Generalized Theory of Rocket Propulsion for Future Space Travel

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This paper introduces a generalized theory of classical rocket propulsion for reusable space vehicles wherein the propellant is atmospheric gas and the propulsive decelerating or accelerating thrust is generated by, respectively, ingesting or expelling atmospheric gas at high velocities. The working fluid expelled during accelerating propulsive maneuvers is automatically replenished during decelerating propulsive maneuvers. The refueling retrothrust is generated by a large-diameter hypervelocity gas inlet diffuser mounted at the front of the vehicle, which ingests gas while traversing through the tenuous upper regions of a planet's atmosphere. The vehicle thereby transfers a portion of its momentum to the collected gas, which was initially at rest in the atmosphere. This transfer of momentum generates a decelerating retrothrust by a reverse application of the theory of classical rocket propulsion. The ingested gas is liquefied, stored onboard the vehicle, and utilized as a propellant for generating the accelerating thrust by electrically accelerating and expelling the gas at high velocities. By expelling the gas with a velocity exceeding the inlet velocity, the vehicle becomes self-refueling. The vehicle could operate either as a self-refueling interorbital transfer vehicle or as an interplanetary transfer vehicle moving from planet to planet utilizing each planetary atmosphere as propulsive working fluid.

Introduction

THE fundamental physical principle that all space propulsion systems utilize for generating propulsive thrust is based on Newton's third law of motion: "For every action there is an equal and opposite reaction." According to the presently conceived theory of rocket propulsion (which will be referred to as the "classical theory of rocket propulsion"), Newton's principle is always applied in one direction—expelling mass (i.e., propellant) stored onboard the vehicle, out of the vehicle at high velocity. The magnitude T of the resulting thrust is

$$T = \dot{m}u \tag{1}$$

where $\dot{m}$ and u denote the mass flow rate and exhaust velocity, respectively.

Reusable space-based vehicles require essentially the same amount of decelerating propulsive thrust as accelerating propulsive thrust. However, in the classical theory of rocket propulsion, there is no functional or operational distinction between generating thrust for either acceleration or deceleration. The generating principle always remains the same—expelling mass stored onboard the vehicle. Decelerating or accelerating thrust is generated by expelling the propellant toward or away from the direction of motion, respectively. Thus, in the theory of classical rocket propulsion, only the magnitude ΔV of the velocity change is important—not its direction. (Decelerating thrust generated from the drag induced by passing a suitably protected space vehicle through the upper atmosphere such as in aerobraking is not considered as originating from a true rocket propulsion system or "rocket engine" within the context of this paper.)

The propulsive power P of the exhaust stream is given by

$$P = \tfrac{1}{2}\dot{m}u^2 \tag{2}$$

and the "mass ratio" is given by the well-known "rocket equation,"

$$M_1/m_2 = \exp(\Delta V/u) \tag{3}$$

where M_1 and M_2 denote the vehicle's mass before and after expelling an amount of propellant $m_p = M_1 - M_2$ to achieve a velocity change ΔV.

It follows directly from Eq. (3) that

$$m_p = M_1[1 - \exp(-\Delta V/u)] \tag{4}$$

and

$$\Delta V = u \ \log(M_1/M_2) \tag{5}$$

Equations (1–5) are among the most important in the field of astronautics because they essentially determine the capabilities and limitations of all presently conceived space vehicles.

The equations clearly show that the efficiency of a vehicle's rocket propulsion system increases exponentially with increasing exhaust velocity. Unfortunately, it is impossible to construct a chemical rocket engine that can generate an exhaust velocity greater than approximately 4.70 km/s (which corresponds to a specific impulse of 480 s). This limit is imposed by basic laws of thermodynamics that cannot be violated. Thus, chemically propelled space vehicles have inherently high mass ratios and therefore consume great quantities of propellant. For example, in order to deliver payloads to geosynchronous Earth orbit (GEO) from low Earth orbit (LEO) by chemically propelled reusable orbital transfer vehicles (OTV's), almost all of the total mass transported to LEO from the Earth's surface must be propellant for the OTV's.[2] Thus, for missions to GEO by chemical OTV's, about 90% of the total weightlifting capability of the Earth-to-orbit launch vehicle will be wasted by having to carry up propellant for the OTV. Although electric propulsion systems do offer a substantial reduction in propellant requirements, they suffer from having inherently long flight times.[3,4] Consequently, for short flight time missions to GEO (such as manned flights), chemical propulsion appears to be unavoidable. This will inevitably result in burdening the Earth-to-orbit launch vehicle with enormous quantities of liquid rocket propellants because the ratio of propellant-to-hardware will be on the order of 10:1. Furthermore, this ratio is even greater for interplanetary missions.

