepts may prove to be more attractive for missions which require significantly larger payloads and higher velocities (e.g., fast trips to the outer planets and interstellar missions).

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