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Advanced Space Propulsion for the 21st Century

Robert H. Frisbee
Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California 91109

Introduction

THE overriding goal of advanced space propulsion is to reduce the costs of doing space missions, and to enable totally new types of missions that basically could not be performed (even at essentially unlimited cost) with existing technology.

For example, with Earth-launch costs on the order of \$10,000 per kilogram (comparable to the current cost of gold!) of delivered payload, one important application of advanced propulsion is to reduce the cost of access to space (and transportation once in space), both in the initial Earth-launch vehicle and then in the in-space vehicles, by reducing the total mass (most of which is propellant) that must be launched from Earth. This can be achieved by either developing more efficient, higher-performance (i.e., higher specific impulse I_{sp}) propulsion technologies, or by reducing the propellant requirements by reducing dry mass, mission velocity ΔV, and so on. (Note however that launch cost is strongly driven by launch frequency;[1] thus, some advanced propulsion concepts attempt to both improve performance and increase launch rate.)

A second goal of advanced propulsion is the ability to perform previously "impossible" missions. An example of an impossible mission is attempting an interstellar mission using chemical propulsion. No matter how large the rocket, no matter how many stages it has, you simply cannot achieve the speeds (typically at least 10% of the speed of light) required for a practical interstellar mission using a chemical propulsion system. More generally, it is not practical to perform a space mission where the mission ΔV is greater than a few (e.g., two to three) times the propulsion system's exhaust velocity V_{exhaust} or, equivalently, I_{sp}.

Ultimately, the performance of any rocket is limited by the Rocket Equation (first derived in 1903 by Konstantin E. Tsiolkovsky):[2]

$$M_0/M_b = \exp(\Delta V/g_c I_{sp}) \tag{1}$$

$$M_0 = M_b + M_p \tag{2}$$

where M_0 is the initial (wet) mass, M_b the final burnout (dry) mass, ΔV the velocity change, g_c the unit conversion between ΔV and I_{sp} ($g_c = 9.8$ m/s^2 for I_{sp} in lb$_f$-s/lb$_m$ [seconds] and ΔV in m/s^2), I_{sp} the specific impulse, and M_p the propellant mass. Additionally,

$$M_b = M_{\text{dry}} + M_{\text{payload}} \tag{3}$$

where M_{dry} is the propulsion system dry mass (without propellant) and M_{payload} is the mass of payload (everything that is not propulsion system dry mass). The wet mass of the vehicle M_0 is an exponential function of the specific impulse I_{sp} and the required mission velocity ΔV, as well as the dry mass M_b of the vehicle, with the dry mass of the vehicle being in turn a strong function of the propulsion system dry mass M_{dry}. Historically, much of the emphasis in advanced propulsion technology has focused on increasing I_{sp} because of its major impact on the Rocket Equation. Thus we see the progression from chemical (I_{sp} around 200–500 lb$_f$-s/lb$_m$ [2–5 km/s]) to electric (I_{sp} around 1000–10,000 lb$_f$-s/lb$_m$ [10–100 km/s]) to nuclear (I_{sp} around 1000 lb$_f$-s/lb$_m$ [10 km/s] for fission to 10,000,000 lb$_f$-s/lb$_m$ [100 km/s] for antimatter). However, another approach is to reduce the propulsion system dry mass. This can be done using, for example, inflatable structures, micropropulsion components, or beamed-energy concepts, where the energy source for the propulsion system is taken completely off of the vehicle.

Another approach is to reduce the total ΔV that must be supplied by the propulsion system. This can be done through aerodynamic means (aeroassist) to slow down, or by gravity assist or aerogravity assist to speed up. Also, some of the ΔV can be shifted from the space vehicle to a fixed ground-based or space-based system, such as by using a launch catapult or a tether.

Finally, the most extreme approach is to "cheat" the Rocket Equation by using some technique to circumvent the assumption inherent in its use. Specifically, the Rocket Equation assumes that all of the propellant being used is carried onboard the vehicle. However, this need not be the case. For example, a jet engine carries a small amount of onboard fuel, but it collects a much larger mass of "free" (free as far as the Rocket Equation is concerned) air for propulsion. Similarly, in space it is possible to collect energy (e.g., solar-thermal or solar-electric power systems), momentum (e.g., solar sails), or propellant mass (e.g., propellants made from lunar or Martian resources) from extraterrestrial resources. Because you do not carry everything with you from the start, but collect energy, materials, etc., as you travel, it is possible to get a major multiplication in performance as compared to the basic, inherent limitations of the Rocket Equation.

Thus, as shown in Table 1, we can categorize the various advanced propulsion concepts by their impact on the Rocket Equation. For example, advanced chemical, nuclear, and electric propulsion concepts seek to increase I_{sp}. Concepts such as micropropulsion and beamed-energy propulsion seek to reduce the dry mass of the propulsion system by either using ultralightweight components, or by taking part of the propulsion system (e.g., the energy source) off of the vehicle. The ΔV that must be provided by the propulsion

Robert Frisbee received a Ph.D. in physical chemistry from the University of California and taught for one year at the California Polytechnic State University, San Luis Obispo. He joined the Jet Propulsion Laboratory (JPL) Solid Propulsion Group in 1979 and is now a senior member of the technical staff in the Advanced Propulsion Technology Group. He has been involved in the NASA-funded Advanced Propulsion Concepts Task at JPL since 1980. He has participated in a number of advanced propulsion and mission design studies in that time, including advanced chemical, nuclear, electric, and laser propulsion. He has also been involved extensively with studies of the use of extraterrestrial materials. Finally, he has appeared on several programs on the cable television Discovery and Learning Channel (including a show on the 25th anniversary of *Star Trek*). He is a Member of AIAA.

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Table 1 Categorization of advanced space propulsion concepts

Concept	Increase I_{sp}	Reduce dry mass	Reduce ΔV	Circumvent the Rocket Equation
Advanced Chemical	X	——	——	——
Nuclear	X	——	——	——
Electric propulsion	X	——	——	——
Micropropulsion	——	X	——	——
Beamed energy	——	X	——	——
Beamed momentum	——	——	——	X
Aero/gravity assist	——	——	X	——
Launch-assist catapults	——	——	X	——
Tethers	——	——	X	——
Extraterrestrial resource utilization	——	——	——	X
Breakthrough physics	——	——	——	X

system is reduced or eliminated by concepts such as aero/gravity assist, launch-assist catapults, or tethers. Also, extraterrestrial resources can be processed to provide an unlimited supply of propellants, thus eliminating the need to carry all of the propellant needed from the beginning of the mission. Alternatively, beamed-momentum (e.g., solar sail) propulsion can eliminate the need for any propellant by using momentum sources available in space (e.g., solar photons). Finally, it might be possible to develop totally new, breakthrough physics theories that make 20th century models like relativity and quantum mechanics seem as quaint and outmoded as 19th century models like Newton's equations of motion and gravity or Maxwell's field equations of electromagnetism. The remainder of the paper discusses these major categories of advanced propulsion concepts.

Advanced Chemical Propulsion

Since the earliest days of rocketry, there has always been a demand for improvements in chemical propulsion. These improvements have involved several areas, such as increased specific impulse I_{sp}, reduced stage dry mass (e.g., lighter engines, tanks, valves, etc., or higher propellant density), engine throttleablity, increased storage lifetime, or improved safety. In each case the improvements have been sought to address a particular mission performance need.

This section deals with a variety of advanced chemical-propellant systems ranging from near-term advanced solid- and liquid-propellant systems, to high-energy fuels and oxidizers, to far-term, very advanced chemical propellants employing, in some cases, atomic (free-radical) or excited metastable species to achieve the maximum possible I_{sp} that can be derived from chemical bond energies. Also, a discussion of low-temperature thermal control concepts is given because of the need to store many of these advanced chemical propellants at cryogenic temperatures for extended periods of time.

Hybrid Solid Rockets

A typical solid rocket motor is very simple; it consists of a high-pressure motor case and nozzle, a solid-propellant "grain," and an ignitor. Typical specific impulse I_{sp} performance ranges from 260 lb_f-s/lb_m (2.55 km/s) for the space shuttle solid rocket boosters to 290–300 lb_f-s/lb_m (2.84–2.94 km/s) for high-performance upper-stage motors used in space. The inherent simplicity of a solid rocket motor results in a relatively low-stage tankage fraction (TF) of around 10%, where TF is defined as the mass of motor case, nozzle, ignitor, etc. (i.e., the dry or empty mass of the motor without propellant) divided by the usable mass of propellant.

By contrast, in a hybrid solid/liquid-propellant rocket,[3] a liquid oxidizer is sprayed onto a solid-fuel grain during the combustion process. (A corresponding "reverse" hybrid uses a liquid fuel and solid oxidizer.) The hybrid-solid/liquid motor attempts to overcome some of the disadvantages of a pure-solid motor (e.g., single-shot use, no throttleablity, modest I_{sp}) while maintaining its advantages (e.g., low-stage tankage fraction, high thrust, simplicity, and reliability); for example, a hybrid motor can be stopped and restarted (and throttled to control thrust), whereas a pure solid has a one-shot burn to completion. Also, a hybrid can use higher-energy fuels and oxidizers (e.g., liquid oxygen vs solid ammonium perchlorate) because the propellants are physically separated to achieve a higher I_{sp}. The primary disadvantage is the added complexity and dry weight (and reduced reliability) of the liquid-propellant storage and feed system; nevertheless, there continues to be much research into hybrid-solid/liquid rockets with conventional propellants as well as for the exotic high-energy density materials (HEDM) propellants discussed below.

"Green" Propellants

Although not typically possessing a particularly high I_{sp}, a number of liquid-propellant systems have been investigated (or reinvestigated) in recent years because of their relatively benign environmental impacts. For example, the monopropellant hydrogen peroxide (H_2O_2) can be difficult to use, but its reaction/decomposition products are much more environmentally friendly than a monopropellant like hydrazine (N_2H_4). Similarly, the monopropellant solution of hydroxyl-ammonium nitrate (HAN)[4] in water has been studied because of its relative ease of handling, and because, in the event of a propellant spill, HAN decomposes to nitrate fertilizers in the soil.

High-Energy Chemical Propellants

Liquid-oxygen/liquid-hydrogen (O_2/H_2) propulsion represents the state of the art in liquid propulsion. Large, pump-fed engines using O_2/H_2 (e.g., Pratt and Whitney RL-10, Space Shuttle main engine) are capable of specific impulse values in excess of 450 lb_f-s/lb_m (4.4 km/s). Smaller, pressure-fed O_2/H_2 engines suitable for robotic planetary spacecraft have an I_{sp} of 423 lb_f-s/lb_m (4.15 km/s). However, the high I_{sp} can be offset by the high dry mass of tanks and feed systems required to store cryogenic propellants (e.g., passive insulation as well as active refrigeration is required for long missions to eliminate boiloff losses). Thus, there has been an ongoing search for propellant systems that are willing to sacrifice high I_{sp} in order to minimize the need for "hard" cryogens like liquid hydrogen (20 K liquid storage temperature). For example, "soft" cryogens like liquid oxygen or methane (CH_4) can be stored at around 100 K; these are often referred to as space storable because they can be passively stored in space without the need for active refrigeration. Even more desirable are Earth-storable propellants that can be stored at near room temperature. Table 2 lists some representative chemical propulsion systems, ranging from today's work-horse propellant combinations like Earth-storable nitrogen tetroxide/monomethyl hydrazine ($N_2O_4/CH_3N_2H_3$, NTO/MMH) and cryogenic O_2/H_2, through the advanced space-storable propellants discussed next,

Table 2 Representative chemical propulsion systems

System	Type	Ref. I_{sp}[a] lb_f-s/lb_m	Ref. I_{sp}[a] km/s
NTO/MMH	Earth storable	317	3.11
O_2/H_2	Cryogenic (20 K)	423	4.15
O_2/CH_4	Space storable	330	3.23
ClF_5/N_2H_4	Space storable	329	3.22
OF_2/C_2H_4	Space storable	375	3.68
N2F4/N_2H_4	Space storable	358	3.51
F_2/N_2H_4	Space storable	376	3.68
OF_2/C_2H_6	Space storable	370	3.63
OF_2/B_2H_6	Space storable	325	3.19
Atomic hydrogen			
15% (wt.) in H_2	Cryogenic (<4 K)	740	7.2
100%	Cryogenic (<4 K)	2100	20.6
Metastable He	Cryogenic?	3150	30.9
Metallic H	Cryogenic?	1700	16.7

[a]Reference I_{sp} for contemporary and advanced chemical systems shown for engine conditions appropriate to small, pressure-fed planetary spacecraft propulsion systems (100 lb_f [445 N] thrust rocket engine, combustion chamber pressure of 100 psi [0.689 MPa], and nozzle area ratio of 81). Ideal I_{sp} shown for HEDM propellants.

to the exotic, far-term HEDM propellants like atomic hydrogen (discussed below).

Several space-storable fluorinated propellants were investigated in the in the late 1960s through the early 1980s by the U.S. Air Force and the Jet Propulsion Laboratory (JPL) for upper-stage applications and more recently by the Strategic Defense Initiative/Ballistic Missile Defense Organization (SDI/BMDO) for ballistic-missile-interceptor applications. ClF_5/N_2H_4, OF_2/C_2H_4, and N_2F_4/N_2H_4, with I_{sp} values of 329, 375, and 358 lb_f-s/lb_m (3.22, 3.68, 3.51 km/s), respectively, were investigated. ClF_5/N_2H_4 is attractive because of the high boiling point of ClF_5, which allows storage in low Earth orbit without active cooling. Additionally, it can be stored at room temperature under normal tank operating pressures and is often considered Earth storable. All three combinations have been tested.

Also, F_2/N_2H_4, OF_2/C_2H_6, and OF_2/B_2H_6 have been investigated in the past, but have no current sponsor. The JPL worked on F_2/N_2H_4 (fluorinated hydrazine) until NASA funding was terminated in the early 1980s. F_2/N_2H_4 has an I_{sp} of 376 lb_f-s/lb_m (3.68 km/s) and is an excellent performer in both pressure- and pump-fed systems and is space storable. OF_2/C_2H_6 is also space storable and has an I_{sp} of 370 lb_f-s/lb_m (3.63 km/s) .

The U.S. Air Force evaluated OF_2/B_2H_6 primarily for SDI/BMDO applications. With an I_{sp} of 325 lb_f-s/lb_m (3.19 km/s), hypergolic reactivity, space-storable usage, and decent performance in both pump-fed and pressure-fed systems, OF_2/B_2H_6 seemed very attractive. However, this system suffers from incomplete combustion. Also, deposition problems have occurred, mainly in the injector face orifice. Boron deposits were extreme enough to cause injector burnout, thus severely impinging on performance capability.[5]

A variety of high-performance, SDI/BMD-developed micropropulsion technologies have been considered for robotic planetary missions as a means of reducing spacecraft size (and allowing the use of smaller, less expensive launch vehicles). However, these SDI/BMD-class "smart rock" and "brilliant pebble" propulsion technologies have mission applications that are significantly different than those of typical planetary spacecraft. For example, a typical SDI/BMD application involves an engine run time of a few tens of seconds; typical planetary orbit insertion, landing, or takeoff (for sample return missions) can require engine run times of minutes to hours. Thus, significant technology development might be required for application of SDI/BMD propulsion technologies for planetary missions.

Finally, many chemical reactions provide a larger specific energy release than the oxygen–hydrogen reaction but are unacceptable rocket propellants because the reaction product, or a significant fraction thereof, 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 were investigated for this application, but were found to not provide any significant benefits over F_2/H_2.[6]

Cryogenic Propellant Thermal Control

Passive thermal control methods are adequate for many space-storable propellants, although with the added mass of high-pressure tanks (e.g., liquid oxygen [LO_2] at 133 K at 20 atm [2 MPa] pressure) and insulation. Also, there are spacecraft configuration issues, such as keeping the propellant tanks from seeing hot spacecraft surfaces or the sun. However, active thermal control is required for liquid hydrogen (LH_2) for long-duration spacecraft missions. This requirement represents the most significant challenge that must be overcome if LH_2 is to be used in planetary spacecraft applications.

The thermodynamic vent is an advanced passive control technique that can significantly extend the range of passive thermal control system applicability by minimizing boiloff. Vuilleumier and molecular sorption refrigerators are active thermal control options. Finally, a hybrid combination of the thermodynamic vent and the sorption refrigerator might prove attractive.

The Brilliant Eyes Ten-Kelvin Sorption Cryocooler Experiment (BETSCE), flown on STS-77 in May 1996, was the first space flight of chemisorption cryocooler technology. BETSCE measured and validated critical microgravity performance characteristics of a hydride sorption cryocooler designed to cool long-wavelength infrared and submillimeter-wavelength detectors to 10 K and below. The technology flight validation data provided by BETSCE will enable early insertion of periodic and continuous-operation long-life (>10 years), low-vibration, low-power-consumption, sorption refrigeration technology into future missions.

HEDM Chemical Propulsion Concepts

The Air Force Phillips Laboratory, Edwards Air Force Base, and the NASA Glenn Research Center have ongoing research programs investigating HEDM chemical propulsion. In this approach high-energy chemical species are added to "normal" propellants to increase their I_{sp}, density, thrust, or safety. Currently, these programs are still in the basic research phase.

HEDM Additives

One example of a potential near-term HEDM system is the addition of a few percent by weight of strained-ring organic compounds (e.g., cubane) or a finely powdered metal to kerosene in a conventional liquid-oxygen/kerosene rocket engine. Although not dramatic, even small improvements in I_{sp} can result in significant savings by either reducing the effective \$/kg launch cost or by allowing the use of smaller, less expensive launch vehicles. Also, adding various metals or other compounds to a propellant can both increase I_{sp} and produce a gelled propellant. In this case the primary benefit of the gelled propellant can be its improved safety characteristics as a result of resistance to spilling or leaking.[7]

Ultra-High I_{sp} HEDM Systems

In these systems a HEDM propellant is produced by adding a high-energy atom or molecule to a cryogenic solid. In this approach the low temperature helps stabilize the high-energy component, and, in the case of free-radical atoms, the solid matrix "locks" the atoms in lattice "holes" to prevent the atoms from diffusing through the solid and recombining (which would prematurely release the energy stored in the atoms). For example, hydrogen (H), carbon (C), or boron (B) atoms could be added to a solid molecular hydrogen (H_2) matrix; alternatively, oxygen atoms (O) or ozone (O_3) could be added to a solid molecular oxygen (O_2) matrix. The solid propellants could then be burned in a hybrid-liquid/solid rocket using, for example, a liquid-H_2 fuel and a solid-O_2 oxidizer doped with a HEDM species.

Examples of potential far-term, ultra-high-performance cryogenic HEDM systems include atomic or free-radical hydrogen (H) in a solid H_2 matrix (at liquid-helium temperatures),[8] electronically excited metastable triplet helium (He^*), and metallic hydrogen. Whereas the advanced chemical systems discussed earlier have been sufficiently investigated to characterize their propulsion feasibility, the far-term, ultra-high-performance cryogenic HEDM systems have yet to have even their basic feasibility demonstrated. They should be considered highly speculative, high-risk/high-payoff concepts that will require substantial research to determine their suitability for propulsion applications. Nevertheless, the potential to have chemical propellants with an I_{sp} comparable to that of nuclear propulsion continues to spur interest in these systems.

For example, if it could be produced and stored, pure (100%) atomic H would have an ideal I_{sp} of about 2100 lb_f-s/lb_m (20.6 km/s); even at a concentration of only 15% (by weight) H in H_2, the I_{sp} is 740 lb_f-s/lb_m (7.2 km/s).[9] For metastable helium the ideal I_{sp} would be about 3150 lb_f-s/lb_m (30.9 km/s). Metallic hydrogen, if used as a propellant, would be allowed to expand or relax to its normal nonmetallic solid state, thus releasing the stored energy that was required to compress the solid to the metallic state. For a stored energy content 30 to 40 times that of trinitrotoluene, a specific impulse in the 1700 lb_f-s/lb_m (16.7 km/s) range is expected. In principal, the solid hydrogen could be further combined with an oxidizer (O_2, F_2) for additional chemical energy.