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Since the present theory of rocket propulsion based on expelling mass is assumed—axiomatically—to be the only method for generating engine thrust, any significant amount of manned space travel beyond LEO will be a very difficult and costly proposition that is not likely to improve even in the distant future. Journeys to the moon's surface by ordinary private citizens will be rare and high-speed voyages to Mars (or other celestial bodies) by ordinary citizens will be essentially impossible. The initial mass problem is so severe that even the development of high-thrust gas-core nuclear rocket engines with a specific impulse of 5000 s will require several million kilograms of propellant just to carry out one fast round-trip mission to Mars with a small number of passengers.[5] Although the cost of manned space travel can be reduced somewhat by developing more economical methods for delivering the required propellant, the amount of propellant is simply too great to allow any significant commercial space travel.[6] The root of the problem lies in the theory of rocket propulsion itself. The purpose of the present paper is to introduce a radically new theory of rocket propulsion that completely eliminates the initial mass problem.[1]

Generalized Theory of Classical Rocket Propulsion

The solution to the initial mass problem of classical rocket propulsion proposed herein is substantially different from all others in that it sets forth a fundamentally new concept for propelling a reusable space-based vehicle. It is based upon utilizing the atmosphere of a planet as propellant and designing the propulsion system to be self-refueling. This is accomplished by first recognizing three simple facts: 1) a reusable space-based vehicle requires approximately as many decelerating retropropulsive maneuvers as accelerating propulsive maneuvers; 2) any propulsive maneuver will be more efficient if carried out when the vehicle is moving closest to the central body when its velocity is maximum; and 3) most of the required decelerating retropropulsive maneuvers can be generated by a reverse application of the classical theory of rocket propulsion by allowing a vehicle to dip into and ingest atmospheric gas initially at rest outside the vehicle using an inlet diffuser, instead of expelling gas initially at rest inside the vehicle using an exhaust nozzle. The third fact represents the most important observation because it provides a thrust-generating principle that allows decelerating retropropulsion to be performed by a system separate from the vehicle's accelerating propulsion system; and this new propulsion system can, by its very nature, provide all of the propellant needed to operate the accelerating propulsion system.

As shown in Fig. 1, the retropropulsion system consists of a hypervelocity gas intake diffuser mounted on the front of the vehicle, and the accelerating propulsion system comprises a system mounted on the end of the vehicle for expelling the ingested gas at high velocity. The complete propulsion system consists of both of these separate systems. By designing the diffuser with a sufficiently large inlet diameter exceeding the maximum diameter of the vehicle's main body, all external aerodynamic drag forces are essentially eliminated. Thus, when the vehicle traverses the upper atmosphere ingesting gas during a "refueling retromaneuver," the retropropulsive thrust is generated by transferring a portion of its momentum to the intgested gas that was initially at rest in the upper atmosphere. This transfer of momentum provides the vehicle with the required decelerating retrothrust by a reverse application of the classical theory of rocket propulsion, while the vehicle's supply of onboard propellant is simultaneously replenished for the next accelerating propulsive maneuver.

The magnitude of the retrothrust can be calculated by a reverse application of Eq. (1). In this case, the propellant mass flow rate $\dot{m}$ refers to the intake mass flow rate $\dot{m}$(in) of atmospheric gas passing into and ingested by the vehicle, and the flow velocity u refers to the gas inlet velocity u(in). The gas inlet velocity u(in) is equal to the vehicle's instantaneous orbital velocity relative to the planet. It can be approximated by the vehicle's perigee velocity (when its altitude is minimum).

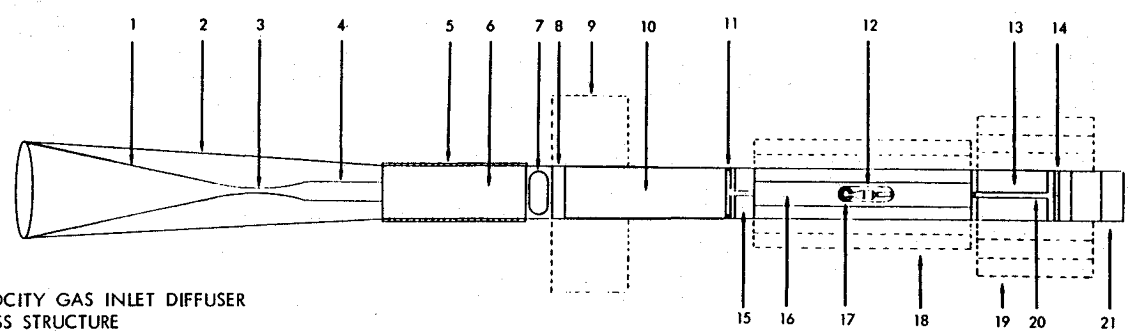
The retropropulsive thrust generated by this system can be extremely high (much higher than the accelerating propulsion system). It is controlled by controlling the perigee altitude of the atmospheric entry trajectory—the lower the altitude, the greater the retrothrust.

The vehicle's velocity change $\Delta V = V_1 - V_2$, before and after the refueling retromaneuver, can be calculated by applying the principle of the conservation of momentum. Thus, if M_1, V_1 and M_2, V_2 denote the vehicle's total mass and velocity before and after the refueling retromaneuver at the entry and exit points of the atmosphere, respectively, it follows that

$$M_1/V_1 = M_2/V_2$$

Consequently, the mass ratio corresponding to a refueling retromaneuver is given by

$$M_1/M_2 = V_2/V_1 \tag{6}$$



1 HYPERVELOCITY GAS INLET DIFFUSER
2 OPEN TRUSS STRUCTURE
3 TRANSPARENT THROAT
4 TRANSPARENT HOT GAS RADIATION CHAMBER
5 HIGH TEMPERATURE COOLING COIL
6 HIGH PRESSURE GAS STORAGE TANK
7 GAS LIQUEFACTION SYSTEM
8 CREW QUARTERS AND CONTROL CENTER
9 RETRACTABLE LOW TEMPERATURE RADIATOR
10 PAYLOAD BAY
11 AUXILARY CHEMICAL ROCKET ENGINES
12 NUCLEAR-ELECTRIC GENERATING SYSTEM
13 MAIN CRYOGENIC STORAGE TANK
14 MICROWAVE PHASED ARRAY TRANSMITTER
15 O_2/H_2 CRYOGENIC STORAGE TANKS
16 NUCLEAR POWER PLANT BAY
17 NUCLEAR RADIATION SHIELD
18 RETRACTABLE NUCLEAR-ELECTRIC RADIATOR
19 RETRACTABLE RADIATOR FOR MICROWAVE GENERATORS
20 ACCESS TUNNEL
21 SUPERCONDUCTING ECR PLASMA ACCELERATOR

Fig. 1 Schematic longitudinal cross section of a self-refueling space vehicle.

and the amount of atmospheric gas ingested by the vehicle is given by

$$m_p = M_1\left(\frac{V_1}{V_2} - 1\right) \tag{7}$$

The retrovelocity change $\Delta V = V_1 - V_2$ is given by

$$\Delta V = V_1\left(1 - \frac{M_1}{M_2}\right) \tag{8}$$

Unlike the classical theory of rocket propulsion, the magnitude of the retrothrust T is a function of time t and depends on the vehicle's instantaneous velocity $V(t)$ and the density $\rho(h)$ of the atmosphere at the vehicle's altitude h. It can be expressed as

$$T(t) = \dot{m}(t)V(t) \tag{9}$$

where the incoming propellant mass flow rate $\dot{m}(t)$ is given by

$$\dot{m}(t) = \pi r^2 V(t)\rho(h) \tag{10}$$

where r is equal to the diffuser's inlet radius. Thus, a detailed analytical analysis of a refueling retromaneuver requires a numerical integration of the vehicle's equation of motion while traversing through the atmosphere. This equation can be expressed as