Metallic hydrogen also illustrates the tradeoffs involved in determining overall stage performance as a function of I_{sp} and propulsion system dry mass. For example, the tankage factor of thousand- or

Table 3 Overview of Nuclear propulsion concepts

System	Typical I_{sp} lb_f-s/lb_m	km/s
Fission		
Solid core	800–1,000	8–10
Gas core	2,000–7,000	20–70
Pulsed (ORION)	2,500–150,000	25–1500
Fission fragment	10^6	10^7
Fusion	20,000–10^6	200–10^7
Antimatter/matter annihilation	10^7	10^8

million-atmosphere pressure diamond-materials tanks that might be required to store metallic hydrogen at cryogenic temperatures would need to be determined to assess overall stage performance (i.e., does metallic hydrogen have a high enough I_{sp} to compensate for a high-stage dry mass).

Nuclear Propulsion

Research in nuclear fission and fusion energy sources, and their application to space propulsion, has an exceptionally long history. Because of the enormous energy densities potentially available from fission and fusion, nuclear energy was recognized early on as having enormous potential for space exploration. (In fact, many early science-fiction writers invoked atomic energy in their stories even before we fully understood the physics involved.) Most of the work in fission propulsion dates from the end of World War II and the Manhattan Project (post-1945); however, some basic plasma physics research relating to magnetic confinement fusion dates from the 1930s! For example, there was a large effort in nuclear fission propulsion during the Space Race of the late 1950s, 1960s, and early 1970s. An extraordinary range of ideas was proposed and continues to be proposed. Table 3 summarizes the range of specific impulses characteristic of the various fission, fusion, and antimatter propulsion concepts.

Fission Propulsion

The energy available from a unit mass of fissionable material is approximately 10^7 times larger than that available from the most energetic chemical reactions. Attempts to harness this energy have taken three general approaches: fission reactors, fission pulse, and direct use of fragments from the fission reaction. The reactor approach uses thermal energy from a fission reactor to heat a propellant working fluid such as hydrogen, and then expand the heated hydrogen through a nozzle to produce thrust. All reactor-based concepts are ultimately limited by the temperature limits of their materials of construction; thus, the specific impulse of these systems range from around 800 lb_f-s/lb_m (8 km/s) for a solid-core heat-exchanger fission reactor, up to 7000 lb_f-s/lb_m (70 km/s) for a reactor core containing a fissioning gaseous plasma.

Higher specific impulses can only be achieved by eliminating the need for a reactor core and using the actual fission products as expellant. For example, in the ORION concept explosion debris from a small atomic pulse unit would be used to drive the vehicle. Finally, in the fission fragment approach daughter nuclei from the fission reaction are used as the expellant.

Solid-Core Reactor Fission Propulsion

As shown in Fig. 1, propellant is heated in this engine as it passes through a heat-generating solid-fuel core. An expander cycle drives the turbopumps, and control drums located on the periphery of the core control the reactivity of the reactor. Material constraints are a limiting factor in the performance of solid-core nuclear rockets. The maximum operating temperature of the working fluid (e.g., hydrogen) must be less than the melting point of the fuel, moderator, and core structural materials. This corresponds to specific impulses of around 800–900 lb_f-s/lb_m (8–9 km/s) with a thrust-to-weight ratio (T/W) or acceleration greater than one g (9.8 m/s^2).

Approximately $7 billion was invested in solid-core nuclear rocket development in the United States from its inception in 1956 with Project Rover at the Los Alamos Scientific Laboratory (LASL, now Los Alamos National Laboratory, LANL) until the end of reactor testing in 1972. This work was directed at the piloted Mars mission and concentrated on the development of large, high-thrust engines. A series of 23 reactors and engines based on hydrogen-cooled reactor technology, ranging from 350–4500 MW and from 25,000–250,000 lbs (110–1100 kN) thrust, was built and testing during the 1960s and early 1970s. The cores of these reactors consisted of clusters of fuel elements through which the hydrogen coolant was passed. The fissionable material in the graphite fuel element was in the form of particles of uranium carbide coated with pyrolytic carbon. The flight-rated graphite engine that was developed as a result of this program was called NERVA (Nuclear Engine for Rocket Vehicle Application). This engine was designed to operate at 1500 MW, provide 333 kN of thrust at a specific impulse of 825 lb_f-s/lb_m (8.09 km/s) and have an engine weight of 10.4 metric tons. It was engineered for a 10-hour life and 60 operating cycles.[10,11] A mockup of the NERVA propulsion system is shown in Fig. 2.

A small nuclear rocket engine (SNRE)[11] was designed by LASL (now LANL) that had a 370-MW engine with 72.6 kN of thrust. Two

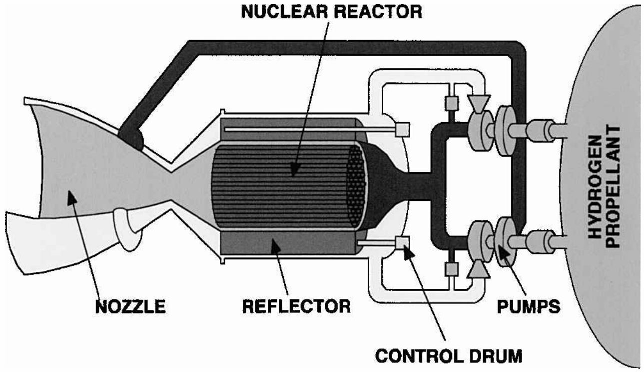


Fig. 1 Solid-core nuclear rocket concept.

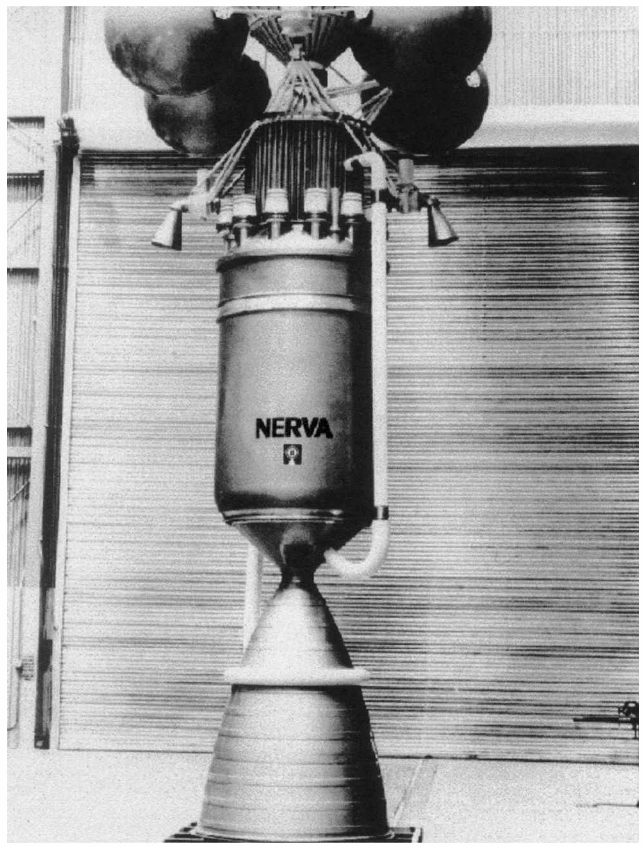


Fig. 2 NERVA propulsion system mockup. (Reproduced with permission of Westinghouse.)

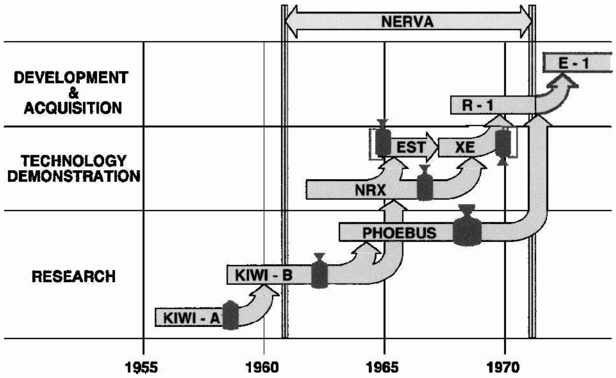


Fig. 3 Nuclear thermal rocket development.

engine designs, both weighing 2.6 metric tons, were proposed: one that operated at a specific impulse of 875 lb_f-s/lb_m (8.58 km/s), and an advanced design that operated at a specific impulse of 976 lb_f-s/lb_m (9.56 km/s). The SNRE was engineered for a 2-hour life and 20 operating cycles. It used a zirconium-hydride moderator to provide the necessary neutronic reactivity in the small core and a high-performance composite (UC-ZrC-C) fuel element.

Finally, it should be emphasized that the NERVA engine development program was very near completion when terminated in 1972 (see Fig. 3). The next step would have involved a flight demonstration in Earth orbit. Since that time, there has been some limited work by NASA Glenn Research Center (GRC, formerly NASA Lewis Research Center) on fuels and materials; also, several NERVA-derivative engines, which would employ modern materials, turbopumps, and turbopump cycles to take NERVA performance into the 900 lb_f-s/lb_m (9 km/s) I_{sp} range, have been proposed by GRC. Additionally, LANL has investigated the issues of ground testing of nuclear rockets such that modern environmental requirements are satisfied.[12] Finally, solid-core nuclear thermal rocket propulsion technology development has been ongoing in the former Soviet Union; as with chemical and electric propulsion technology, the free exchange of information now possible has alerted U.S. researchers to a number of innovative technical approaches developed by their Russian counterparts. Thus, solid-core nuclear thermal rocket propulsion (including the liquid-oxygen [LOX]-augmented and bimodal variants discussed next) represents a relatively mature advanced propulsion technology. By contrast, the more far-term fission-thermal concepts, such as the particle-bed, gas-core, fission-pulse, and fission-fragment systems, should be considered progressively more speculative and less well defined.

LOX-Augmented Nuclear Thermal Rocket

Current NERVA-type nuclear-thermal-rocket (NTR) engine materials technology requires the use of chemically reducing propellants (e.g., hydrogen, ammonia, etc.); strongly oxidizing propellants like liquid oxygen (LOX) cannot be used because they would attack the nuclear fuel and engine materials. One way to use oxygen propellant is the LOX-augmented nuclear thermal rocket (LANTR) concept,[13] originated by NASA Glenn Research Center, which involves the use of a conventional H_2 propellant NTR with O_2 injected into the nozzle. The injected O_2 acts like an afterburner and operates in a reverse-scramjet mode. This makes it possible to augment (and vary) the thrust (from what would otherwise be a relatively small NTR engine) at the expense of reduced I_{sp} (i.e., 940 lb_f-s/lb_m [9.21 km/s] I_{sp} and 15,000-lb_f [67 kN] thrust in pure-H_2 NTR mode vs 647 lb_f-s/lb_m [6.34 km/s] and 41,300-lb_f [184 kN] in LANTR mode at an oxidizer-to-fuel ratio of 3).

There are several potential benefits of the LANTR concept. For example, the cost of ground-test facilities for NTR testing scale with engine thrust (because of the need to scrub the engine exhaust of any nuclear materials); this approach can enable low-cost testing of a small NTR engine capable of producing high thrust in LANTR mode. (The LANTR mode could be tested in a nonnuclear facility separately from the NTR engine testing by using resistively heated [i.e., nonnuclear heated] H_2.) Additional systems-level benefits include reduced gravity losses for liftoff from the moon or for escape/capture in low Earth orbit. Also, because of the potential to use free O_2 made on the Moon it should be possible to reduce the mass (and corresponding Earth-launch costs) of propellants that must be supplied from the Earth. Finally, the O_2 used in the LANTR engine could be derived from several extraterrestrial sources in addition to Earth's Moon, such as water from the moons of Mars or the outer planets, or carbon dioxide from the atmosphere of Mars.

Current work in this area involves mission analysis studies. Also, researchers at NASA GRC recently demonstrated LANTR-type supersonic combustion of oxygen and hydrogen in the supersonic gas flow of hot hydrogen in a rocket nozzle. The hydrogen was preheated (upstream of the nozzle) in a nonnuclear, electrically heated heat-exchanger core to simulate a NERVA reactor core.

Bimodal Hybrid Nuclear-Thermal/Nuclear-Electric Propulsion

In this concept[14] NTR (e.g., solid-core NERVA) is used for high T/W maneuvers in a high-gravity field to minimize gravity losses and trip time. Then, outside of the deep gravity well of a planet or moon the system switches over to a nuclear-electric propulsion (NEP) mode for low-T/W, high-I_{sp} interplanetary transfer. Electric power for the NEP system is obtained by operating the nuclear-thermal rocket reactor at a low thermal power level (so that no NTR H_2 propellant is required for reactor thermal control) with a closed-loop fluid loop (e.g., heat pipes or pumped fluid loop) used to extract heat from the reactor. This thermal energy is in turn used in a static or dynamic thermal-to-electric power conversion system for electric power production. Various nuclear-thermal/nuclear-electric/electric thruster combinations are possible; the most common approach is to assume a solid-core NERVA-type reactor combined with a dynamic power conversion system supplying electric power to ion thrusters.

The mission benefits of this approach are highly mission dependent because there is a tradeoff between the high T/W (e.g., vehicle $T/W > 0.1$) and relatively low I_{sp} (e.g., 800–1000 lb_f-s/lb_m [8–10 km/s]) of the NTR mode and the low T/W (e.g., vehicle $T/W < 10^{-3}$) and relatively high I_{sp} (e.g., 2000–5000 lb_f-s/lb_m [20–50 km/s]) of the NEP mode. For example, T/W impacts gravity losses and thus overall or effective mission ΔV, and the masses of the various system components (e.g., NTR reactor and propellant storage/feed subsystems and the NEP thermal-to-electric power conversion, power conditioning, thruster, and propellant storage/feed subsystems) impact the overall vehicle dry mass.

Finally, even without a dedicated NEP system the bimodal electric power approach can have benefit by eliminating the need for a separate, dedicated electric power system for the various spacecraft systems. For example, the bimodal system can be used to supply tens of kilowatts of electric power (kW_e) for active refrigeration and payload (e.g., crew life support in a piloted mission) power requirements.

Particle-Bed Reactor Fission Propulsion

In the particle-bed (fluidized-bed, dust-bed, or rotating-bed) reactor the nuclear fuel is in the form of a particulate bed through which the working fluid is pumped. This can permit operation at a higher temperature than the solid-core reactor by reducing the fuel strength requirements and thus give specific impulses of around 1000 lb_f-s/lb_m (10 km/s) and T/W greater than one.

Brookhaven National Labs (BNL)[15] has investigated the rotating-bed concept. The engine would have a power of 1050 MW, provide a thrust of 230 kN, and have an engine weight of 4.2 metric tons. The core of the reactor is rotated (approximately 3000 rpm) about its longitudinal axis such that the fuel bed is centrifuged against the inner surface of a cylindrical wall through which hydrogen gas is injected. The fuel bed can be fluidized, but fluidization is not essential. This rotating-bed reactor has the advantage that the radioactive particle core can be dumped at the end of an operational cycle and recharged prior to a subsequent burn, thus eliminating the need for decay heat removal, minimizing shielding requirements, and simplifying maintenance and refurbishment operations.

In the particle-bed concept a porous frit is used to contain the nuclear fuel pellets in the (nonrotating) reactor core. However, the

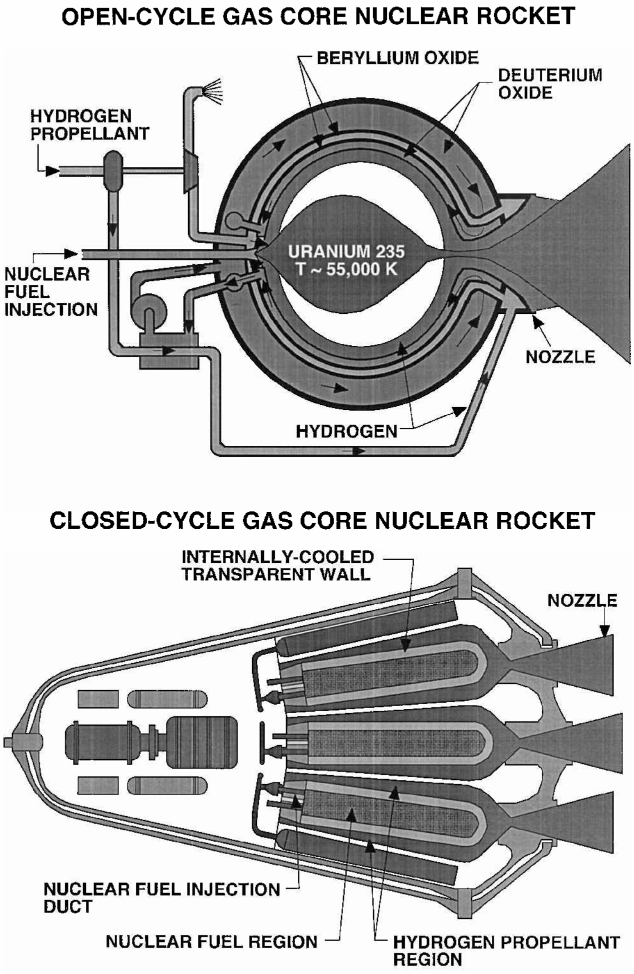


Fig. 4 Gas-core nuclear rocket concepts.

particle-bed concept suffers from a problem of not being able to match power generation with coolant flow. Research and analysis of the particle-bed reactor was also supported by the Department of Defense and the Strategic Defense Initiative Office under the Timberwind program. This was initially a classified program, now declassified, that ran from the late 1980s to the early 1990s.[16]

Gas-Core Reactor Fission Propulsion

Short of using fusion or antimatter, the highest reactor core temperature in a nuclear rocket can be achieved by using gaseous fissionable material. In the gas-core rocket concept radiant energy is transferred from a high-temperature fissioning plasma to a hydrogen propellant. In this concept the propellant temperature can be significantly higher than the engine structural temperature. In some designs the propellant stream is seeded with submicron particles (up to 20%) to enhance heat transfer. Both open-cycle and closed-cycle configurations have been proposed; Fig. 4 illustrates the two concepts. Radioactive fuel loss and its deleterious effect on performance is a major problem with the open-cycle concept. Fuel loss must be limited to less than 1% of the total flow if the concept is to be competitive.

The open-cycle gas-core nuclear rocket[17] relies on flow dynamics to control fuel loss. With both the open- and closed-cycle concepts cooling the engine walls is a major engineering problem. For example, a radiator can be used to actively augment the cooling of the gas-core engine. Specific impulse for a regeneratively cooled engine is limited to approximately 3000 lb_f-s/lb_m (30 km/s). However, addition of an active cooling system for the engine structure, in addition to regenerative cooling, permits use of a higher plasma temperature, resulting in a specific impulse of up to 7000 lb_f-s/lb_m (70 km/s). An open-cycle gas-core engine was estimated in the 1960s to weigh about 200 metric tons; more recent estimates by LANL result in a 60-metric-ton vehicle with a thrust of 15,000 lb_f (67 kN).

Interestingly, the mission benefits of open-cycle gas-core fission rockets are similar to those of fusion rockets (see the following) in that they both are capable of performing ultrafast (i.e., <4 month round-trip) piloted Mars missions. This is caused, in part, by the higher T/W of the gas-core systems, even though the lower T/W fusion systems can achieve a higher I_{sp}.

The closed-cycle gas-core ("nuclear lightbulb") nuclear rocket[18] concept avoids the nuclear fuel loss of the open-cycle gas-core engine by containing the nuclear plasma in a quartz capsule. Thermal radiation from the plasma passes through the quartz capsule to be absorbed by the hydrogen propellant. The nozzle and quartz wall are regeneratively cooled by the hydrogen propellant. A stage using a large lightbulb engine (6000 MW power, 445 kN thrust, 56.8 metric tons engine weight) would be quite large (about 216 tons) and have a low-stage mass fraction (0.57), although the specific impulse would be almost 2080 lb_f-s/lb_m (20.4 km/s) with a T/W near one. A small lightbulb engine (448 MW power, 44.7 kN thrust, 15.1 metric tons engine weight) has been designed to be small enough to be compatible with the shuttle cargo bay with a specific impulse of about 1550 lb_f-s/lb_m (15.2 km/s) and a T/W of about 0.3.

One of the critical issues for the open-cycle engine is the fluid (gas) flow dynamics used to contain the fissioning plasma. A spherical plasma configuration was envisioned in the 1960s; however, a more recent design from LANL,[19] based on detailed computational-fluid-dynamics computer models, is one in which the fissioning plasma is confined by swirl patterns into a torroidal (doughnut) shape. Experimental work on this configuration could be done using radio-frequency heated (rather than fissioning) plasmas.