$$\frac{\mathrm{d}\boldsymbol{V}}{\mathrm{d}t} = -\frac{\mu\boldsymbol{R}}{R^3} - \frac{\rho\pi r^2 V^2 \hat{\boldsymbol{V}}}{M(t)} \tag{11}$$

where $\boldsymbol{R}$ and $\boldsymbol{V}$ denote the vehicle's instantaneous position and velocity vectors, respectively. The gravitational constant is denoted by μ. The unit vector $\hat{\boldsymbol{V}} = \boldsymbol{V}/V$ and the mass function $M(t)$ represents the vehicle's instantaneous total mass given by

$$M(t) = M_0 + \int_0^t \dot{m}(t)\mathrm{d}t \tag{12}$$

where M_0 denotes the vehicle's total mass before the retromaneuver and t denotes the instantaneous time variable during the retromaneuver. The vehicle enters the atmosphere at time $t = 0$.

The numerical integration of the equation of motion (11) has been carried out to simulate detailed refueling retromaneuvers at many different planets. It was observed that relatively minor variations in the density function $\rho(h)$ (as a function of the altitude h) has a substantial effect on the retrothrust and the postretrotrajectory. Thus, accurate trajectory calculations require the most accurate tabulated values of density vs altitude that are available.

In the classical theory of rocket propulsion, the mass ratio Eq. (3) is always greater than 1. However, in the "refueling retro theory," the mass ratio Eq. (6) is always less than 1. By combining both theories into a single "generalized theory of rocket propulsion," it is possible to achieve space travel where the mass ratio is unity—or better yet, where the mass ratio is less than unity. According to this theory, gas entering the diffuser during a retropropulsive maneuver is cooled and accumulated at high pressure. It is subsequently liquefied and stored inside thermally insulated storage tanks onboard the vehicle. The vehicle utilizes this gas later as propellant to generate accelerating thrust by expelling it at high velocity u(out) via the classical theory of rocket propulsion. Consequently, in this generalized theory of rocket propulsion, all of the atmosphere surrounding all of the planets in the entire solar system can be regarded as "stored propellant." The propellant stored inside the vehicle is used for accelerating the vehicle, while the propellant stored outside the vehicle is used for decelerating the vehicle. Thus, in this theory, the propellant becomes a reusable inertial medium that is simply transferred from one storage location to another storage location depending upon whether the vehicle is being accelerated or decelerated.

It follows directly from Eqs. (4) and (7) that if the exhaust velocity u(out) can be made to exceed the maximum inlet velocity u(in) and if the magnitudes of the accelerating and decelerating ΔV are equal, then the mass ratio will be equal to or less than unity and the vehicle will be self-refueling. A mass ratio less than unity means that the vehicle will ingest more atmospheric gas during decelerating maneuvers than it expels during accelerating maneuvers. This will be very desirable for Earth orbital operations because the oxygen can be separated from the ingested air (during the liquefaction process) and used for propelling small chemical rocket vehicles such as lunar or planetary landing vehicles. (Self-refueling vehicles could not be used for landing on celestial bodies.) Only the nitrogen component of the air would be used for the accelerating propulsion system.

The possibility of generating large amounts of oxygen as a free by-product from self-refueling vehicle operations is important because only the hydrogen component of O_2/H_2 chemical rocket vehicles would have to be transported from the Earth's surface. Since the corresponding oxygen-to-hydrogen mass ratio is 8:1, the cost of transporting propellant to these vehicles would be reduced by a factor of 8.

Since not all retropropulsive maneuvers can be performed by ingesting atmospheric gas during round-trip interorbital or interplanetary missions, the exhaust velocity u(out) will usually have to be significantly greater than the ingestion velocity u(in) if the vehicle is to be self-refueling. The determination of the minimum exhaust velocity u_m required to achieve self-refueling operation depends upon the particular mission. For round-trip missions between LEO and GEO, this minimum exhaust velocity is 22.40 km/s ($I_{sp} = 2286$ s) and is independent of vehicle mass. If the vehicle is designed to operate with a mass ratio less than unity, where all of the oxygen ingested is separated from the air and given to chemical rocket vehicles, the minimum required exhaust velocity is 27.65 km/s ($I_{sp} = 2820$ s). These minimum exhaust velocities are easily obtainable using electromagnetic plasma accelerators energized by an onboard nuclear-electric generating plant.