Nuclear Pulse Rocket (ORION)

Better utilization of the energy yield from the fission reaction is possible with the nuclear pulse concept, where much higher effective exhaust temperatures are possible because of the short interaction time of the propellant with the structure of the vehicle (i.e., there is no need to continuously contain a high-temperature fission plasma, as in the gas-core fission concepts). Fission pulse propulsion was the conceptual basis of the ORION project.[20] In this concept small fission pulse units (<0.1 kton explosive yield) would be dropped at the rate of one pulse unit every 1 to 10 s and exploded at a distance of from 100 to 1000 ft (30–300 m) from the vehicle. The blast from the explosion interacts with a pusher plate, which transmits the impulse to the vehicle through a shock attenuation system.

The ORION vehicle studied by NASA for piloted Mars missions was 10 m in diameter, 21 m long, and had a loaded weight of 585 metric tons. The vehicle was to be assembled on orbit from propulsion, payload, and pulse unit modules launched into Earth orbit by up to eight two-stage Saturn V launches. The mass fraction for the propulsive stage (less payload) was 0.80. In this system some of the pusher plate is also evaporated (ablated) to decrease the effective I_{sp} and to increase thrust; for example, the specific impulse was in the range of 1840–2550 lb_f-s/lb_m (18.0–25.0 km/s) and with a T/W of about 4. Approximately 2000 pulse units would have been required for a 250-day round-trip Mars mission.

Today, various international treaties forbid any atmospheric nuclear explosions, as well as forbidding nuclear weapon storage or explosion in space. Even if these treaties were amended to permit nuclear fission pulse propulsion in space, there would still be formidable technological, operational, and political issues to be overcome before an ORION system could ever be used. Nevertheless, tests on a subscale vehicle using chemical explosives were performed in the early 1960s to demonstrate the pulse propulsion concept. These tests did successfully demonstrate such features as pulse unit feed and delivery, pulse detonation standoff distance and timing, shock absorber operation, and pusher plate interaction with the atmospheric shock wave from the (chemical explosive) explosion. After approximately $11 million spent over seven years, research on the ORION concept ended in 1965.

Fission-Fragment Propulsion

Most fission propulsion concepts use energy from a fission reactor to heat a propellant "working fluid" gas (e.g., hydrogen), which then expands through a nozzle to produce thrust. Ultimately, engine materials structural temperature limits restrict these systems to specific impulses of less than 7000 lb_f-s/lb_m (70 km/s). Fission-fragment propulsion involves permitting the energetic fragments produced in the nuclear fission process to escape directly from the reactor; thus, the fission fragments, moving with a velocity of several percent of the speed of light, are the propellant working fluid. Because these fragments are heavily ionized, they can be directed by magnetic fields to produce thrust for propulsion. Specific impulse in excess of 1,000,000 lb_f-s/lb_m (10^4 km/s, corresponding to an exhaust velocity of 3% of the speed of light) is possible.

A conceptual fission-fragment rocket system that uses nuclear fuels like americium or curium to achieve a high specific impulse and high specific power has been designed by the Idaho National Engineering Laboratory and the Lawrence Livermore National Laboratory.[21] Another approach to fission-fragment propulsion has been developed in which the fissionable material is arranged as a sheet, somewhat similar to a solar sail.[22] A related concept uses antiprotons to induce the fission in the sail.[23]

Fission-fragment propulsion can be considered for an interstellar precursor mission or eventually a near-star interstellar flyby mission because of its potentially high specific impulse (e.g., 0.03c) and high specific power. Additionally, fission-fragment propulsion can be considered for fast planetary missions where high power and high specific impulse are required.

Fusion Propulsion

There are two principle schemes for providing the confinement necessary to sustain a fusion reaction: inertial confinement fusion (ICF) and magnetic confinement fusion (MCF). These confinement schemes result in two very different propulsion system designs. There are literally dozens of different ICF, MCF, and hybrid-ICF/MCF fusion reactor concepts;[24] two possible systems are discussed next and shown in Fig. 5. However, although there is a significant ongoing Department of Energy (DoE) program aimed at demonstrating controlled fusion for terrestrial power plants, space-based fusion propulsion systems are still highly speculative and should be considered relatively far term. Even with a successful DoE terrestrial fusion demonstration, development of a space-based system would represent a significant challenge.

In several cases these fusion propulsion concepts are space-based propulsion spin-offs of the types of fusion reactor technology being developed by the DoE terrestrial fusion research program. However, it is important to remember that an important figure of merit that has driven the focus of DoE research is the desire to ultimately provide a terrestrial reactor power system that provides low-cost electricity (i.e., low dollars per kilowatt-hour) to consumers. Because the figures of merit for a propulsion system are so different (e.g., specific impulse [I_{sp}], specific mass [kg/kW of jet power], etc.), a fusion reactor type (technology) selected for space propulsion and power applications might be very different than one selected for terrestrial electric power production, although the need for high efficiency in both types of systems might ultimately drive us towards similar systems for both ground and space applications.

ICF—Pulsed Fusion

ICF requires high-power lasers or particle beams to compress and heat a pellet of fusion fuel to fusion ignition conditions. In operation, the pellet of fusion fuel (typically deuterium-tritium [D-T]) is placed at the focus of several high-power laser beams or particle beams. The lasers or particle beams simultaneously compress and heat the pellet. Compression of the pellet is accomplished by an equal and opposite reaction to the outward explosion of the surface pellet material. Heating of the pellet results from both the compression and the inputted laser energy (or particle-beam kinetic energy). The pellet's own inertia is theoretically sufficient to confine the plasma long enough so that a useful fusion reaction can be sustained; hence, this fusion reaction is inertially confined.

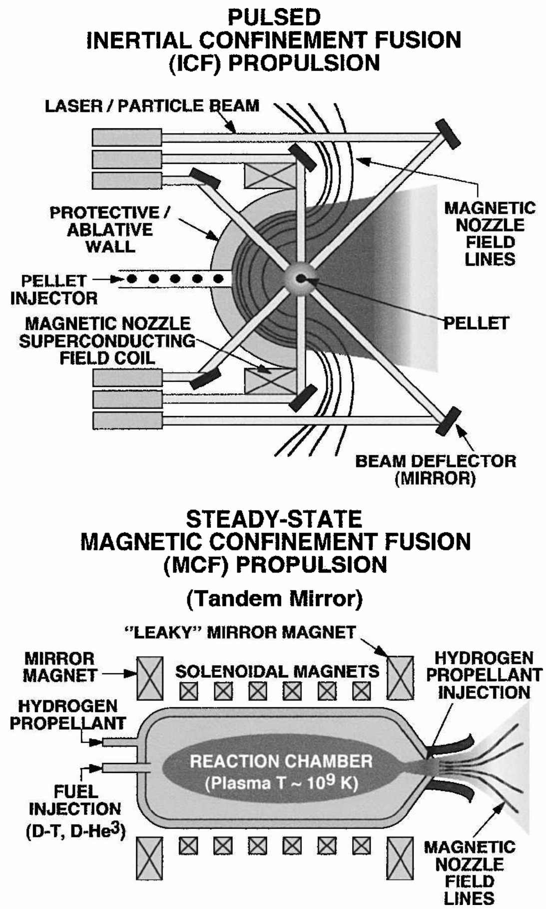


Fig. 5 Fusion propulsion concepts.

MCF–Steady-State Fusion

In contrast to ICF, a MCF reactor confines the fusion plasma with strong magnetic fields. This can be accomplished because the fusion plasma is composed primarily of ions and electrons that can be confined by magnetic Lorentz forces. The energetic fusion plasma is carried to the magnetic nozzle along magnetic drift surfaces in the reactor. For maximum performance for missions within the solar system, it is necessary to mix the plasma with the propellant (e.g., hydrogen) to reduce the specific impulse and increase the thrust level.

Fusion Propulsion Mission Benefits

There are several mission benefits provided by fusion propulsion. Present concepts for fusion propulsion systems are based on technology expected to be available early in the 21st century. Low-I_{sp} systems are best suited to two principal mission types: piloted exploration of the solar system and interplanetary cargo hauling. High-I_{sp} advanced fusion systems might enable interstellar missions.

Studies suggest that an early benefit of fusion propulsion would be the potential for fast piloted missions to a wide variety of planetary targets (e.g., Mars, Jupiter, Saturn). For example, the Vehicle for Interplanetary Space Transportation Applications (VISTA) ICF vehicle study[25] indicated that a piloted ICF-powered spacecraft could accomplish a 60- to 100-day round-trip Mars mission carrying 100 metric tons of payload. Piloted five-year round-trip missions to Jupiter and Saturn also appear to be feasible. This is typical of the performance of fusion-powered spacecraft for missions within the solar system; for these applications the I_{sp} potential of fusion (10^6 lb_f-s/lb_m [10^4 km/s]) is intentionally reduced to around 20,000 lb_f-s/lb_m (200 km/s) by addition of excess hydrogen working fluid to increase thrust. Because of the large ΔV capability (typically hundreds of kilometers/second) and moderately high thrust levels that might be available from fusion-powered spacecraft, they are not as affected by launch windows as existing systems. In addition, missions such as transporting large bulk cargo payloads, moving asteroids, or interstellar flybys become feasible with advanced fusion propulsion systems.

For example, a design study was conducted by the British Interplanetary Society to evaluate the feasibility of ICF for interstellar travel. The vehicle was called Daedalus[26] and was designed for an interstellar flyby with a total ΔV of 0.1c. Daedalus was engineered as a two-stage vehicle with a total mass at ignition of 53,500 metric tons and a final payload of 830 metric tons. The burn time for each stage was estimated to be about two years. The specific impulse for each stage was approximately 10^6 lb_f-s/lb_m (0.03c).

Antimatter Propulsion

Matter–antimatter annihilation offers the highest possible physical energy density of any known reaction substance. The ideal energy density ($E/M = c^2$) of 9×10^{16} J/kg is orders of magnitude greater than chemical (1×10^7 J/kg), fission (8×10^{13} J/kg), or even fusion (3×10^{14} J/kg) reactions. Additionally, the matter–antimatter annihilation reaction proceeds spontaneously, therefore not requiring massive or complicated reactor systems. These properties (high energy density and spontaneous annihilation) make antimatter very attractive for propulsively ambitious space missions (e.g., interstellar travel). This section describes antimatter propulsion concepts in which matter–antimatter annihilation provides all of the propulsive energy; a related concept, in which a small amount of antimatter triggers a microfission/fusion reaction, is discussed next.

Antimatter Propulsion Issues

Not surprisingly, antimatter production, storage, and utilization represent major challenges. Numerous fundamental feasibility issues remain to be addressed, such as scaling up antimatter production rates and efficiencies, storage in a high-density form suitable for propulsion applications, and design and implementation of a complete propulsion system containing all of the ancillary systems required to contain the antimatter on the vehicle and ultimately use it in a thruster. Nevertheless, research aimed at addressing these issues is ongoing at a modest level.

Note that for a propulsion application, proton–antiproton annihilation is preferred over electron–positron (antielectron) annihilation because the products of proton–antiproton annihilation are charged particles that can be confined/directed magnetically. By contrast, electron–positron annihilation produces only high-energy gamma rays, which cannot be directed to produce thrust and do not couple their energy efficiently to a working fluid (and also require significant shielding to protect the vehicle and its payload). Thus, in the annihilation of a proton p^+ and antiproton p^-, the initial annihilation products include neutral and charged pions (π°, π^+, π^-). In this case the charged pions can be trapped and directed by magnetic fields to produce thrust. However, pions do possess mass (about 22% of the initial proton–antiproton annihilation pair rest mass), so that not all of the proton–antiproton mass is initially converted into energy (although the pions subsequently decay into lighter particles and additional energy). This results in an energy density of the initial proton–antiproton reaction of only 78% of the ideal limit or 6.8×10^{16} J/kg.

For these reasons antimatter for propulsion applications is typically assumed to be in the form of antiprotons, neutral antihydrogen atoms (an antiproton with a positron), or antimolecular hydrogen (anti-H_2). The antiproton is identical in mass to the proton but opposite in electric charge and other quantum numbers. Antiprotons do not exist in nature and currently are produced only by energetic particle collisions conducted at large accelerator facilities (e.g., Fermi National Accelerator Laboratory, FermiLab, and BNL in the United States, the European Organization for Nuclear Research [CERN] in Geneva, Switzerland, or the Institute for High Energy Physics [IHEP] in Russia). This process typically involves accelerating protons to relativistic velocities (very near the speed of light) and slamming them into a metal (e.g., tungsten) target. The high-energy protons are slowed or stopped by collisions with nuclei of the target; the kinetic energy of the rapidly moving antiprotons is converted into matter in the form of various subatomic particles, some of which are antiprotons. The antiprotons are electromagnetically separated from the other particles. Note that antiprotons annihilate spontaneously when brought into contact with normal matter; thus, they must be contained by electromagnetic fields in high vacuums. This greatly complicates the collection, storage, and handling of antimatter. Finally, current production/capture/accumulation technology has an energy efficiency of only about one part in 10^9 (i.e., 10^9 units of energy are consumed to produce an amount of antimatter that will release one unit of energy upon annihilation).[27]

Mission propulsion requirements for antimatter require milligrams (10^{21} antiprotons) of antimatter for simple orbit transfer maneuvers, to tens of grams for interstellar precursors,[23] to tons antimatter for interstellar flybys. Currently the highest antiproton production/capture/accumulation level (not optimized for rate or efficiency) is of the order of 10 nanograms per year, although planned upgrades to CERN can increase these production rates by a factor of 10–100. Additionally, only a much lower level of antiprotons have actually been collected, cooled, and stored after production.

Currently, portable antiproton traps are being developed that would allow filling of the trap at an antiproton production facility (e.g., CERN, FermiLab) and transporting the stored antiprotons to a remote research facility. Pennsylvania State University (PSU) completed a Mark I portable antiproton Penning Trap in 1999. It was designed to hold $\sim 10^8$ antiprotons. An improved high-performance antimatter trap, with a 10,000-fold higher capacity is currently under construction at NASA Marshall Spaceflight Center (MSFC).

The technology of scaling production, collection, and cooling rates up to levels required by space missions is still very much in the future. Additionally, the question of high-density storage of antimatter has not been answered. Current concepts for antimatter storage include storing it as neutral anti-H_2 ice suspended in an electromagnetic trap, as slightly charged cluster ions suspended in an electromagnetic trap, and as individual antiprotons stored at quasi-stable lattice points in solid-state crystals.

Antiproton-Catalyzed Microfission/fusion Propulsion

An alternative approach to conventional VISTA-type fusion propulsion systems is the inertial-confinement antiproton-catalyzed microfission/fusion nuclear (ICAN) propulsion concept under study at PSU.[28] In this approach to ICF propulsion, a pellet containing uranium (U) fission fuel and deuterium-tritium (D-T) fusion fuel is compressed by lasers, ion beams, etc. At the time of peak compression, the target is bombarded with a small number (10^8–10^{11}) of antiprotons to catalyze the uranium fission process. (For comparison, ordinary U fission produces two to three neutrons per fission; by contrast, antiproton-induced U fission produces $\sim$16 neutrons per fission.) The fission energy release then triggers a high-efficiency fusion burn to heat the propellant, resulting in an expanding plasma used to produce thrust. Significantly, unlike pure antimatter propulsion concepts that require large amounts of antimatter (because all of the propulsive energy is supplied by matter–antimatter annihilation), this concept uses antimatter in amounts that we can produce today with existing technology and facilities. This technology could enable 100- to 130-day round-trip (with 30-day stop-over) piloted Mars missions, 1.5-year round trip (with 30-day stop-over) piloted Jupiter missions and three-year one-way robotic Pluto orbiter mission (all with 100-MT payloads).

A recent variation on the ICAN concept is AIMStar,[29] which uses an electromagnetic trap (rather than laser or particle beam implosion) to confine a cloud of antiprotons during the antimatter-induced microfission step. This concept can enable the construction of very small systems (at least as compared to a conventional ICF VISTA fusion rocket) because a large ICF-type pellet implosion system is not required. Finally, as discussed in the section on Fission Fragment Propulsion, a related antimatter-induced fission concept uses antiprotons to induce fission in fissionable material arranged as a sheet, somewhat similar to a solar sail.[23]

Electric Propulsion

In electric propulsion, electric energy (from solar cells or a nuclear-electric reactor) is used to energize the propellant working fluid to yield much higher specific impulses than those available from chemical reactions. This has the benefit of dramatically reducing the propellant requirement for a given spacecraft velocity change ΔV or, alternatively, increasing the ratio of final or dry mass M_b divided by the initial or wet mass M_0 of the vehicle.

Note however that the electric propulsion system must carry along an electric power supply to provide energy for the expelled propellant; by contrast, chemical propellants constitute both their own energy source and expellant mass. Thus, the overall mission benefit of electric propulsion involves a tradeoff between propellant mass savings (because of higher I_{sp}) and power system mass, as compared to the chemical propulsion system.

Thus, chemical and electric propulsion systems have intrinsic differences. For example, chemical propulsion is said to be energy limited because the chemical reactants have a finite amount of energy per unit mass (i.e., their enthalpy of combustion or reaction) that ultimately limits their achievable exhaust velocity or specific impulse. However, because the propellants are their own energy source the rate at which energy is supplied to the propellant (which is ultimately limited by the reaction kinetics) is independent of the mass of propellant, so that very high powers and thrust levels can be achieved. By contrast, electric propulsion systems are typically not energy limited; an arbitrarily large amount of energy can be delivered (from the external solar- or nuclear-electric power system) to a given mass of propellant so that the exhaust velocity (or I_{sp}) can be an order of magnitude larger than that of a chemical system. Instead, electric propulsion systems are power limited because the rate at which energy from the external source is supplied to the propellant is proportional to the mass of the power system. This has the result of limiting the thrust of the electric propulsion system for a given vehicle mass. Because of this, electric propulsion vehicles are typically low T/W (i.e., low acceleration) vehicles.

Interestingly, even though electric propulsion vehicles have a low T/W they can have a larger total amount of impulse (I_{sp} multiplied by propellant mass) than a chemical system. Thus, even though the chemical system can have a high T/W, its propellant is quickly expended at a low I_{sp}. By contrast, the low-thrust electric propulsion system can be operated for hours to years and ultimately build up a larger total impulse.

Thus, we see that in general terms electric propulsion can provide significant mass savings, as compared to chemical propulsion, because of its higher I_{sp}. However, trip time benefits for electric propulsion can be a complicated interplay between T/W and the local gravity field. For example, low-T/W electric propulsion missions in cis-lunar space (e.g., low Earth orbit to geosynchronous orbit or lunar orbit) are invariably slower than chemical because the electric propulsion system is deep in the Earth's gravity "well," (i.e., the electric propulsion system has a much lower T/W than the local gravity field). By contrast, in heliocentric space the electric propulsion system has at least a medium T/W compared to solar gravitation, so that with sufficient T/W and system run time (i.e., acceleration multiplied by time) the electric propulsion system can achieve a much higher terminal velocity to reduce trip time. Also, the higher I_{sp} of the electric propulsion system allows it to use less propellant mass than a chemical system would need in order to fly a high-ΔV, short trip time trajectory. Thus, for planetary missions electric propulsion trip times can be significantly less than those of a chemical system, especially for the outer planets where there is the opportunity for long run times for the electric propulsion system to build up to a high terminal (cruise) velocity.

Finally, we often tend to forget that as an advanced propulsion technology electric propulsion actually has a long developmental history stretching back almost 70 years. Considerable research and development, culminating in several flight experiments, was performed in the 1960s during the heat of the Space Race. More recently, various electric propulsion devices have been used on commercial and scientific spacecraft.

Electric Propulsion Subsystems

As shown in Fig. 6, an electric propulsion system consists of a power (e.g., solar or nuclear) system, power conditioning, thruster, and propellant storage and feed subsystem.

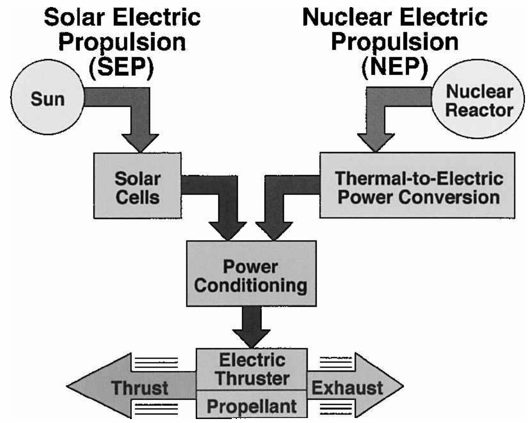


Fig. 6 Electric propulsion systems.

Power

Energy can be obtained from either sunlight or from a nuclear reactor. In the case of solar electric propulsion (SEP), solar photons are converted into electricity by solar cells. In nuclear electric propulsion (NEP), thermal energy from a nuclear reactor is converted into electricity by either a static or dynamic thermal-to-electric power conversion system. Static systems have the advantage of no moving parts for high reliability, but they have low efficiency (typically $<10\%$). Dynamic systems have moving parts (e.g., turbines, generators, etc.) and do not scale well for small systems, but they do have higher efficiency (typically 20–30%). Interestingly, the economy of scale seen with nuclear dynamic power systems allows a significant reduction in system specific mass; for example, at 100 kW_e the power system might have a specific mass around 30–40 kg/kW_e, whereas at 100 MW_e the specific mass might be less than 5 kg/kW_e (Ref. 30).

Power Conditioning

Power conditioning systems are required to convert the voltage from the power system to the form required by the electric thruster. For example, an SEP power system produces low-voltage dc (typically $\sim$100 V); this would need to be converted (via transformers, etc.) to kilovolt levels for use in an ion thruster. By contrast, a Hall thruster (see below) operates at about the same voltage level as the solar arrays; thus, negligible power conditioning is needed for a direct drive combination where the power system output matches the thruster input. Finally, the power conditioning system is often referred to as the power processing unit; this is in turn part of the vehicle's overall power management and distribution subsystem.

Thrusters

As shown in Table 4, various combinations of thruster and propellant are possible, depending on the specific application. These are discussed in more detail next. Electric propulsion represents an extremely active area of ongoing research and development in the United States (NASA,[31] U.S. Air Force,[32] industry,[33] and academia[34]), Europe,[35] Russia,[36] and Japan.[37]

Types of Electric Thrusters

Electric propulsion thrusters can be divided into three broad categories. Electrothermal thrusters use electric energy to simply heat the propellant and add additional enthalpy. Electrostatic thrusters use charge potential differences to accelerate propellant ions. Finally, electromagnetic thrusters use electromagnetic body forces ($\boldsymbol{J} \times \boldsymbol{B}$) to accelerate a propellant plasma. Each of these three categories of thrusters is discussed next.

Electrothermal Thrusters

Electrothermal thrusters use electric energy to heat the propellant by resistive heating for resistojets or by passing the propellant gas through a plasma discharge. The plasma can be generated through a high-current discharge in arcjets or pulsed electrothermal thrusters

Table 4 Representative types of electric propulsion thrusters

Thruster	Typical electric power range	Typical I_{sp} lb_f-s/lb_m	km/s
Electrothermal			
Resistojets	100s of W	300–400	3–4
Arcjets			
Hydrazine	kW	500–600	5–6
Hydrogen	10s of kW	900–1,200	9–12
Ammonia	kW to 10s of kW	600–800	6–8
Electrostatic (Xe propellant)			
Gridded ion engines	W to 100 kW	2,000–10,000	20–100
Stationary plasma thrusters	100s of W to 10s of kW	1,000–2,500	10–25
Thruster with anode layer	100s of W to 10s of kW	1,000–4,000	10–40
Electromagnetic			
Magnetoplasmadynamic			
Steady-state, lithium	100s of kW to MW	3,000–9,000	30–90
Steady-state, hydrogen	>MW	9,000–12,000	90–120
Pulsed plasma thruster	10s to 100s of W (average)	1,000–1,500	10–15
Pulsed inductive thruster	10s of kW	3,000–8,000	30–80

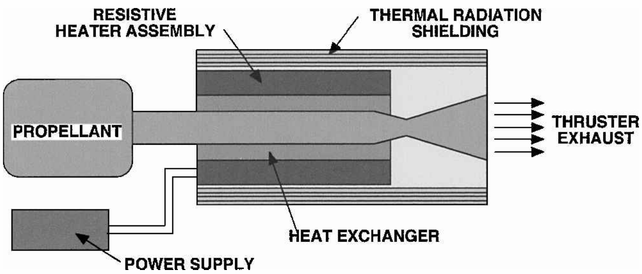


Fig. 7 Resistojet schematic.

(PET) or by absorption of microwaves in microwave electrothermal thrusters (MET). Resistojets and arcjets represent state-of-the-art electrothermal propulsion; they are used for attitude control and stationkeeping on a wide variety of commercial satellites. The PET and MET are still in the research phase.

Resistojets. In a resistojet an electric resistive heater surrounds a heat exchanger through which propellant passes. The propellant is superheated and then ejected through an expansion nozzle. Because of the propellant's high energy (gained by heating), an exhaust velocity much greater than that for a cold gas is achieved. Many resistojet configurations have been conceived and developed. Propellant gases used for resistojets include ammonia, biowastes, hydrazine, and hydrogen. A schematic of a resistojet is shown in Fig. 7.

Hydrazine (N_2H_4) is used in what is called an augmented hydrazine thruster because the energy added by the resistojet augments that obtained by the catalytic decomposition of the hydrazine (e.g., 200–220 lb_f-s/lb_m [2.0–2.2 km/s] I_{sp} for the hydrazine thruster without augmentation). Specific impulse I_{sp} values for the hydrazine resistojet are on the order of 300 lb_f-s/lb_m (2.9 km/s) (comparable to a bipropellant like nitrogen-tetroxide/mono-methyl hydrazine [NTO/MMH]), and thrusters with input power levels of a few hundred Watts and 60–90% efficiency are used routinely in space flight operation.

Arcjets. Arcjets are electrothermal devices that heat the propellant to a higher temperature than can be obtained through combustion processes resulting in higher specific impulse and better propellant efficiency. Several types of arcjets have been configured and are classified by their method of propellant heating. The dc arcjet discussed here is the most highly developed and is being used on commercial communication satellites for north–south stationkeeping.

The dc arcjet has a cylindrically symmetric geometry as shown in Fig. 8; it consists of a cathode, an anode that forms the plenum chamber, constrictor channel and nozzle, and a propellant injector. In operation a high-current (up to several hundred Amperes), low-voltage ($\sim$100 V) arc is established as a laminar column from the cathode tip, through a constrictor channel, and attaches to the anode

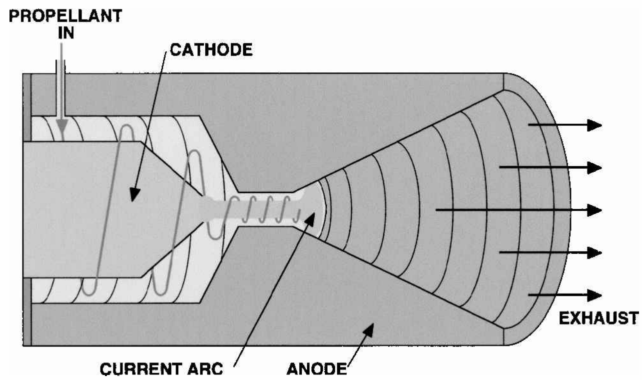


Fig. 8 DC arcjet schematic.

in an axially symmetric diffuse arc. Propellant gas is swirled into the constrictor through injection ports located behind the cathode. (Swirling is done to stabilize the arc, constrain the hot gas discharge column to the axis of the vortex, cool the electrodes and chamber walls, and bring the gas into longer and more effective contact with the arc.) The attainable thrust is limited by the power available, whereas the specific impulse is limited by the nozzle materials.

The dc arcjet possesses the highest thrust-to-power ratio of all electric propulsion devices and has been demonstrated at input power levels ranging from a few hundred Watts to 200 kW_e. Typical engine efficiency with ammonia propellant is 30% at a specific impulse of 800 lb_f-s/lb_m (8 km/s). Specific impulses in the range of 900–1300 lb_f-s/lb_m (9–13 km/s) have been demonstrated by using hydrogen propellant at power levels of 30–200 kW_e. The specific impulse for hydrazine is typically 500–600 lb_f-s/lb_m (5–6 km/s) and an efficiency of about 35% at power levels of 0.5–2 kW_e. Arcjet thrust and specific impulse increase with engine power while efficiency decreases.

Hydrazine arcjets have been used on a variety of commercial satellites not only because of their high I_{sp}, but also because they can replace chemical monopropellant thrusters while retaining much of the rest of the propulsion system (tanks, valves, filters, etc.), therefore significantly reducing the overall system cost.

PET. The PET produces thrust by ejecting a pulsed, high-velocity plasma out of a conventional supersonic nozzle. The geometry of a PET is cylindrical (of very small diameter, e.g., 5 mm) with a cathode at one end of the pressure chamber and an anode at the other end. The cathode end is closed and incorporates liquid-propellant injectors; the anode end is open with the anode forming a supersonic nozzle.

The operation of a PET is relatively simple. As propellant (i.e., liquid hydrogen, hydrazine, or water) enters the pressure chamber, a capacitor initiates a high-pressure, electrothermal discharge. The

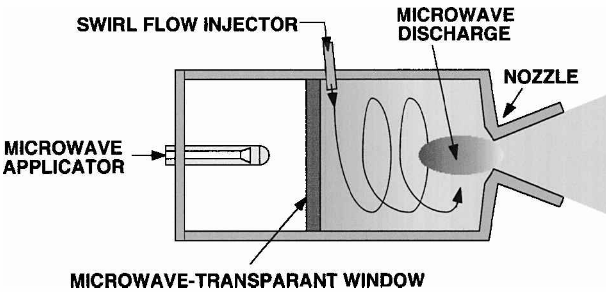


Fig. 9 MET schematic.

discharge ionizes and superheats the propellant gas, which then expands out of the thruster through the supersonic nozzle.

Performance of the PET varies with the propellant choice. A specific impulse of 1400 lb_f-s/lb_m (14 km/s) and efficiency of 54% has been achieved using water for propellant. Higher performance values can be obtained by using liquid hydrogen (predicted specific impulse of ~2900 lb_f-s/lb_m [29 km/s] and ~70% efficiency), whereas heavier propellants (e.g., liquid hydrazine) decrease specific impulse. Additionally, the thrust level can be varied by changing the energy per pulse value, or the pulse frequency.

Development of the PET thruster has been underway in the United States since the early 1980s. However, little work has been done on this thruster concept to demonstrate acceptable electrode erosion for the several million pulse lifetimes projected for most mission applications. No space tests of PET thrusters have occurred.

MET. In the MET concept shown in Fig. 9, microwave energy (typically 2.45 GHz, but lower frequencies down to 900 MHz have been used as well) is fed into the thruster where standing electromagnetic field patterns are set up. Propellant injected into the thruster is heated by the microwave energy, in particular in the maximum field regions. Small amounts of electrons, present in all room-temperature gases, are accelerated in the strong microwave fields in the maximum field regions, causing ionization of neutral atoms until breakdown occurs and a microwave plasma forms. At low flow rates and power levels this plasma remains localized in the maximum field regions and acts as a heating element for the remainder of the propellant flow. The heated propellant is then exhausted through a nozzle to produce thrust. One of the key advantages of this thruster concept over other electrothermal thruster concepts is the fact that it uses no electrodes. Consequently, lifetimes of both pulsed thrusters for attitude control purposes and steady-state thrusters are expected to be significantly higher than those obtainable with arcjets, for example.

Only preliminary thrust stand measurements have been performed with microwave electrothermal thrusters, and no space tests have been performed. Data on thruster efficiencies, however, exist based on thrust and impulse values that have been estimated using numerical calculations. Tests have been performed mostly with nitrogen and helium, although hydrogen and ammonia propellants have been studied as well. Thruster efficiencies obtained with nitrogen range from 40–50% and achievable specific impulse is estimated at about 300 lb_f-s/lb_m (3 km/s). Ammonia values range from 50–70% efficiency and between 400–500 lb_f-s/lb_m (4–5 km/s) specific impulse.

Finally, an interesting tradeoff is possible with the microwave electrothermal thruster in that it can use either an internal, integrated microwave source, or an external source separate from the thruster. Thus, there is a potential for significant synergism between the thruster and other microwave sources, such as high-powered onboard radar or telecommunications systems, as well as the potential for using microwave power beamed from a remote source, as discussed in the Beamed Energy section.

Electrostatic Thrusters

Electrostatic thrusters use charge potential differences to accelerate propellant ions. Strong electric fields are created in the engine, which then accelerate the (positive) ions to high velocities (I_{sp}). This includes ion engines, Hall-effect thrusters, field emission thrusters, and colloid thrusters.

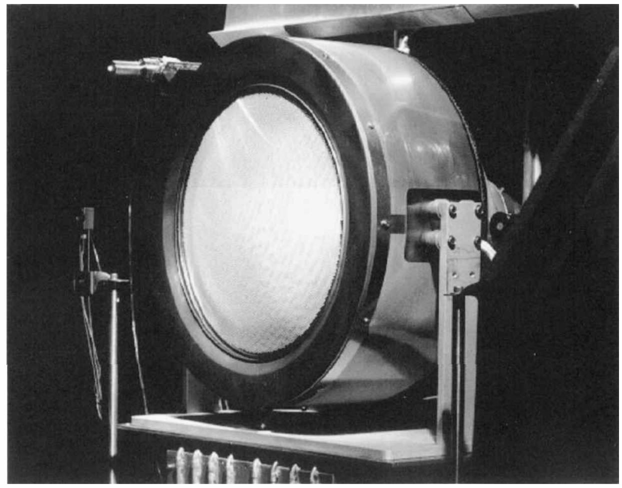


Fig. 10 NSTAR ion thruster during endurance testing at JPL. (Picture reproduced with permission NASA.)

Ion thrusters. Ion propulsion systems have been seriously considered for spacecraft propulsion since the 1950s. Because of their potential for providing both high I_{sp} (>2500 lb_f-s/lb_m [25 km/s]) and high efficiencies (>60%), ion propulsion is well suited for primary propulsion for planetary missions requiring high ΔV. There are numerous types of ion engines categorized according to their source of positive ions. Thrusters that have been experimentally investigated include the contact ion engine, microwave ion engine, plasma separator ion engine, radio frequency (RF) and microwave ion engines, radioisotope ion engine, and the dc electron-bombardment engine. Of these thrusters the dc electron-bombardment ion engine has received the most research and development attention in the United States.

Ion thrusters have been in use for some time in Earth-orbiting spacecraft for stationkeeping applications. However, the first deep-space mission employing ion engines was the NASA New Millennium Program first technology demonstration spacecraft (Deep Space-1 [DS-1]), launched in 1998, which successfully performed an asteroid and comet flyby using the NSTAR (NASA Solar Electric Propulsion Technology Application Readiness) ion propulsion system. The NSTAR engine, shown in Fig. 10, is a 30-cm-diam electron bombardment ion thruster using xenon as propellant. This engine processes a maximum thruster input power of 2.3 kW_e and provides 92 mN of thrust with a specific impulse I_{sp} of 3300 lb_f-s/lb_m (32.3 km/s). The service life requirement of the engine is 8000 hours; a spare engine used to demonstrate a qualification life of 12,000 hours recently completed a 30,000-hour extended life test. Higher-I_{sp}, higher-power, and longer-life ion engines are currently under development for the more demanding needs of future deep-space missions, such as the Project Prometheus proposed NEP Jupiter Icy Moon Orbiter (JIMO) mission.[38]

There has also been considerable interest in ion thrusters in Europe and Japan. For example, AEA Technology at Culham, England, has been developing the UK-10 ion thruster as well as a larger version, the UK-25, as part of the United Kingdom national program directed by the Space Department at the Royal Aircraft Establishment, Farnborough, England. The UK-10 can be used primarily for satellite stationkeeping or possibly attitude control. It is a 10-cm-diam electron bombardment ion thruster that uses xenon as propellant. This thruster has a divergent-field discharge chamber design and employs electromagnets rather than the permanent magnets used in U.S. ion thrusters. At 660 W_e of input power, this thruster provides 25 mN of thrust with a specific impulse of about 3350 lb_f-s/lb_m (32.8 km/s) and a thruster efficiency of about 60%.

The RIT-10 is a RF ion thruster developed by Daimler-Benz Aerospace AG (DASA) in Germany, based on research at the University of Giessen, Germany. It is nominally operated at a power level of 585 W_e and produces a thrust of 15 mN at a specific impulse of 3400 lb_f-s/lb_m (34 km/s). It uses xenon as propellant. The thruster efficiency is about 64%.

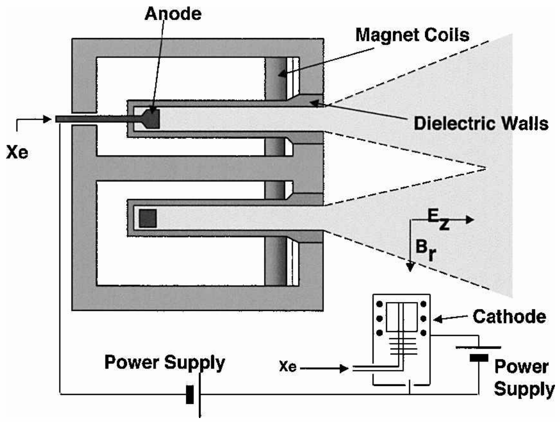


Fig. 11 Hall SPT schematic.

Finally, Mitsubishi has developed a 12-cm-diam, divergent-field xenon ion engine designed to provide north–south stationkeeping. It was flown on the ETS-VI (Engineering Test Satellite).

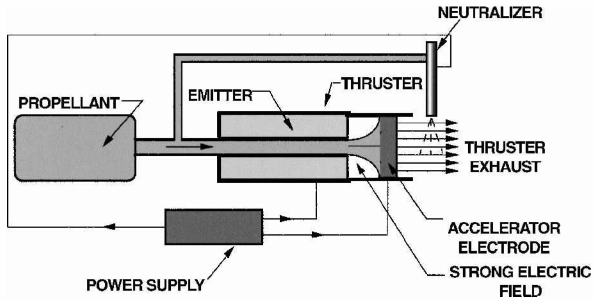
Hall thrusters. As shown in Fig. 11, Hall thrusters are gridless ion engines that produce thrust by electrostatically accelerating plasma ions out of an annular discharge chamber. The concept of a Hall thruster was originally conceived in the United States, but it was only in the former Soviet Union where it was successfully developed into an efficient propulsion device. Two types of Hall thrusters were developed: the stationary plasma thruster (SPT) at Design Bureau FAKEL and the thruster with anode layer (TAL) at the Central Research Institute for Machine Building. These thrusters were introduced to the West in 1992 after a team of electric propulsion specialists, under the support of the Ballistic Missile Defense Organization (BMDO), visited Soviet laboratories and experimentally evaluated the SPT-100 (i.e., a 100-cm-diameter SPT thruster). More recently the T-100 and T-160 stationary plasma thrusters were developed at the Keldysh Research Institute for Thermal Processes, Moscow.

Over 100 SPT thrusters have flown on various Soviet and Russian satellites. The efficient ion production characteristic of Hall thrusters, along with their efficient electrostatic ion acceleration process, enables Hall thrusters to produce an absolutely unique combination of I_{sp} and efficiency (around 50%) for specific impulses in the range of 1500–2500 lb_f-s/lb_m (15–25 km/s). This capability makes the Hall thrusters ideal for near-Earth space missions where this I_{sp} range is optimum.

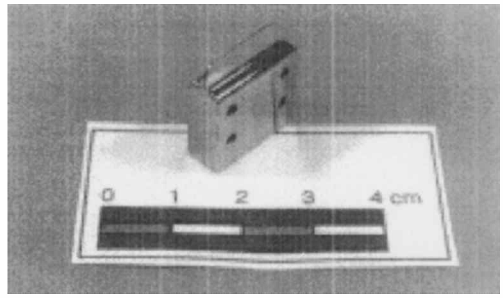
Finally, as mentioned earlier, from a systems-level perspective the relatively low voltage required by these devices greatly simplifies their power processing requirements. For example, Hall thrusters have the potential of operating directly off of the bus dc voltage available from a solar array; by contrast, an ion engine typically requires a dc-ac-dc power inverter to convert the low-voltage dc from the solar arrays to the high-voltage dc (e.g., typically kilovolts dc) required by the accelerator screen.