Unlike the classical theory of rocket propulsion, this generalized, self-refueling theory of rocket propulsion is independent of vehicle mass. Hence, it is possible to design the vehicle with a very high dry mass without having to supply propellant to operate it. This allows the vehicle to be very large in order to take advantage of the favorable scaling laws of nuclear-electric power plants. For example, a nuclear-electric power plant of the type proposed by Rosa[7,8] with a mass of 100,000 kg will be capable of generating several gigawatts of electric power. This allows the vehicle to have a fairly high thrust-to-weight ratio that is required for short-flight-time manned missions.

Refueling Retropropulsion System

As described previously, the retropropulsive maneuver is initiated by allowing the vehicle to dip down into the tenuous upper reaches of a planet's atmosphere. The vehicle traverses the atmosphere ingesting large quantities of the gas into the diffuser which generates decelerating retrothrust by a reverse application of classical rocket propulsion. Essentially all of the kinetic energy of the gas entering the diffuser is converted into thermal energy. For Earth orbital operations, the incoming gas stream will be moving at approximately 10 km/s. This corresponds to a specific energy of 5×10^7 J/kg. In an ordinary hypervelocity diffuser, all of this energy is converted into thermal energy and the resulting stagnation temperature would be on the order of 20,000 K. In order to propel space vehicles under the generalized theory of rocket propulsion (via self-refueling vehicles), all of this thermal energy must be expelled from the vehicle without melting any of its structure. This introduces a fundamental design problem that at first glance, may appear to be insoluble.

One solution to this problem was found by making large portions of the diffuser walls transparent using material such as fused silica-quartz glass so that most of the thermal energy can be radiated at optical wavelengths through the walls without heating them. The walls of the subsonic region of the diffuser directly behind the throat are also made transparent and are extended several meters downstream in order to expel additional thermal energy via radiation at optical wavelengths. Those portions of the diffuser walls that are not transparent are designed as "blackbody" radiators and fitted with high-temperature radiator fins to enhance the total radiative heat transfer. This diffuser design will make it possible to expel almost all of the incoming thermal energy by direct radiation into space without heating the vehicle's structure. The general design of the hypervelocity inlet diffuser is illustrated in Fig. 1.

All of the transparent walls of the radiating structure have a double-wall construction with a transparent pressurized coolant circulating in between to maintain the inner walls at a relatively low temperature. Since pure fused silica-quartz glass is almost 99% transparent at optical wavelengths, very little thermal energy will be absorbed by the transparent walls. The inside surface of the inner walls of the diffuser can also be cooled by making some portions porous and injecting cooling gas. This would provide a transparent insulating thermal blanket between the hot boundary layer and the inside walls of the diffuser. The basic design principles of this cooling system are similar to those used in the so-called nuclear "light bulb" rocket engine.[5] However, in the present case, the object is to cool the gas by radiative heat transfer rather than to heat it as in the nuclear light bulb engine—which is much easier.

The perigee altitudes during refueling retromaneuvers for Earth orbital operations will range between 65 and 70 km. At these altitudes, the air density is fairly high and the gas-dynamics will be in the continuum flow regime. Under equilibrium conditions, the total thermal power radiated by the diffuser will be equal to the propulsive power of the incoming gas. Hence, the actual temperature of the gas passing through the throat can be approximated by

$$T_0 = \left[\frac{\dot{m}V^2}{2\bar{\epsilon}_0\sigma A_0}\right]^{1/4} \tag{13}$$

where $\bar{\epsilon}_0$ is the average emissivity of the gas, $\sigma = 5.669 \times 10^{-8}$ W/m$^2\cdot$K^4, and A_0 the area of the transparent walls. The gas entering the hypervelocity diffuser is cooled via radiative heat transfer, compressed, and decelerated to sonic velocity as it passes through the throat at temperature T_0. The corresponding gas pressure p_0 and density ρ_0 at the throat are given by[9]