Field emission electric propulsion thruster. The field emission thruster is any one of a family of devices that uses an electric field to extract atomic ions from the surface of a metal. For propulsion applications the most common source of ions is a metallic liquid. In these sources a strong electric field is established with a pair of closely spaced electrodes. The free surface of liquid metal exposed to this field is distorted into a series of conical protrusions in which the radius of curvature at the apex becomes smaller as the field is increased. When the field reaches a threshold value (which is on the order of 10^6 V/mm for cesium), atoms on the surface of the tip are ionized and eventually removed. They are then accelerated to a high velocity by the same electric field that produced them. Expelled ions are replenished by the flow of liquid propellant in the capillary feed system. A separate neutralizer is required to maintain charge neutrality of the system.

By far the most extensively investigated application of this process for propulsion is represented by the field emission electric propulsion (FEEP) technology that has been under development



a)



b)

Fig. 12 FEEP thruster: a) FEEP schematic and b) Cesium propellant FEEP thruster from Italy. (Picture reproduced with permission of ESA.)

by the European Space Agency (ESA) since the mid-1970s. An example is shown in Fig. 12.

The variety of potential applications for FEEP technology includes small spacecraft attitude control, ultra-high-precision pointing (especially in spacecraft constellations), and proportional thrust throttling for drag compensation. Because of these many potential applications, much recent work has focused on extensive testing of all thruster subsystems. Emerging flight system designs are compact, self-contained units without any external propellant tanks, tubing, or valves.

Colloid thruster. The colloid thruster produces thrust by electrostatically accelerating very fine droplets of an electrically charged, conducting fluid. In the more common configuration the droplets are formed by flowing the liquid through a needle with inner diameter on the order of hundreds of microns. As the liquid exits the needle orifice, a droplet is formed. The needle is biased to a potential of 5–10 kV positive with respect to ground. An accelerating electrode is placed in close proximity to the needle orifice and is biased negatively to a potential of several kilovolts. The electrostatic forces on the charged droplet cause it to break off with a net positive charge. In steady-state operation such a needle would emit a stream of such droplets with a very narrow velocity distribution. Although still in a research phase, colloid thrusters show promise for delivering the small impulse bits required for precision pointing and stationkeeping applications.

Electromagnetic Thrusters

Electromagnetic thrusters use electromagnetic body forces ($\boldsymbol{J} \times \boldsymbol{B}$) to accelerate a propellant plasma. At first glance they have some similarities to electrothermal arcjet and microwave electrothermal thrusters. However, electrothermal thrusters simply use a plasma discharge to add thermal energy to the propellant. By contrast, electromagnetic thrusters use true electromagnetic forces generated in a very high-current (typically thousands of amperes) plasma discharge to accelerate the propellant plasma.

The current flowing in the plasma discharge has two effects. First, it serves to ionize the propellant. Second, and most important, the high current produces an intense magnetic field (much like an electromagnet). It is this magnetic field that then pushes the ions in the plasma out of the engine at high velocity (I_{sp}). Several types of electromagnetic thrusters are discussed next.

Pulsed plasma thruster. The pulsed plasma thruster (PPT) shown in Fig. 13 is a device in which electrical power is used

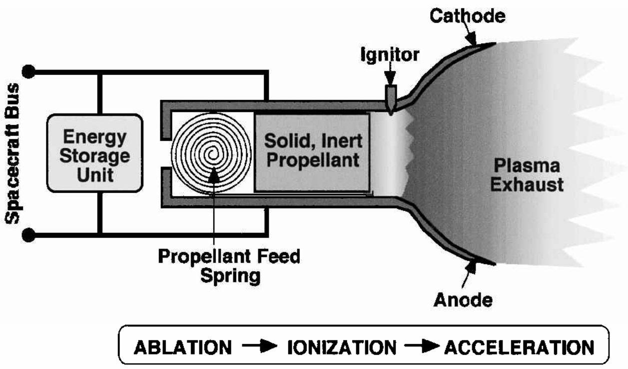


Fig. 13 PPT schematic.

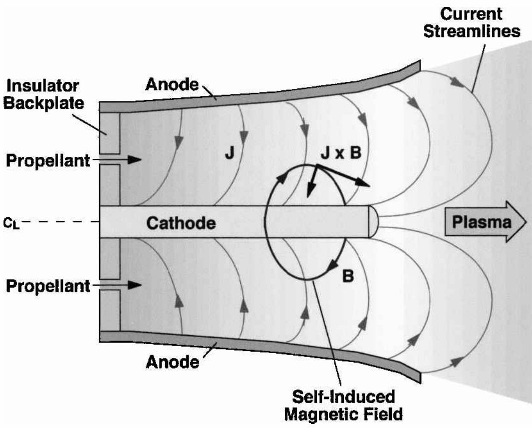


Fig. 14 MPD thruster schematic.

to ablate, ionize, and electromagnetically accelerate atoms and molecules from a block of solid propellant material (e.g., Teflon®). The thrust generated during a single pulse is on the order of tens to hundreds of microNewtons; this low thrust per pulse results in the ability of the PPT to deliver very small impulse bits that are desirable for some precision pointing missions. The Teflon PPT has a long heritage and has demonstrated performance that makes it desirable for a variety of orbit raising, stationkeeping, attitude control, and fine pointing missions.

The remaining electromagnetic thruster concepts in this section are high-power devices (e.g., typically hundreds of kW_e to MW_e per thruster) and are generally considered for use in large, high-powered electric propulsion systems typical of those that could be used for piloted missions. All are currently undergoing research for future applications. The magnetoplasmadynamic (MPD) thruster is the best characterized, followed by the pulsed inductive thruster (PIT), with the variable specific impulse magnetoplasma rocket (VASIMR) thruster the least developed.

MPD thruster. The MPD or Lorentz-force-accelerator thruster has been under investigation since its inception in 1964. This thruster type can be operated in either steady-state or pulsed mode. As shown in Fig. 14, it has an axisymmetric geometry (annular anode surrounding a central cathode) and produces thrust via the Lorentz body force ejecting a high-velocity plasma stream.

During operation, a large current (thousands of amperes) flows between the coaxial electrodes and both ionizes and accelerates the propellant gas. The large current induces a significant azimuthal magnetic field. The magnetic field and the current create a $\boldsymbol{J} \times \boldsymbol{B}$ body force (Lorentz force) that axially accelerates the plasma, providing thrust. This is known as a self-field MPD thruster. The applied-field MPD thruster operates essentially the same way, but an external solenoidal magnetic field is applied to enhance the plasma acceleration process. With an applied field an MPD thruster can operate with a lower discharge current because the applied field can greatly enhance the acceleration mechanism. (The thrust produced by the electrothermal expansion of the propellant is usually insignificant at higher power levels.) In steady-state operation the high-current (kiloamps), low-voltage (100–200 V) arc discharge attaches diffusely to both electrodes. The majority of the plasma current is provided by thermionic emission from the hot cathode (>2500 K).

The MPD thruster can operate on a variety of nonoxidizing propellants. It is capable of providing specific impulse I_{sp} of 1000–12,000 lb_f-s/lb_m (10–120 km/s) (possibly higher) with a peak efficiency of up to 75% (depending on the propellant and power level). Both I_{sp} and efficiency increase as power level increases. Lithium propellant has the best reported performance below 10 MW_e, and hydrogen produces the best performance above 10 MW_e. Gaseous propellants have not yet demonstrated high efficiencies at moderate specific impulses.

Pulsed inductive thruster. The pulsed inductive thruster (PIT) uses a flat induction coil (approximately 1 m diameter) and a fast gas valve to inject a few milligrams of propellant over the coil. Once the gas has been injected, a bank of high-voltage, high-energy storage capacitors is discharged providing a large azimuthal current pulse to the coil. The time-varying electromagnetic field caused by the current pulse ionizes the propellant gas and causes the ionized gas to accelerate away from the coil. Because the energy is inductively coupled into the plasma, the device can be designed so that the plasma has minimal contact with thruster surfaces, resulting in minimal erosion of thruster components.

Another advantage is that the PIT can be operated on a variety of propellants, such as hydrazine, ammonia, argon, and carbon dioxide, and at specific impulses ranging from 1000–6000 lb_f-s/lb_m (10–60 km/s). Typically the demonstrated efficiency ranges between 20–40% below 3000 lb_f-s/lb_m (30 km/s) and between 30–60% in the 3000–8000 lb_f-s/lb_m (30–80 km/s) range.

VASIMR thruster. The VASIMR[39] represents an application to propulsion of RF and microwave heating methods and magnetic confinement technologies originally developed for fusion power research. The device is electrodeless and uses both electrothermal and electromagnetic processes to convert electrical power into directed kinetic energy.

This system, still in the research state, utilizes a cylindrical geometry. Magnetic coils (which in an actual rocket would be superconducting) produce a strong magnetic field that confines and guides a hydrogen plasma (the propellant), insulating it from the material wall.

By controlling the aft magnetic "gate," it might be possible to modulate the effective throat area and hence the thrust. In addition, by controlling the exhaust gas temperature through RF power and the (preionized) hydrogen flow rate the specific impulse can also be adjusted independently of the thrust and power. This ability to vary the thrust and specific impulse independently (and at constant power) enables the performance to be tailored to a specific mission to optimize acceleration, and thus trip time, or payload mass fraction.

Research on this system is ongoing; feasibility issues remaining to be addressed include overall efficiency of converting input electric power into directed thruster jet power, mixing of cold hydrogen with hot plasma to adjust thrust and I_{sp}, and, finally, overall mission benefits compared to more conventional electric propulsion options (e.g., ion thrusters).

Micropropulsion

To reduce mission cost and risk, NASA is currently pursuing the goal of developing microspacecraft, like the one shown in Fig. 15, in various size ranges. Ultimately, the goal is to replace the billion-dollar class "flagship" missions (with their single, unique, multi-ton spacecraft) with missions employing large numbers of small, low-cost microspacecraft. This has the potential not only to reduce cost (in part because of production of multiple copies of similar microspacecraft), but also to reduce mission risk; for example, if one microspacecraft in a flotilla fails, there would still be many replacements available to complete the mission. However, in order to maintain a high degree of mission capability all of the various microspacecraft subsystems will have to decrease significantly in

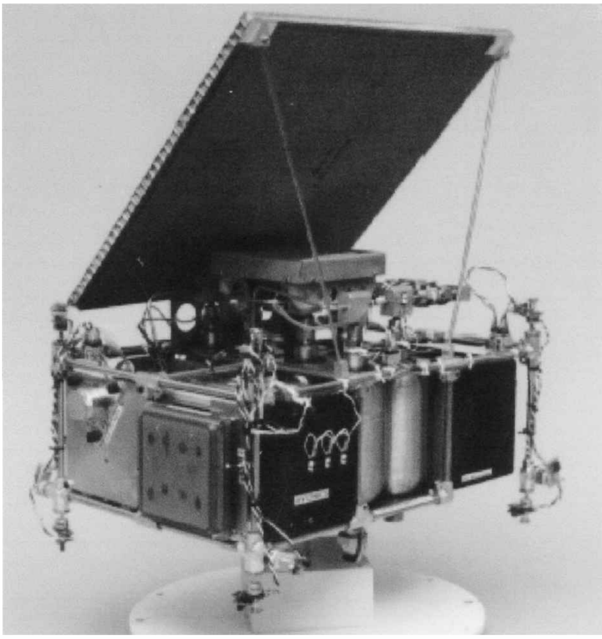


Fig. 15 JPL MTD II deep-space microspacecraft functional model. (Picture reproduced with permission of Ross Jones, JPL.)

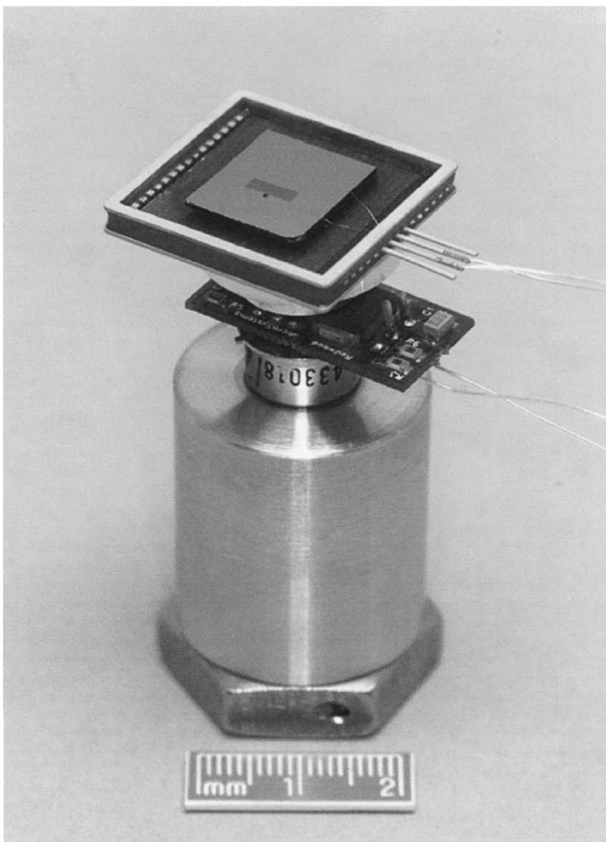


Fig. 16 Subliming solid microthruster. (Picture reproduced with permission of Juergen Mueller, JPL.)

size and weight and be adapted to the unique microspacecraft requirements. For example, research in micropropulsion thruster technology is making use of microelectromechanical systems (MEMS) technologies to fabricate ultrafine nozzle throats in the tens of micrometers size range or below, like that seen in the microthruster prototype shown in Fig. 16, in order to facilitate the required small thrust and impulse bit requirements of both primary (i.e., ΔV maneuver) and attitude control propulsion for the microspacecraft.[40] MEMS-based thruster technology might also be combined with small, but conventionally machined valves, such as solenoid valves, to arrive at MEMS-hybrid thruster versions, or combined with future MEMS-based valves on a single chip, possibly even integrated with the necessary control electronics. This work is still very much in a basic research phase, although the general commercial interest in development of microscale (or even nanoscale) technologies might allow significant cross-fertilization between ground- and space-based applications.

Also, although the preceding discussions have focused on the use of micropropulsion for both primary (ΔV) and attitude control propulsion for microspacecraft, a second major potential application for micropropulsion is for attitude control and stationkeeping for more conventional sized spacecraft that require extremely fine pointing and positioning accuracy. Applications in this category could include future constellations of space-based interferometer-type telescopes that are designed ultimately not just to detect, but actually image Earth-sized planets around stars at distances out to 40 light years. These spacecraft are typically physically separated (by as much as thousands of kilometers) and require ultra-high-precision pointing and position (separation distance) control (e.g., position control to fractions of a wavelength of light) in order to optically combine the light images from multiple spacecraft telescopes. To achieve the required level of pointing angle and position accuracy, they would use a combination of micropropulsion (for coarse control) and electromechanical actuators (for fine control).

Beamed-Energy Propulsion

In beamed-energy propulsion a remote energy source, such as the sun or a ground- or space-based laser or microwave transmitter, transmits power to the vehicle via a beam of electromagnetic radiation (near-visible or microwave wavelengths). There, the beam is collected and used to power the propulsion system. In beamed-energy propulsion there is the potential for significant weight reduction and thus improved performance on the spacecraft, because a heavy power supply (e.g., nuclear reactor) is not carried on the vehicle.

Two different wavelength regions (near-visible [visible and infrared] and microwave) are typically considered. These can then be used directly in a thermal propulsion system or indirectly in an electric propulsion system by first converting the incoming beamed energy into electricity. This results in four general categories of systems, as shown in Table 5 and Fig. 17.

The solar/laser/microwave systems can all be used for orbit-to-orbit in-space applications; these are discussed next. However, only the laser or microwave systems have sufficient power density to allow their use as Earth-to-orbit (ETO) launch systems (discussed later in the section). Interestingly, the beam power requirements for the beamed laser/microwave in-space systems are quite modest (typically 0.1 to 10 MW) (Ref. 41). By contrast, ETO launch systems require very large powers (on the order of 0.1–1 MW of beam power per kilogram of vehicle mass).

Also, solar and near-visible laser systems tend to have very similar vehicle systems and configurations. In fact, a very attractive technology growth path involves development first of the solar thermal (or electric) vehicle, followed by a proof-of-concept demonstration of the laser option using the same (solar) vehicle.

One issue in laser and especially microwave beamed-energy systems is the variation in transmitter and receiver size with wavelength

Table 5 Beamed-energy propulsion concepts

Wavelength	Near-visible (optics diam. ~ 10 m)	Microwave (optics diam. ~1 km)
Direct energy use	Laser thermal propulsion (LTP)[a]	Microwave thermal propulsion (MTP)
Indirect energy use	Laser electric propulsion (LEP)[b]	Microwave electric propulsion (MEP)

[a] Solar analog: solar thermal propulsion (STP).
[b] Solar analog: solar electric propulsion (SEP).

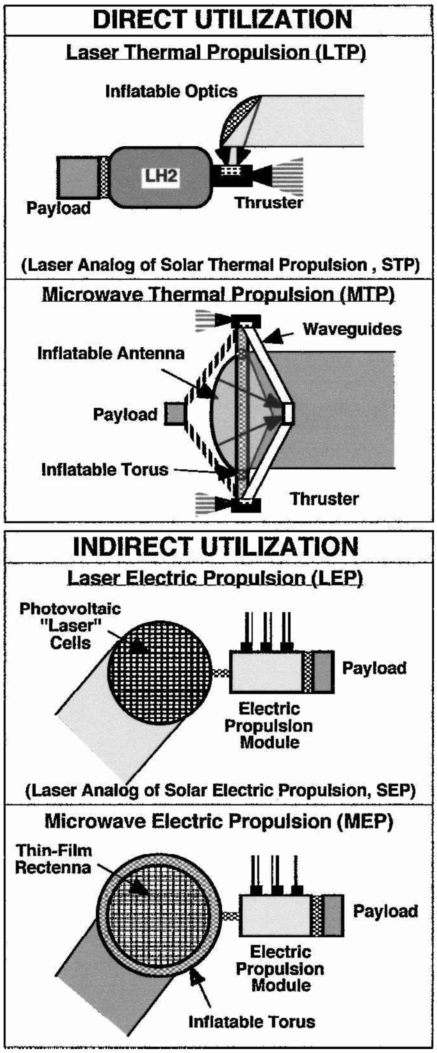


Fig. 17 Beamed energy in-space vehicle concepts.

and beaming distance.[42] Generally, microwave-based systems are limited to beaming distances corresponding to the distance between Earth or low Earth orbit (LEO) and geosynchronous Earth orbit (GEO). In this case a 10-km-diam transmitter would be required for a 1-km-diam receiver for GEO distances at 12.2-cm wavelengths (2.45 GHz). These dimensions represent a practical upper limit to near- or mid-term space-based receivers and ground-based transmitters.

Interestingly, much of the technology required for power beaming through the Earth's atmosphere has already been demonstrated by SDI/BMDO and the astronomical telescope community. This includes adaptive optics transmitters, as well as feedback technologies to compensate for atmospheric turbulence and thermal blooming (which causes beam defocusing).

In-Space Applications

In both Solar Thermal Propulsion (STP) and Laser Thermal Propulsion (LTP), sunlight or visible/infrared (VIS/IR) laser light is focused into a thruster to heat a propellant such as hydrogen. Because the beam spot intensity is higher in laser thermal propulsion than solar thermal propulsion, it is possible to couple the beam energy directly into the propellant to permit a higher I_{sp} than that from solar thermal propulsion. These rockets could provide performance similar to that of nuclear rockets in terms of I_{sp}, with thrust intermediate between that of the high- and low-thrust propulsion systems. For example, a solar-thermal rocket would have an I_{sp} of 800–1000 lb_f-s/lb_m (8–10 km/s) and a T/W of 10^{-2} to 10^{-3} for a 20-day LEO-to-GEO trip time. For comparison, a laser thermal system might reach an I_{sp} of 1500–2500 lb_f-s/lb_m (15–25 km/s) using inverse Bremsstrahlung coupling. Thus, both solar thermal and laser thermal propulsion systems represent medium-high I_{sp}, medium-thrust propulsion systems that fill a mission niche between fast but heavy chemical propulsion, and slow but very fuel-efficient (i.e., higher I_{sp}) electric propulsion. Both the Air Force Phillips Laboratory and the NASA MSFC are developing the technology for a demonstration flight of a solar thermal propulsion system.[43]

Microwave thermal propulsion (MTP) is the microwave analog to laser thermal propulsion. One type of MTP thruster, the microwave electrothermal thruster (MET) discussed in the Electric Propulsion section, is an analog of the LTP thruster in that microwave energy is focused into a thruster to excite and heat a propellant. However, a different microwave energy coupling mechanism can be employed, involving electron–cyclotron resonance (ECR) or ion–cyclotron resonance (ICR) heating; strictly speaking, these are not thermal systems because the microwave energy is coupled directly to the propellant.