$$p_0 = \left(\frac{\dot{m}}{A_t}\right)\left(\frac{\mathcal{R}T_0}{\gamma\mathfrak{M}}\right)^{1/2} \tag{14}$$

and

$$\rho_0 = \left(\frac{\dot{m}}{A_t}\right)\left(\frac{\mathfrak{M}}{\gamma\mathcal{R}T_0}\right)^{1/2} \tag{15}$$

respectively, where A_t is the transverse cross-sectional area of the throat. The constant $\mathcal{R}$ is the universal gas constant (8.3145 J/mole$\cdot$K), γ the gas constant, and $\mathfrak{M}$ the equivalent molecular weight of the atmospheric gas entering the diffuser.

After passing through the throat, the gas flow is reduced to a relatively low subsonic speed by gradually increasing the inside wall diameter, which is the subsonic portion of the diffuser. Although the diameter of the gas stream is increased through the subsonic region, there is relatively little pressure loss due to back pressure that is maintained nearly equal to p_0. The subsonic portion of the diffuser is extended downstream several meters to form a long cylinder. Since the gas moving from the throat of the diffuser through the subsonic region and through this cylinder is still incandescent, the walls of this entire structure are also made transparent to expel additional radiation at optical wavelengths. Assuming isobaric flow and radiative heat transfer, the gas temperature T_1 at the end of the transparent cylinder is given approximately by

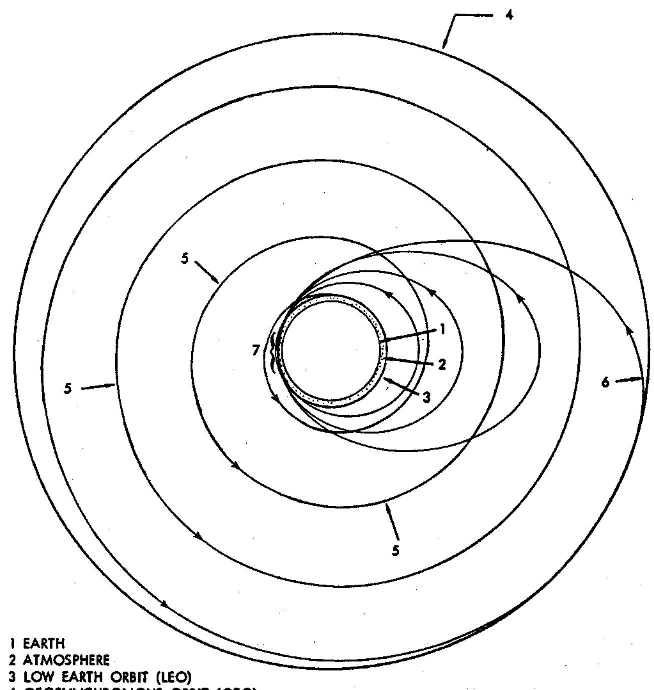


Fig. 2 Round-trip trajectory of a self-refueling transfer vehicle from LEO to GEO.

$$T_1 = \left[\frac{3\bar{\epsilon}_1\sigma A_1}{\dot{m}C_p} + \frac{1}{T_0{}^3}\right]^{-1/3} \tag{16}$$

where A_1 denotes the total radiating surface area of the structure (which comprises the subsonic portion of the diffuser and the connecting cylinder) and $\bar{\epsilon}_1$ the average emissivity of the gas moving through it. The specific heat capacity of the gas (at constant pressure) is denoted by C_p.

The gas leaving the radiating cooling cylinder at temperature T_1 is then fed into a secondary cooling system comprising a long spiraling coil that winds around a portion of the vehicle's outer circumferential periphery (see Fig. 1). This coil is fitted with small radiator fins that extend inside the coil to increase thermal conductivity. The fins (and surface of the coil) have an emissivity of about 0.95 for maximum radiant heat transfer. The coil is divided into many thermally insulated sections to maximize the radiative heat transfer to space. (The initial portion of this coil may have a transparent surface to maximize radiative heat transfer.)