Laser electric propulsion (LEP) is the laser analog to solar electric propulsion (SEP), in which sunlight is converted into electricity by photovoltaic cells and the electricity used to power electric propulsion thrusters. In LEP the solar photovoltaic cell array, which is doped to maximize its efficiency at the laser's wavelength (e.g., 0.85 μm for gallium arsenide cells), is illuminated with laser light. This makes it possible to have an efficiency roughly double that of the corresponding solar cell. Also, the laser beam can have a much higher intensity than that of sunlight at 1 astronautical unit (AU), thus resulting in an effectively lower solar array specific mass.

The final system is microwave electric propulsion (MEP). In this concept a rectenna (rectifying antenna) is used to convert microwaves to electricity (with an efficiency around 90%), which is then used to power electric thrusters as in an SEP/LEP system.

Earth-to-Orbit Beamed Energy Propulsion

Near-visible (VIS/IR) and microwave beamed-energy powered launch vehicles have been studied extensively by government and university researchers.[44] The basic propulsion concept involves generating the laser or microwave beam at the transmission station (ground or space based), beaming the energy to the vehicle, and using the energy to heat a propellant working fluid to produce thrust. Various combinations of propellants (airbreathing or onboard liquid or ablated solid) are possible. For example, laser-supported combustion could be used to heat air; a small amount of onboard propellant would then be used for final orbit insertion upon exit from the atmosphere.

Finally, microwave-powered vehicles can also make use of indirect thruster modes (in addition to the microwave analog of laser supported combustion modes) by using lightweight rectennas for beam-to-electricity power conversion.

The Air Force Phillips Laboratory, with additional support from NASA, has been conducting a series of proof-of-concept experiments to demonstrate the feasibility of airbreathing Earth-to-orbit laser propulsion. Because of the availability of only modest laser power levels, only small, simple vehicle designs can be tested. In 2001, open-air free-flight tests of a 12.2-cm-diam, spin-stabilized vehicle reached an altitude of 71 m (233 ft) (Ref. 45).

Systems/Infrastructure Issues

Beamed-energy propulsion systems attempt to lower space operations costs by placing the complex and massive parts of the propulsion system on the ground (or in orbit) for easy construction, supply, repair, etc. Although there are no intrinsic technological "show stoppers" to beamed-energy propulsion, there are serious issues associated with development and infrastructure costs. This is because of the high beam power levels (e.g., many GW required for launching a vehicle from the surface of the Earth). Thus a similar situation is found to that of the launch-assist catapult or space elevator (tether) concepts discussed below, where a potentially very expensive infrastructure must be amortized over many launches to be attractive.

One way to amortize this infrastructure that is unique to beamed-energy systems is that they can supply many users. For example, a beamed-energy system could be envisioned filling a capacity like that of a terrestrial power grid. Power could be supplied to high-T/W Earth or Moon launch vehicles, orbit-to-orbit or Earth-orbit escape low- or high-T/W vehicles, and lunar base power needs, thus broadening the scope of the user base over which the infrastructure is amortized. Finally, VIS/IR beamed-energy orbit transfer vehicles share many technologies with their solar-thermal propulsion counterparts (e.g., inflatable optics, thrusters, cryogenic H_2 storage and feed systems, etc.). This suggests a potential technology investment strategy starting first with demonstration of solar-thermal propulsion orbit transfer vehicles, followed next with development of MW-class lasers for laser-thermal orbit transfer vehicles, and concluding with development of GW-class laser or microwave systems for Earth-to-orbit launch vehicles.

Beamed-Momentum Propulsion

In beamed-momentum propulsion the momentum carried by a stream of particles (e.g., photons or charged particles) is used to push the vehicle; in effect, the stream of particles become the propellant that supplies the momentum to move the spacecraft. This is in contrast to a beamed-energy system, where the beamed energy (sunlight, laser/microwave beam) provides thermal energy (or indirectly electricity) that is used to energize onboard propellant. Thus, a beamed-momentum propulsion system represents an example of a propellantless propulsion system, with both the energy and propellant system taken off of the vehicle.

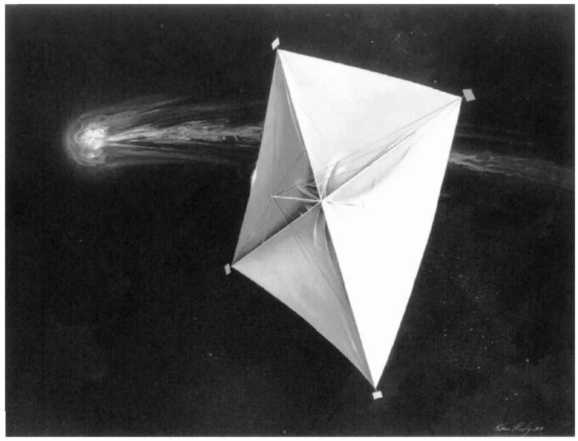
Two general types of beamed-momentum systems are considered: those that use momentum exchange between photons (solar/laser/microwave sails) and a reflective sheet or sail, and those that use momentum exchange between charged particles and an electromagnetic field (electromagnetic sails).

Solar Sails

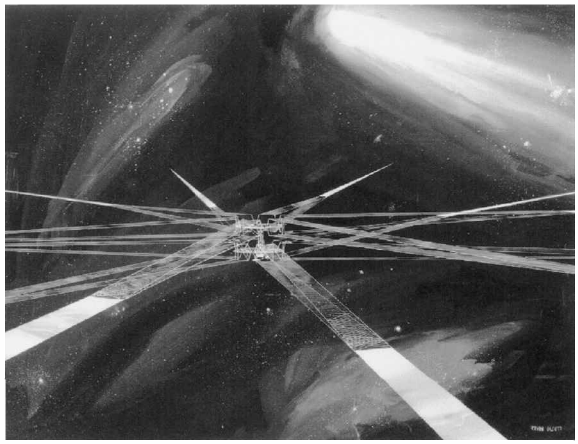
A solar sail is a propulsion concept that makes use of a flat surface of very thin reflective material supported by a lightweight deployable structure.[46] As shown in Fig. 18, there are several types of solar-sail implementations that have been considered; these include different attitude control options (three-axis vs spinning), different geometries (square vs circular disk vs rectangular blades), and structures (deployable booms vs inflatable structures). Solar sails accelerate under the pressure from solar radiation (essentially a momentum transfer from reflected solar photons), thus requiring no propellant. Attitude control can be accomplished by steering vanes or by placing the payload on an articulated boom (for center-of-mass vs center-of-pressure yaw and pitch control). Because a solar sail uses no propellant, it has an effectively infinite specific impulse; however, the T/W is very low, 10^{-4} to 10^{-5} for the 9 N/km^2 (5.2 lb$_f$/mile^2) solar pressure at Earth's distance from the sun, resulting in the potential for long trip times in and out of planetary gravity wells.

Solar sails can substantially reduce overall trip time and Earth-launch mass for high-ΔV robotic missions in comparison to conventional chemical propulsion systems. Solar sails have also been shown to have a potential benefit for use in interstellar precursor missions. For interplanetary cargo missions (e.g., to Mars), substantial reductions in launch mass requirements are possible in comparison to conventional chemical systems, although trip times can be longer. As cargo haulers (solar system supertankers), solar sails can provide potentially significant cost savings because they are essentially reusable as is and do not require costly refueling for new missions.

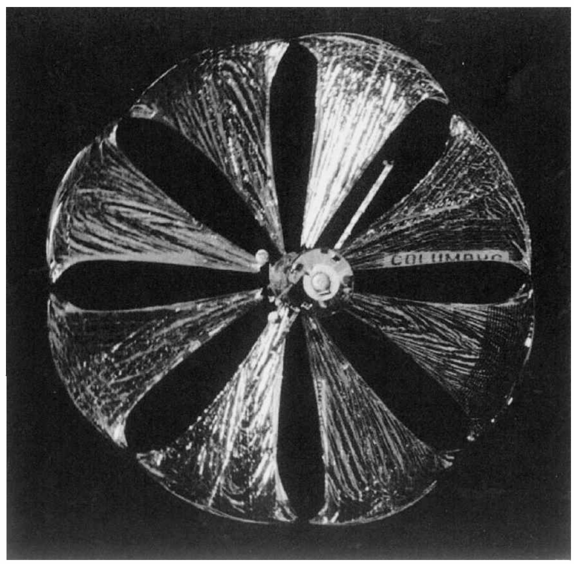
Many studies have indicated that the most important next step for development of solar sails is the launch and deployment of a small experimental sail. There have been no operational solar sail tests of yet, but a spinning disk-shaped sail structure (Znamya) was deployed in space by the Russians from a Progress tanker after its resupply mission was completed. Interestingly, the relative low cost of solar sails, as compared to chemical propulsion stages, makes it possible for universities and private organizations to construct sails for testing. For example, the Planetary Society attempted to launch a solar sail on a Russian commercial launch vehicle in 2001, but the sail was unable to successfully deploy and operate because of a failure in the launch vehicle. A reflight is scheduled for 2003 (Ref. 47).



a) Square sail (Picture reproduced with permission of NASA)



b) Heliogyro sail (Picture reproduced with permission of NASA)



c) Russian Znamya sail (Picture reproduced with permission of Charles Garner, JPL)

Fig. 18 Solar-sail examples.

Beamed-Momentum (Laser/Microwave) Light Sails

One important limitation in solar sails is the $1/R^2$ drop in sunlight intensity as one moves out of the solar system. Nevertheless, solar sails can be used for deep space or interstellar precursor missions by first spiraling in close to the sun (e.g., to 0.10 to 0.25 AU) and using the increased sunlight pressure to drive them out of the solar system. It is also possible to perform interstellar missions with a laser-driven light sail. This concept[48] is uniquely suited to interstellar missions because it is one of the few ways that sufficient energy (per unit mass) can be imparted to a vehicle to achieve the high velocities (>0.1c) required for interstellar missions. This is possible because the spacecraft engine (lasers) is left back in Earth's solar system; a somewhat arbitrarily large amount of energy (number of photon's per unit of sail mass) can be imparted to the vehicle's propellant (photons) to accelerate the vehicle. (In fact, input power is ultimately limited by the imperfect reflectivity of the receiver optics; solar or laser light absorbed by the receiver material must be radiated so the maximum power that can be received is a function of the material reflectivity, emissivity, and maximum temperature limits.)

Note however that for interstellar distances, very large optics and laser power levels are required. For example, a laser operating at 1-μm wavelength requires a transmitter lens with a diameter of 1000 km to illuminate a 1000-km-diameter receiver (sail) at 40 light years (LY). Similarly, a very high power level (and ultralight sail) is required for reasonable acceleration (typically 0.036 g for flybys to 0.2 g for rendezvous) of the vehicle. For example, the laser power required for a robotic flyby mission to 4.3 LY with a maximum cruise velocity of 0.4c is 14 terawatts (TW), which is comparable to the average power produced by all of human civilization. However, any interstellar mission, regardless of the propulsion system, will require high power levels to achieve the high speeds required. Even today we achieve nontrivial propulsion power levels for ambitious space missions; for example, the Saturn V rocket generated a power on liftoff corresponding to about 0.8% of humanity's total power output in 1969.

The microwave sail (Starwisp)[49] concept is the microwave analog to the laser LightSail. This approach has the advantage that the vehicle can be made ultralightweight for robotic interstellar mission flybys, thereby reducing both the transmitter power requirements and the size of the transmitter optics (because the microwave sail can be accelerated at high g to its final coast velocity while still relatively near the Earth). To achieve this low mass, the sail consists of wire mesh with holes in the mesh less than $\frac{1}{2}$ the wavelength of the microwaves. Under these conditions the sail acts like a solid sheet with respect to the incoming microwave photons. (A related concept, the "perforated" solar/light sail, has also been proposed for visible-light sails.)

Ultimately, beamed-momentum light sails represent a major development challenge, both because of the extraordinarily demanding technologies and because of the extraordinarily large scale of the systems. Nevertheless, they do represent one of the few ways to perform interstellar missions with reasonable trip times.

Electromagnetic Sails

In electromagnetic (EM) sails charged particles (mostly protons) from the solar wind are reflected by a magnetic field, analogous to the reflection of solar photons off of a solar sail's reflective sheet. Thus, EM sails are the charged-particle analogs of solar sails. Two examples of EM sails are shown in Fig. 19. In principal, a solar-wind sail could be built using a physical sheet of material, but the momentum per unit area carried by the solar wind is so much less than that from photons as to require an impossibly lightweight sheet; instead, a (massless) magnetic field, tens to hundreds of kilometers in diameter, substitutes for the solar-sails sheet. Interestingly, EM sails provide many of the same potential benefits as solar sails and have some of the same drawbacks; for example, sunlight intensity and solar-wind density both drop off as the square of the distance ($1/R^2$) from the sun. (The solar wind maintains a roughly constant velocity of around 300–800 km/s throughout the solar system, but the momentum force decreases because of the expansion of the solar wind and thus increasing dilution of individual particles at increasing distance from the sun.)

One significant feasibility issue with EM sails is their ability to reflect the radially outward flowing solar wind to produce radial and tangential thrust. Because the magnetic bubble that reflects the

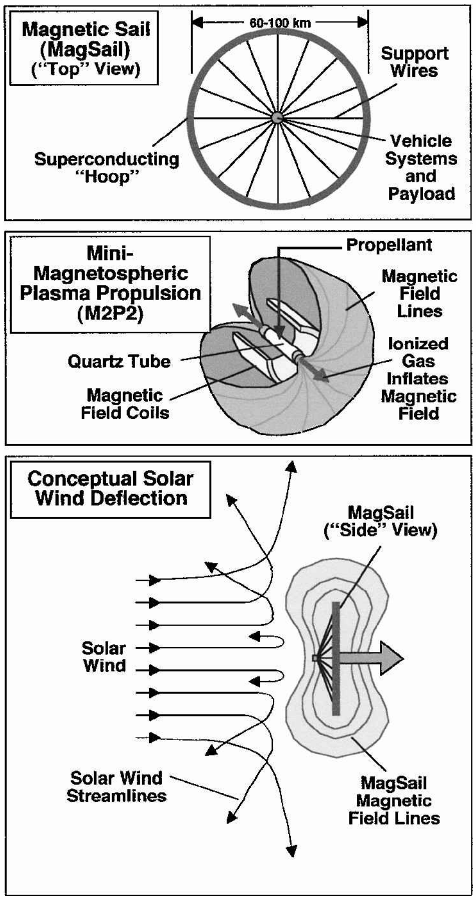


Fig. 19 Electromagnetic sail concepts.

solar wind is generally highly symmetric, it might be difficult to generate adequate tangential thrust (like that produced by tilting a solar sail), thus making it potentially more difficult to maneuver an EM sail into a planetary rendezvous orbit or to move inward toward the sun. However, EM sails would be ideally suited for outer-plane flybys or interstellar precursor missions because they efficiently utilize the radial outward force from the solar wind. Furthermore, even a planetary rendezvous mission can be performed by using an EM sail to leave Earth and a second, separate propulsion system (especially one using aerobraking with a planet's atmosphere or, potentially, magnetobraking with a planet's magnetosphere) used to perform orbit insertion at the target planet (e.g., Mars, Jupiter, Saturn, Uranus, Neptune, but not Pluto for aerobraking/magnetobraking). However, although there is significant performance potential for EM sails there remain many unresolved feasibility issues relating to the basic physics of interaction of the solar wind with the magnetic bubble, as well as systems-level considerations for implementation in a propulsion system (e.g., conventional vs superconducting magnet).

Magnetic Sail (MagSail)

The first proposed EM sail was the magnetic sail, or MagSail, concept.[50] The MagSail consists of a cable of superconducting material, millimeters in diameter, which forms a hoop that is tens to hundreds of kilometers in diameter. The current loop creates a magnetic dipole that diverts the background flow of solar wind. This deflection produces a drag force on the MagSail radially outward from the sun. In addition, proper orientation of the dipole can produce a lift force that could provide thrust perpendicular to the radial drag force.

Minimagnetospheric Plasma Propulsion

A newer concept, the minimagnetospheric plasma propulsion (M2P2) sail, was inspired by research in planetary magnetospheres.[51] These magnetospheres, around planets like Earth

Table 6 Comparison between electromagnetic sails and solar sails

System	M2P2	MagSail	Solar Sail
Thrust (N/km^2)[a]	0.001	0.001	9
Hardware dimensions	Small (~few m)	Large (60–100 km diam.)	Medium (0.1-few km)
Mass (dry)	Small	Large (~100 MT)	Medium
Propellant use	0.25 kg/day per N thrust (I_{sp} = 35,000 lb_f-s/lb_m-343 km/s)	None	None
Min. Earth ops altitude	Heliocentric (LEO?)	Heliocentric	1000–2000 km
Acceleration	Constant (disk inflates as $1/R^2$)	$1/R^2$ (Fixed size)	$1/R^2$ (Fixed size)
Electric power system	Yes (1 kW_e per N thrust)	Yes (for startup only)	No
Tangential thrust	Limited	Limited	Yes

[a]Ideal thrust per unit area of magnetic barrier or wall (i.e., cross section of magnetic bubble), or solar-sail sheet.

and Jupiter, are large magnetic bubbles caused by trapped ions (plasma) inflating the naturally occurring magnetic fields around those planets that possess a permanent magnetic field. For example, the magnetosphere around Jupiter reaches as far as several hundred times the planet's radius.

The M2P2 sail uses an artificially generated minimagnetosphere that is supported by magnets on the spacecraft and is inflated by the injection of low-energy plasma into the magnets. (Thus, M2P2 is not, strictly speaking, a true propellantless propulsion system; however, the amount of propellant needed to produce the plasma is small, resulting in an effective I_{sp} of 35,000 lb_f-s/lb_m [343 km/s].) This plasma injection allows the deployment of the magnetic field in space over large distances (comparable to those of the MagSail) with field strengths that can be achieved with existing technology (i.e., conventional electromagnets or even permanent magnets). Additionally, one potentially significant benefit of the M2P2 sail is the small size of the physical hardware (even though the magnetic bubble is very large); this eliminates the need for the deployment of large mechanical structures that are presently envisaged for MagSails or solar sails.

Finally, one important unique capability of the M2P2 is its ability to provide constant thrust (at least within the solar system); by contrast, the thrust produced by sunlight or solar wind for both solar sails and MagSails decreases as $1/R^2$. In the case of the M2P2, as the density of the solar wind decreases (as the vehicle moves away from the sun), the M2P2 magnetic bubble increases in size, so that the two effects cancel each other out to produce constant thrust independent of distance from the sun. (This is not possible for solar sails or MagSails because of their fixed, finite physical size.)

Particle Beam Drivers and MagOrion

Although the MagSail and M2P2 were originally envisioned for use with the solar wind, it would also be possible to use particle accelerators to fire a beam of charged particles at the EM sail, in much the same way that photons are employed in a laser/microwave sail.[52] Another option would be to use the charged particles produced by a nuclear explosion. This would be the EM sail analog of the Orion nuclear pulse concept, with an electromagnetic (rather than physical) pusher plate, hence the name Magnetic Orion or MagOrion.[53] Interestingly, in the original Orion concept some of the material of the pusher plate evaporates (ablates) with each pulse. This serves to both to keep the pusher plate cool and to add additional propellant mass. The result is an increase in thrust, although at the expense of some specific impulse I_{sp}. By contrast, there is no mass in the MagOrion's magnetic pusher plate, and so the effective I_{sp} (15,000–45,000 lb_f-s/lb_m [150–450 km/s]) makes this concept well suited for interstellar precursor missions.

Comparison Between Electromagnetic Sails and Solar Sails

Solar and EM sails have different advantages and disadvantages and different potential areas of application. Table 6 lists some general characteristics of the different systems.