The gas leaving the secondary cooling system at temperature T_2 is injected directly into a large thermally insulated high-pressure holding tank mounted inside the inner periphery of the cooling coil (see Fig. 1). This tank is reinforced both externally and internally with high-strength Kevlar. No attempt is made to liquefy the incoming gas while the vehicle is passing through the atmosphere. Rather, the gas is simply injected into and accumulated in the holding tank in a pressurized gaseous state and liquefied after the vehicle leaves the atmosphere. In order to make the liquefaction process as easy as possible, the pressure inside the holding tank is maintained as high as possible. This condition can be satisfied by designing the diffuser such that the throat pressure p_0 given by Eq. (14)

is very high. The holding tank is designed with a plurality of interconnecting compartments equipped with one-way, pressure-activated check valves to maintain maximum pressure.

Assuming that the diffuser has an inlet diameter of 24 m and a throat diameter of 0.20 m and that the holding tank is 50 m long with an outside diameter of 14 m, the maximum values of T_0, T_1, T_2, and p_0 corresponding to an entry velocity of 10 km/s into the Earth's atmosphere will be about 5200, 2400, 840 K, and 175 atm, respectively.

After the vehicle leaves the atmosphere, a large retractable low-temperature radiator system is extended and the high-pressure gas is circulated through it at a low continuous rate, thereby cooling it to about 270 K. Since the gas emerging from the low-temperature radiator has a pressure of about 80 atm, its entropy is very low and it is easily liquefied by conventional expansion methods,[10] which also result in the production of mechanical work that can be used to drive compressors or other machinery. The liquefied gas is separated into its component parts and stored in separate cryogenic storage tanks. For Earth orbital operations, only nitrogen is used for the accelerating propulsion system and the oxygen is used for propelling chemical rocket vehicles (such as small lunar or planetary landing modules). It should also be pointed out that in some missions, liquefaction may be unnecessary.

Although the huge transparent glass-walled hypervelocity diffuser described here (which represents the heart of the refueling retropropulsion system) may appear to be inherently fragile and incapable of withstanding the extremely hot thermal environment, it should be noted that pure fused silica-quartz glass is among the strongest of all high-temperature refractory materials. Furthermore, by carrying out a required retromaneuver by employing several successive passes into the atmosphere with relatively high perigee altitudes, the peak thermal loads and mechanical stresses can be significantly reduced. The total retromaneuver does not have to be executed during one pass through the atmosphere.

Accelerating Propulsion System

As previously pointed out, self-refueling space vehicles operating under the generalized theory of rocket propulsion are not mass sensitive. Consequently, they can be designed with very high dry mass to take advantage of the favorable scaling laws of Rosa-type, high-power nuclear-electric generating plants. With gigawatt power levels available onboard the vehicle, the gas can be converted into a "low-temperature" plasma using a small fraction of the power and then accelerating it to very high exhaust velocities with the remaining portion.

In the preferred embodiment, the accelerator is a giant multigigawatt superconducting electron cyclotron resonance (ECR) plasma accelerator.[1] Electric power obtained from the nuclear-electric power plant is converted into microwaves by high-efficiency microwave generators (operating at X-band) and fed into a beam-forming phased array transmitter several meters in diameter. The beam power is several gigawatts and is directed into a large-diameter superconducting solenoid that comprises the driving coil of the ECR accelerator. There is relatively little power loss in the conversion to microwaves since the efficiency of the generators will be about 95% and they will have very low specific mass on the order of 10 kg/MW.[1] (The favorable scaling laws also apply to high-power microwave generators.) The total thrust-generating capability of this giant superconducting ECR propulsion system is comparable to that generated by chemical rocket engines, thus enabling the vehicle to have the relatively high thrust-to-weight ratio required for manned space travel. This plasma accelerator is ideal for self-refueling vehicles because it can accelerate any arbitrary gas or combination of gases found in the atmosphere of any planet to extremely high velocities. Since there are no moving parts, no high-voltage systems, and no electrodes, there can be no mechanical failure or electrical failure due to short circuits or eroding electrodes. The exhaust velocities are well above the minimum required for self-refueling vehicle operation and can easily exceed 100 km/s.