Note that one reason for the large size of the MagSail's magnet loop is that a simple magnetic dipole's (i.e., MagSail's) magnetic field drops off as the cube of the distance ($1/R^3$). Thus, a large physical loop is needed to produce a large magnetic bubble or wall for reflecting the solar wind. By contrast, in a planetary magnetosphere (and, in theory, M2P2) the magnetic field drops off only linearly with distance ($1/R$) as a result of injection of plasma into the field. In this case the M2P2 can be a physically small device and still project a significant magnetic field strength at large distances.

Aero/Gravity Assist

Aero-, gravity-, and aerogravity-assist maneuvers represent a propellantless method of supplying ΔV for a variety of space missions. For example, aeroassist employs aerodynamic forces, rather than propulsive maneuvers, to minimize the propulsion required for a variety of missions to bodies with atmospheres. Gravity assist uses gravitational interactions between a spacecraft and a planet to transfer some of the planet's orbital momentum to the spacecraft. Finally, aerogravity assist uses aerodynamic flight through a planet's atmosphere as a means of increasing the effectiveness of a gravity-assist maneuver.

Aeroassist

Aeroassist is a broad term that represents a wide range of applications for the use of aerodynamic vehicles in space exploration. The key point is to use atmospheric forces (drag and/or lift) in the planetary atmosphere of interest to create a preplanned behavior of an aerodynamic space vehicle. This technique of using a planet's atmosphere can provide for aeromaneuvering to a specific landing site, as with the Space Shuttle, as well as deceleration, as in the cases of aerobraking and aerocapture, as shown in Fig. 20. A related concept, aerogravity assist, can provide acceleration by combining aerodynamic lift with gravity assist. All aeroassist or aerogravity assist maneuvers can also be used for orbit plane changes (although the plane change capabilities of aerobraking are limited). Finally, as the aeroassist maneuver becomes more demanding thermal protection becomes more challenging. Also, guidance, navigation, and control algorithms for energy management during the aeroassist have yet to be demonstrated for high-speed planetary applications.

Aerobraking

In aerobraking a spacecraft in a high orbit, like GEO, makes a propulsive burn into a new elliptical orbit whose low point (perigee for Earth or periapsis for a generic body) is inside the atmosphere. Air drag at perigee reduces the velocity so that the high point of the elliptical orbit (apogee or apoapsis) is lowered. One or more passes through the atmosphere reduce the apogee to the desired altitude at which point a propulsive burn is made at the new orbit's apogee so as to raise the new elliptical orbit's perigee up out of the atmosphere and circularize the orbit. Generally, the time of flight in the atmosphere is limited, and the total heat flux and peak temperatures are not too extreme. Usually, a dedicated aeroshell is required for high-speed aeroassist involving large orbit changes; however, small orbit changes can be accomplished without a dedicated heat shield. This has been demonstrated by the Magellan spacecraft at Venus and the Mars Global Surveyor at Mars to circularize and lower an

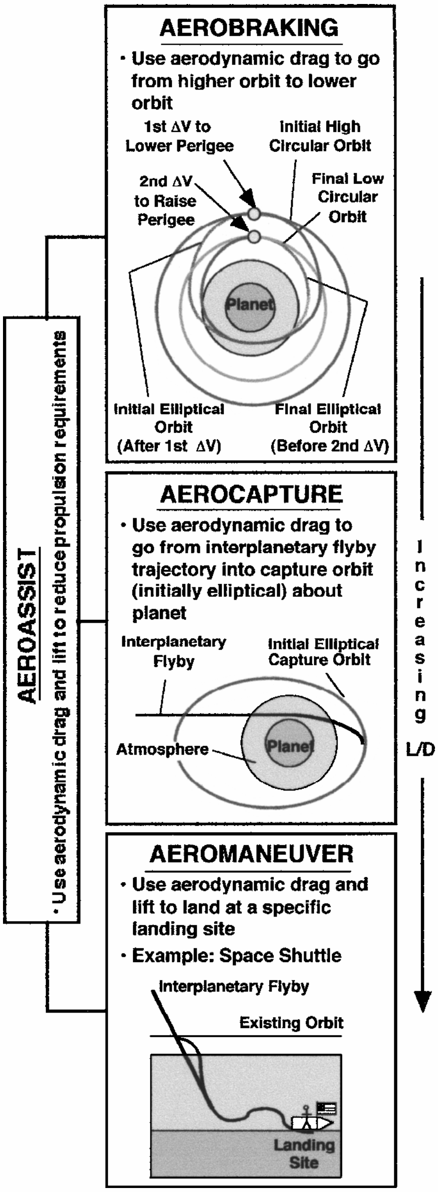


Fig. 20 Aeroassist concepts.

initially high elliptical orbit. In this case the aerobraking surfaces were the spacecraft itself and its solar arrays. No special coating or thermal protection systems were added to the spacecraft, although the spacecraft were configured before atmospheric entry to have an aerodynamically stable shape.

Aerobraking (as well as aerocapture) is an extremely powerful technique for reducing the propulsive requirements of a mission. For example, it requires a ΔV of 4.3 km/s to go from LEO to GEO (or GEO to LEO) using propulsion only. A return trip from GEO to LEO using aerobraking would only require a ΔV of 2.0 km/s; aerobraking saves 2.3 km/s in propulsive ΔV. Any reduction in propulsive ΔV can result in a large decrease in the weight of the propulsion system (propellant, tanks, etc.). This decrease in propulsion system weight can more than compensate for the added weight of the aerobraking system. Thus, an overall increase in the amount of payload that can be delivered is possible using aerobraking, as compared to an all-propulsive system.

Aerocapture

Aerocapture is similar to aerobraking, with the distinction that aerocapture is employed to reduce the velocity of a spacecraft flying by a planet so as to place the spacecraft into orbit about the planet with one atmospheric pass only. This technique is very attractive for planetary orbiters because it permits spacecraft to be launched from Earth at high speed, to give a short trip time, and then reduce the speed by aerodynamic drag at the target planet. Without aerocapture a large propulsion system would be needed on the spacecraft to perform the same reduction of velocity, thus reducing the amount of delivered payload. Aerocapture is also attractive when combined with high-performance solar-powered propulsion systems (e.g., SEP, solar sails, etc.). For example, a SEP system could be used to build up speed in the inner solar system (where sunlight is plentiful) to inject the spacecraft on a fast trajectory to the outer solar system; the SEP system would then be jettisoned and aerocapture used for orbit insertion at the target.

The aerocapture maneuver begins with a shallow approach angle to the planet, followed by a descent to relatively dense layers of the atmosphere. Once most of the needed deceleration is reached, the vehicle maneuvers to exit the atmosphere. To account for the inaccuracies of the atmospheric entering conditions and for the atmospheric uncertainties, the vehicle needs to have guidance and control as well as maneuvering capabilities. Most of the maneuvering is done using the lift vector that the vehicle's aerodynamic shape (i.e., lift-to-drag ratio [L/D]) provides. Upon exit, the heat shield is jettisoned to minimize heat soak, and a short propellant burn is accomplished to raise the orbit periapsis. The entire operation requires the vehicle to operate autonomously while in the planet's atmosphere. Generally, because aerocapture entry velocities are very high, the integrated heat loads are fairly high (usually higher than a direct entry and landing). This sets new requirements on the thermal protection system and causes it to be slightly more massive than for a regular direct entry.

Gravity Assist

Gravity assist is a propellant-free maneuver that is routinely used to accelerate (or decelerate) a spacecraft in order to shorten trip times. Instead of using large propulsive maneuvers to supply the required ΔV, gravity assist uses the gravitational field of planets to increase or decrease the orbit's energy.

On a planetary scale the spacecraft makes a hyperbolic trajectory around the planet. At an infinite distance from the planet (at the edges of its sphere of influence), the spacecraft has a velocity V_{inf} that has the same magnitude (relative to the planet) at arrival and at departure. Only its direction will be changed. Thus, on a planetary scale the spacecraft does not gain anything. However, on a heliocentric (solar system) scale the velocity of the planet has to be added to the velocity of the spacecraft, and because the direction of the velocity vector on a planetary scale has changed the resultant velocity vector on a heliocentric scale will be changed. (It can either be decreased or increased.) Note however that the total energy of the system of the spacecraft + planet remains the same. The spacecraft has accelerated, and the planet decelerated. Because the planet is so much heavier than the spacecraft, the deceleration of the planet is infinitesimally small.

The amount of change in velocity on a heliocentric scale is related to the amount of deflection of the spacecraft's trajectory on a planetary scale. This deflection is mainly dependent on the spacecraft arrival conditions and on the planet's gravitational field. A strong gravitational field will deflect the trajectory more than a weak one.

Aerogravity Assist

Aerogravity assist involves the same concept as gravity assist except that it involves the use of a planet's atmosphere. With large planets, such as Jupiter, gravity-assist maneuvers are very efficient because of the high gravitational field of the planet and therefore to the high turning angle (~90 deg) that they can provide. The increase in velocity during a gravity assist maneuver is related to the turning angle (amount of bending) or gravitational field of the planet. For planets with small gravitational fields, like Venus, Mars, or Earth, a way to increase this turning angle is to use their atmospheres and the lifting capabilities of the vehicle. A lift vector turned downward (pointing toward the planet) during the atmospheric flight will tilt the vehicle's trajectory toward the planet and therefore increase the overall angle that the vehicle has turned on a planetary scale.

The amount of angular deflection is directly proportional to the vehicle's L/D. Studies of aerogravity assist typically require L/D on the order of 10, values typically provided by a class of specialized aerovehicles called waveriders.[54]

Waveriders are high lift, low-drag, sharp-edged vehicles, for which high-temperature-resistant materials are critical. This technology is much further from reality than aeroassist. It is however a far-reaching capability that could have significant impact on the mass and/or time of flight for distant missions.

Launch-Assist Catapults

The concepts described in this section attempt to lower the cost of access to space by using a system that has a large, fixed infrastructure component that is ground- or space-based (for easy construction, supply, repair, etc.) combined with a minimal expendable (or reusable) propulsion system on the spacecraft. The basic approach is to provide most or all of the required mission (e.g., launch) velocity with the fixed system, leaving only a minimal requirement for propulsion on the spacecraft. The systems discussed next include a variety of chemical and electromagnetic catapults that can be used to launch spacecraft from the ground (e.g., from the surface of the Earth, Moon, etc.) or from orbit (e.g., LEO) or that can be used as an onboard propulsion system by catapulting propellant reaction mass out of the catapult thruster.

The most famous literary example of a launch assist catapult is Jules Verne's use of a 900-ft-long cannon to launch a piloted lunar vehicle in the classic *From the Earth to the Moon* (1865), although for the sake of the story it was necessary to assume that the crew could survive the roughly 20,000-g acceleration of the launch. In fact, launch acceleration is an important discriminating figure of merit for these concepts when used as launchers because human-occupied payloads necessarily limit the acceptable launch loads to around 3 g. Similarly, the ability to be scaled to large vehicle and payload sizes (e.g., payloads on the order of tens of tons to LEO) is also an important discriminator between the various launch-assist catapult concepts.

Types of Launch-Assist Catapults

The concepts considered here include both those that make use of chemical combustion or physical compression to produce a high-pressure gas that pushes a projectile down a tube or barrel, and those that employ electromagnetic forces to accelerate a "sled" or carrier that contains the vehicle. Typically, the chemical systems are restricted to Earth-launch applications because of the need to supply the combustion/pressurant gas; by contrast, the electromagnetic systems can be used as Earth- or space-based launchers or as onboard thrusters because electromagnetic forces are used to accelerate the payload or reaction mass. The various systems are summarized in Table 7 and Fig. 21.

Chemical Systems

The chemical catapult systems include cannons (in which an initial charge of chemical propellant is ignited to produce a high-pressure gas which expands in the gun barrel to accelerate the projectile down the length of the barrel),[55] light gas guns (in which a high-pressure gas is sequentially forced into the gun barrel as the projectile moves down the barrel),[56] and ram accelerators (in which a lightweight barrel or tube is filled with combustible gasses that are compressed and burned behind the projectile as it moves down the tube in a manner analogous to the operation of a ramjet).[57] Note that the canon is inherently limited to high-acceleration, small-payload projectiles; by contrast, the other systems are in principle scalable to longer lengths (to reduce acceleration) and larger projectiles (payloads). The cannon represents the most extensively developed of the chemical systems; for example, the High Altitude Research Program (HARP) canon of the 1960s was aimed at developing "gun" (cannon) launch into space; the record for gun launch to space was achieved at the U.S. Army Yuma Proving Grounds (with a 16-in. gun) with an 85-kg (185-lb) projectile fired to an altitude of 180 km (112 miles).[55] However, significant development would be required for all of the chemical systems in order to achieve the muzzle velocities (around 11 km/s) required for injection of payloads directly into Earth orbit.

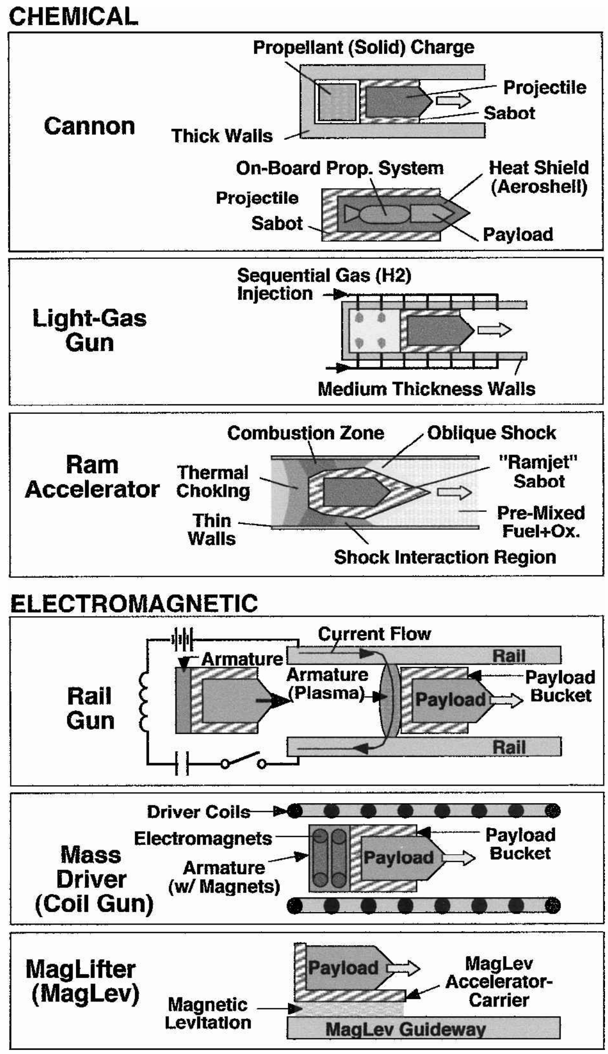


Fig. 21 Launch-assist catapult concepts.

Table 7 Summary of launch-assist catapults

Type	Earth launch only	EP-thruster option	Low-accel. option
Chemical			
Cannon	X	——	——
Light gas gun	X	——	X
Ram accelerator	X	——	X
Electromagnetic			
Rail gun	——	X	——
Mass driver	——	X	X
MagLifter	——	X	X

Electromagnetic Systems

The electromagnetic launch systems include electromagnetic guns such as the rail gun[58] and mass driver (coil gun)[59] and the launch-assist catapult magnetic levitation (MagLifter) launcher.[60] As with the canon, the rail gun is inherently limited to high-acceleration, small-payload projectiles; by contrast, the mass driver and MagLifter are scalable to longer lengths and larger projectiles (payloads). In particular, the MagLifter would have a modest length and acceleration because it only needs to reach speeds up to just under Mach 1, at which point a single stage to orbit (SSTO) launch vehicle is released to fly the rest of the way to orbit. This might be an especially attractive implementation of an SSTO launch system because the MagLifter provides for a significant amount of launch ΔV in the lower atmosphere, as well as injecting the SSTO vehicle on an optimum flight-path angle (e.g., 45–55-deg elevation) at a high altitude (as the SSTO vehicle leaves the barrel of the MagLifter). Because the demands on the MagLifter are modest compared to already demonstrated magnetic levitation (MagLev) railroad systems (e.g., only a roughly two-fold increase in speed), the MagLifter can represent a near-term application of electromagnetic launch. By contrast, major feasibility issues remain unresolved for the rail gun (rail erosion/lifetime, efficiency, energy storage) and mass driver (system power switching, ultimate muzzle velocity).

Applications

Broadly speaking, these systems are interesting because of their potential for use as launch catapults from Earth, the Moon, orbit,[58] or other places. They have the potential for order-of-magnitude cost reductions per kilogram launched (if the launch rate is high enough) over current launch capabilities, and they might enable more frequent launches from the same location. Launch personnel workloads will be decreased, and space on conventional launch vehicles will be opened up for payloads that require more attention than the "dumb," acceleration-insensitive payloads envisioned for these launch catapult systems. Note however that, like the laser/microwave beamed-energy systems, there remains the issue of the cost (and amortization) of the launch-assist catapult system infrastructure and its impact on final operations costs.

Electromagnetic catapults are also interesting because of their potential for use as reaction engines in a solar or nuclear electric propulsion (SEP or NEP) system. In this configuration the accelerated mass becomes the reaction mass of the rocket engine with performance similar to those of other electric propulsion concepts (e.g., I_{sp} of 800–1500 lb_f-s/lb_m [8–15 km/s] (Ref. 61) and engine thrust-to-weight ratios of 3×10^{-4} are typical]. Because any material can be launched in the payload buckets or projectiles of the catapults, they can be essentially omnivorous, using materials that might otherwise be waste (e.g., rock or soil, etc.). These devices also have the potential of being very efficient electrically (60–95%).

Tethers

Space tethers are long cables in space that are used to couple spacecraft to each other or to other masses and that allow the transfer of energy and momentum from one object to another. They can be used to perform a number of the functions of propulsion systems and thereby cheat the Rocket Equation. Tether concepts range from simple near-term concepts such as orbit raising or lowering to far-term space elevators reaching from the surface of the Earth into space.

Tether Applications

Some of the more near-term applications envisioned for tethers include "trolling" the upper atmosphere from the Space Shuttle, in-space orbit raising/lowering, surface-to-orbit launch, and electromagnetic propulsion or power production. When used for orbit raising or launch, tethers can be either stationary (i.e., hanging) or rotating (i.e., bolos). Descriptions of some of these near-term tether technologies are discussed next.

Stationary Tethers

As just mentioned, it is possible to use tethers to reel payloads in or out from an orbiting vehicle, such as the Space Shuttle Orbiter or the Space Station, to reduce orbit transfer vehicle (OTV) propulsion requirements. Note however that energy and momentum are still conserved; the Space Shuttle Orbiter or Space Station serve the purpose of a massive tether Station that minimizes altitude changes in the center of mass of the system (payload, tether, and tether station), as shown in Fig. 22. Thus, for payload orbit raising the tether station will decrease in altitude; too large a momentum transfer could even cause the system to deorbit. For a reusable system additional propulsion is required on the tether station. The advantage here is

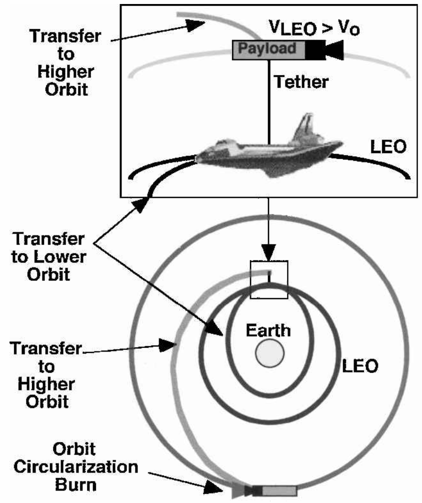


Fig. 22 Example of a tether used for orbit raising.

that the propulsion system on the tether station is already in place; only additional propellant needs be resupplied. This eliminates the cost and complexity of using a dedicated OTV to perform the orbit raising. Finally, small moons (such as Deimos or Phobos) or asteroids could be used as the anchor point for the tethers; the large mass of the moon would eliminate the need for a reboost propulsion system.

A number of tether space experiments have flown; the earliest was during the Gemini 11/12 missions in 1967 using a 30-m tether. More recently, the small expendable deployer system (SEDS) missions,[62] deployed from an expendable launch vehicle, have demonstrated the longest tethers to date (20 km). They also have the unique distinction of being the first manmade objects in space to be visible from the ground as a line (rather than point) source of reflected sunlight. Finally, designs for multistrand tethers have been developed to mitigate the problem of space debris impacts cutting the tether.[63]

Rotating Tethers (Bolo and Rotavator)

Another version of the tether concept is that of the rotating tether or bolo. This has an advantage for orbit raising in that the angular velocity of the tether can be used to match the orbital velocity of the pickup or drop-off points. Also, rotating tethers can be used in artificial gravity applications; they would be lighter than a rigid truss frame connecting the two halves of a rotating habitat. In this case the two halves would be reeled in or out to vary gravity during a mission. However, the dynamics and control of the tether during spin up or spin down and the perturbation caused by crew movement, etc., need to be addressed.

One potentially near-term application of rotating tether systems is their use to augment Earth-to-LEO launch, as well as LEO-to-lunar orbit transportation systems.[64] For example, a bolo system can be used to minimize the ΔV that a launch vehicle must provide to place a payload in LEO, GEO, or lunar orbit. In this system there is a careful coordination between the orbital speed of the center of mass of the tether and the tip speed of the rotating tether so as to produce a properly matched set of pickup and drop-off velocities. Thus, if the LEO tether has an orbital velocity of 7.7 km/s and a tip speed of 2.4 km/s it is possible for the launch vehicle to supply only 5.3 km/s in ΔV to achieve a rendezvous with the lower end of the tether. For LEO deliveries the payload would be reeled-in to LEO altitude and released. For a GEO delivery the payload would be jettisoned from the top of the LEO tether's swing on a LEO-to-GEO transfer ellipse. (The tether's tip speed of 2.4 km/s corresponds to the perigee ΔV of a LEO-to-GEO transfer ellipse.) A GEO tether, with an orbital speed of 3.1 km/s and a tip speed of 1.3 km/s (corresponding to the apogee ΔV of a LEO-to-GEO transfer ellipse without plane change), would rendezvous with the payload at the apogee of the transfer ellipse and either reel in the payload to GEO or release it at the top of the tether's swing to send the payload on a lunar or Earth-escape trajectory. Finally, a rotating tether in lunar orbit would capture the payload and reel it into lunar orbit.

In principal, if the amount of mass moving up through the system equaled that moving down, there would be an overall conservation of momentum and thus no need for propulsion. In practice, the center-of-mass tether station would require some propulsion capability to return the system to its nominal altitude after an operational cycle. This could be done with either chemical or electric propulsion, or, as discussed next for the Earth-orbit electrodynamic tether systems, by use of a separate electromagnetic tether (with power supplied by solar arrays) that pushes or pulls against the Earth's magnetic fields to provide propulsive force.

A more far-term example of a rotating tether system is the rotavator concept.[64] A rotavator is a long bolo in low orbit around a planet (or moon) in which the tether length, center-of-mass orbital altitude, and tip speed are selected so as to produce an essentially zero horizontal velocity at contact with the ground. This system could directly enable an Earth-to-orbit transfer to high Earth orbits, translunar trajectories, or Earth escape by reaching down from space to lift payloads from the Earth or to deposit payloads onto the Earth.

To reach the surface of the planet, the orbital altitude should be equal to half the length of the rotating cable. By proper adjustment of the cable rotation period to the orbital period of the center of mass of the cable (plus or minus the planetary rotation period), the relative velocity of the planetary surface and the tip of the cable can be made zero at the time of touchdown, allowing for easy payload transfer. A half-rotation later, the payload is at the top of the trajectory with a cable tip velocity that is twice the orbital velocity.

Although present-day materials (e.g., Kevlar®, etc.) do not allow the construction of rotavators around Earth or Venus, they can be built for Mars, Mercury, and most moons, especially including Earth's Moon. For Earth-orbit applications the rotavator's extreme length (8500 km total) and orbital dynamics stresses require the use of advanced materials and construction. For example, a tapered, rather than constant-diameter, cable is used to minimize cable mass. Also, although a lunar rotavator could be constructed using Kevlar, an Earth-orbit rotavator would require a material comparable to that of carbon nanotubes or crystalline diamond filaments.

Electrodynamic Tethers for Power Generation and Propulsion

A final near-term application of tethers involves tether interactions with planetary electro-magnetic fields. For example, an electrodynamic (ED) tether, which has a current running through it (with the current loop completed from the tip of the tether back to the spacecraft by electron conduction through the space plasma), can interact with the Earth's magnetic field to produce power like a generator; however, as power is extracted the orbit will decay unless propulsion is used. Conversely, if electric power is available (e.g., from solar cells) the current interacting with the Earth's magnetic field can produce force on the tether to act as a propulsion system. Electric power generation was demonstrated on the Shuttle TSS-1 (tethered satellite system) flights in 1992 and 1996. Interestingly, the 1996 flight also inadvertently demonstrated the orbit raising capability of tethers when the cable was severed as a result of current heating of a weak spot in the tether's insulation.

NASA MSFC is currently preparing a Pro-SEDS (propulsive small expendable deployer system) flight demonstration of a propulsive ED tether.[65] This mission will use a power-generating ED tether to deorbit a chemical upper stage (after it is used to inject a satellite towards GEO). This will have the effect of removing the upper stage as a source of space debris, without the need for any onboard chemical propellant. Once demonstrated, when ProSEDS is used on future flights, the chemical propellant that would ordinarily have to be kept in reserve to deorbit the spent stage could be used to inject a larger payload on a GEO transfer orbit.

For this mission the ED tether will use a small amount of the electric power generated by the ED tether for operation of the ProSEDS system. The bulk of the electric power will be dissipated by a simple resistive load. As just discussed, because energy is conserved, extraction of electrical energy causes a decrease in the orbital energy of the stage, ultimately causing the stage to spiral in until air drag causes it to reenter and burn up in the atmosphere.

Earth-to-GEO Space Elevator

The most extreme example of a tether system is the Earth-to-GEO space elevator[66] or SkyHook or Beanstalk). In this system[64] the space elevator center-of-mass station is in GEO; the tether hangs down 35,785 km to the Earth with no relative horizontal velocity. A second tether section, 110,000 km long, extends up to provide an orbital dynamics and mass balance to the Earth-to-GEO section. Payloads would travel up or down the tether; if they were released in LEO, they would need a propulsion system (or a launch-assist catapult on the space elevator at the LEO altitude) to increase their orbital velocity from that of GEO (3.1 km/s) to that of LEO (7.7 km/s). Payloads released above GEO would be released into a transfer ellipse to higher altitude, for example, release along the upper section of the tether at an altitude of 78,000 km would provide for Earth escape. Like the rotavator already discussed, the GEO space elevator requires advanced materials (carbon nanotube or diamond-filament) tapered cables, but a lunar or Martian GEO space elevator could be constructed with existing materials like Kevlar. Thus, the significant investments being made in carbon nanotube and diamond-film technology for commercial applications can have a major reverse spin-off impact on Earth-to-orbit transportation by enabling the rotavator and GEO space elevator concepts.

Finally, space elevators, at least on Earth, should be considered as a far-term concept (although they appear technically feasible for use on the smaller moons in the solar system) because of the need for advanced materials. However, beyond the technological demand is the issue of the sheer size of these systems. Nevertheless, even though this concept has infrastructure requirements rivaling those of major historical construction projects (e.g., the interstate highway system), it also holds the promise of reducing per-launch costs down to those associated with the intrinsic electric energy cost of raising an object in the Earth's gravity field and accelerating it to orbital velocity (e.g., 1–2 \$/kg from Earth to LEO).

Extraterrestrial Resource Utilization

One method of significantly extending our reach into space is to make use of materials (e.g., propellants, structural materials, shielding) derived from extraterrestrial sources. For example, in a sample return mission propellant required for the return trip could be made from indigenous materials at the landing site. This eliminates the need to carry propellant for the return trip all the way out from Earth, resulting in considerable savings in weight. This saving in weight, however, is reduced somewhat by the weight of machinery required to make the propellant at the landing site.

As shown in Fig. 23, a number of extraterrestrial resource utilization (ETRU) concepts have been demonstrated for producing propellants for chemical rockets. For example, a water electrolysis cell can be used to convert water (H_2O) into chemical propellant

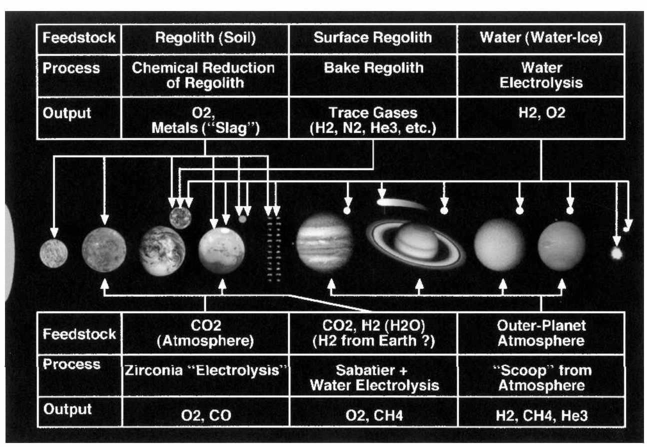


Fig. 23 Extraterrestrial resource utilization.

fuel and oxidizer, H_2 and O_2. This could also be used to produce hydrogen for a nuclear thermal rocket with the oxygen simply dumped overboard (or burned in a LANTR system). The ready availability of water-ice on the Earth's moon, Mars and its moons, the outer-planet moons, and comets or asteroids makes this an attractive approach for sample return or multiplanet, multimoon missions.

Water is not the only potential propellant feedstock. Several schemes have been devised to produce propellants from carbon dioxide (CO_2) in the Martian atmosphere[67] because of the major savings in Earth launch mass that can be realized for piloted missions. For example, O_2 could be produced that could be burned with fuel carried from Earth (e.g., methane, CH_4). Note that in many of these concepts, oxidizer (O_2) production is emphasized because the oxidizer weight is typically 5 to 10 times the fuel weight in a chemical propulsion system. Alternatively, the carbon monoxide (CO) produced in the CO_2 decomposition reaction could be used as fuel; although an O_2/CO propulsion system would have low performance ($I_{sp} \sim 260$ lb$_f$-s/lb$_m$ [2.5 km/s]), the ready availability of free propellant can compensate for the low performance. Finally, if water is available on Mars for ETRU it can be combined with CO_2 to produce methane and oxygen propellants by the Sabatier reaction. Alternatively, CO_2 from the Martian atmosphere can be burned directly with a reactive fuel such as magnesium (brought from Earth).[68]

It is even possible to process soil to produce oxygen, so that, quite literally, any rock in the solar system can be used as propellant feedstock. One approach under consideration is the use of lunar soil (regolith)[69] to provide oxygen for cis-lunar chemical propulsion.[70] It might even be possible to derive fairly pure metals like a aluminum from a lunar regolith processing system; the metals could be then burned with excess O_2 to give a high T/W propulsion system with an I_{sp} around 200–300 lb$_f$-s/lb$_m$ (2–3 km/s) (Ref. 71). Again, the low I_{sp} is countered by the ability to use totally nonterrestrial materials for propellant.

On a larger scale involving the future industrialization of space, ETRU methods will be particularly important because they provide a virtually unlimited supply of propellant and other raw materials. A lunar oxygen production system has already been mentioned that could supply extensive commercial cis-lunar space transportation operations or space industrialization. Hydrogen is also very valuable for both propulsion and industrial uses; unfortunately, the Moon is lacking in known large sources of hydrogen other than as water-ice. Fortunately, there appears to be significant deposits of water-ice in permanently shadowed craters or permafrost layers at the lunar poles. Actually, any source of extraterrestrial hydrogen (water, methane, ammonia, etc.) could be used so that bodies containing these chemicals, like the asteroids, Mars, or comets and their nuclei, could become important sources of hydrogen. Failing the discovery of a readily available source of hydrogen, nuclear thermal or electric propulsion systems could be developed that used lunar-produced oxygen as propellant mass, such as the LANTR concept discussed in the Nuclear Propulsion section.

Also, there are a number of trace gases that, although they do not represent large masses, can be important for life support (e.g., nitrogen [N_2]) or other applications. For example, the isotope He^3 is important as a nuclear fusion fuel for aneutronic (neutron-free) fusion propulsion and power concepts; it is present in small quantities in lunar regolith and in the atmospheres of the outer planets.

A final category of ETRU concepts are those that make use of an indigenous planetary atmosphere as the propellant working fluid mass. For example, H_2 from Jupiter's atmosphere could be used. These concepts would include ramjets, detonation propulsion schemes, and a "burn-anything" nuclear thermal rocket. In all of these "scooper" schemes, a mass of free propellant working fluid is collected on site and therefore does not need to be carried along from Earth. An energy source (nuclear reactor, etc.) is used to heat this mass of propellant. For those systems using an atmosphere like that of Earth or Venus, the I_{sp} is fairly low because the average molecular weight of the atmosphere is so large compared to that of H_2. However, this low I_{sp} is again countered by the ready availability of propellant mass and, for a planet like Venus, by the ability to operate in a high-pressure atmosphere.[72]

Finally, the ultimate ETRU concept is the Bussard Interstellar Ramjet,[73] in which interstellar hydrogen is scooped to provide propellant mass for a fusion propulsion system. Interstellar hydrogen would be ionized and then collected by an electromagnetic field. Onset of ramjet operation is at a velocity of about 4% the speed of light (c). Although the Bussard Interstellar Ramjet is very attractive for interstellar missions because of its unlimited range and potential for ultrarelativistic speeds ($\gg 0.5c$), there are several very major feasibility issues associated with its operation, such as fusion of hydrogen (e.g., it might be necessary to collect interstellar deuterium and discard the hydrogen), design of the electromagnetic scoop, and momentum drag from the collected hydrogen vs thrust from the fusion engine (with an exhaust velocity of only $3\%c$).[74]

Breakthrough Physics Propulsion

The term breakthrough physics propulsion (BPP)[75] covers a range of topics that represent cutting-edge theory and experiment that have the potential not only to revolutionize transportation and communications but also to produce as fundamental a paradigm shift in humanity's view of the nature of physical reality in the 21st century as did relativity and quantum mechanics at the beginning of the 20th century. For example, there were a number of serious problems in physics at the end of the 19th century (e.g., the sun's energy output over time, radioactivity, Mercury's orbit, the photoelectric effect, blackbody UV emission, and atomic line spectra) that could not be understood based on the reigning theoretical models of the day (e.g., Newton and Maxwell). The problems in 19th century physics were addressed by totally new theoretical and experimental paradigms (e.g., relativity and quantum mechanics). Today, there are equally vexing problems that are not understood by our current theories (e.g., missing mass of the universe, the new cosmological constant, naked singularities, time machines not forbidden, missing solar neutrinos, imaginary mass neutrinos, and instantaneous quantum state communication). It is the nature of breakthrough physics that at this very moment a new Albert Einstein or Max Planck might be creating the new models of the universe that will revolutionize our understanding of nature in the 21st century and beyond.

The main objective of the NASA BPP program is to advance science so as to provide for new foundations for breakthrough propulsion technology. Specifically, the goal is to produce *incremental*, *credible*, and *measurable* progress toward conquering the ultimate breakthroughs needed to revolutionize space travel and enable interstellar voyages. The technical aspects being pursued can be divided into three categories:

1) *Mass*: Discover new propulsion methods that eliminate or dramatically reduce the need for propellant. This includes such propellantless concepts as inertialess space drives, gravity shielding/antigravity, and thrusting against the zero-point vacuum field.

2) *Speed*: Discover how to circumvent existing limits to dramatically reduce transit times. This includes such faster-than-light transportation concepts as wormholes and warp drives.

3) *Energy*: Discover new energy methods to power these propulsion devices. This includes approaches such as extracting zero-point energy (Casimir Effect) from the vacuum of space itself.

Programmatic progress has included identification of issues and the potential for a research program. This was followed with a solicitation for proposals emphasizing experimental testing of theoretical predictions of anomalous (i.e., nonclassical/relativistic) behavior. An advisory counsel was convened to review the proposals, and selected tasks were funded. Results have been published in peer-reviewed journals. Unfortunately, funding for all revolutionary propulsion (including BPP) was cut in 2003.

Summary

As can be seen, there are an extraordinary number of advanced propulsion concepts. Virtually any one of these could revolutionize space exploration. However, the historical reality is such that it typically takes decades to go from concept to flight. As specific examples, Tsiolkovsky identified O_2/H_2 as the ultimate propellant combination for chemical propulsion, yet this technology did not enter routine service until the 1950s. Similarly, ion thruster development was begun in earnest in the late 1950s, yet the first deep-space solar electric propulsion (SEP) system did not fly until 1998.

There are several factors that inhibit the rapid development of advanced propulsion technology. For example, basic research is often tied to a graduate student's life cycle (e.g., 4 +years). One very serious issue is the dramatic cost increase over the developmental life of a research program as one goes from basic research or paper studies (typically a few \$100K) to several \$100M for space flight missions.

Nevertheless, although costly, flight demonstration missions (e.g., New Millennium DS-1 SEP) are critical for acceptance by project managers who historically are very risk adverse. And although it is a cliché, it is nevertheless true that "Nothing succeeds like success"—since the success of DS-1, many proposals have been submitted for SEP missions and are now being approved for study. Even the Project Prometheus JIMO NEP system, which would require advanced, high-power, high-I_{sp} ion engines, might have been perceived as far riskier without the success of the NSTAR ion engine on DS-1.

Based on these observations, we can make some predictions for the future use of advanced propulsion technologies. (Of course, any of these predictions could be altered by changes in national policy, e.g., an Apollo-scale commitment in space, or, alternatively, a major breakthrough in our understanding of physics.) In the near-term (5–15 years) we would expect to see a continued robotic exploration of the solar system. In this time frame we are basically limited to what is already in development (as opposed to basic research). Thus we can predict the use of SEP and NEP (e.g., Project Prometheus JIMO) employing advanced ion and/or Hall thrusters. Also, we can anticipate the use of aeroassist (with medium-high lift-to-drag ratio [L/D] aero-brake/capture as opposed to very low L/D aerobraking used today) at the target planet, with chemical or SEP used for injection. Other possibilities include solar sails, solar thermal propulsion, and momentum exchange tethers.

In the midterm (15–30 years) we can expect a return to human missions beyond low Earth orbit, including exploration of the Moon and Mars. For these missions with their large payloads and a premium placed on trip time for the piloted portion of the mission, nuclear thermal propulsion (NERVA/LANTR, bimodal) is a likely candidate. High-power (MW_e-class) SEP and NEP might be attractive for cargo missions that are less time sensitive than the piloted portion of the mission. Extraterrestrial resource utilization can be used to produce propellants on the Moon, Mars, or the moons of Mars. Also, towards the end of this time frame, we might see ultra-high-power 100-MW_e class (multimegawatt) NEP for piloted Mars missions. These systems could compete with nuclear thermal propulsion because the economy-of-scale in the NEP systems results in a dramatic decrease in specific mass (kg/kW_e),[30] and thus increase in T/W (acceleration), as compared to more modest-power NEP vehicles. Finally, we will continue aggressive robotic exploration of the solar system and beyond; in this era we should begin to see an increase in the use of microtechnologies for spacecraft systems including propulsion, as well as our first tentative steps into interstellar space with precursor missions beyond the heliosphere using advanced solar sails or NEP.

In the far term (30+ years) we should see the realization of routine, low-cost, fast access to anywhere in the solar system. However, in order to do this we will need to operate space systems at an unprecedented scale of performance and size. These very demanding technologies, which are in the basic research phase today, might include options like gas-core fission, fusion, or antimatter-catalyzed fission/fusion propulsion. There is also a category of systems that use a large, preexisting infrastructure as a means of reducing the operating costs of space missions, such as launch-assist catapults, laser propulsion ETO launch vehicles, or the space elevator. However, for these systems there remains the issue of capital investment and amortization of the initial infrastructure; in effect, we have to ask "Who builds the interstate highway system?" before the first dollar of revenue is collected.

Finally, in the very far term (22nd century ?) we can at least dream of interstellar missions using advanced fission, fusion, antimatter, or laser sails, for, as Tsiolkovsky said a century ago, "Earth is the cradle of humanity, but one cannot live in a cradle forever."[2]

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