As shown in Fig. 1, the vehicle is also equipped with a large plurality of auxiliary O_2/H_2 chemical rocket engines mounted in retractable pods around the circumferential periphery at the end of the payload bay. These engines provide emergency backup propulsion in case of a major failure in the nuclear-electric power plant or in the ECR engine.

Future Space Travel with Self-Refueling OTV's

Figure 2 describes a typical round-trip transfer trajectory of a self-refueling OTV transporting a high mass payload from LEO to GEO and returning to LEO. The vehicle is accelerated out of LEO using its high-thrust ECR engine while drawing maximum power from its nuclear-electric power plant. Since the payload mass is assumed to be very high, the propulsive maneuver from LEO to GEO is continuous and requires several days. It is shown in Fig. 2 by the expanding spiraling trajectory. The transfer is executed in an optimal fashion that simultaneously reduces the inclination to zero. Thus, as the vehicle approaches GEO altitude, its orbit becomes circular and its inclination becomes zero. (Large payloads that cannot fit inside the vehicle's payload bay are secured to the inlet diffuser by small removable clamps and boosted to GEO outside the vehicle.)

After rendezvousing with an orbiting space station and delivering the payload, the vehicle is returned to LEO by an initial retro and plane change maneuver. Since there is no atmosphere at GEO, this initial retromaneuver back to LEO must be performed by the vehicle's ECR engine in the usual way by expelling propellant in the direction of motion. As is shown in Fig. 2, this maneuver is designed to lower the perigee altitude of the vehicle's return trajectory so that it dips inside the upper reaches of the Earth's atmosphere. This allows all major subsequent retromaneuvers to be performed by passing

Table 1 Flight parameters corresponding to four refueling retropropulsive maneuvers from GEO to LEO

Parameter	1st retro	2nd retro	3rd retro	4th retro
a_1	24,307	14,337	10,256	7875
e_1	0.7349	0.5506	0.3718	0.1814
p_1	70.00	70.00	70.00	73.42
M_1	750,000	793,450	843,728	909,287
ϕ_1	−4.150	−3.795	−3.311	−2.375
θ_1	−9.800	−10.699	−12.249	−15.582
V_1	10.323	9.755	9.171	8.506
h_{min}	68.743	68.476	67.952	70.360
d_{max}	0.696	0.604	0.528	0.311
P_{max}	26,524	23,024	20,141	11,844
V_2	9.755	9.170	8.504	7.857
θ_2	11.056	12.656	16.435	36.115
ϕ_2	3.951	3.475	2.653	0.851
θ_{12}	20.856	23.355	28.684	51.697
T_{12}	234	278	366	718
m_p	43.457	50.292	65,587	74,316
ΔV	0.566	0.582	0.663	0.645
M_2	793,457	843,741	909,314	983,603
A_2	15,859	7696	2930	237
e_2	0.5508	0.3720	0.1820	0.0154
a_2	14,336	10,245	7870	6510
T_{23}	2.373	1.435	0.965	0.726

Summation of refueling retromaneuvers at Earth

Total retro ΔV	2.456 km/s
Total air ingested	233,652 kg
N_2	179,506 kg
O_2	54,146 kg

into and ingesting portions of the Earth's atmosphere, thereby automatically refueling the vehicle for the next mission. The approach and atmospheric entry trajectory is designed to give a sufficiently high perigee altitude to avoid generating excessive deceleration loads that may overstress the vehicle's structure. Thus, as illustrated in Fig. 2, the refueling retropropulsive maneuvers on the way back to LEO comprise a series of several atmospheric re-entry trajectories designed to lower the vehicle's apogee altitude until it reaches LEO. When the apogee reaches LEO altitude, a positive propulsive maneuver is executed when the vehicle passes through its apogee (with the ECR engine) which lifts the perigee altitude out of the atmosphere and circularizes the orbit in LEO. This final propulsive maneuver is very small and requires a ΔV of only about 60 m/s.

Detailed numerical trajectory calculations have been carried out to analyze quantitatively the preceding mission. In order to achieve a low specific mass for the nuclear-electric power plant, the vehicle's dry mass was assumed to be 750,000 kg. (50,000 kg of O_2/H_2 propellant for the auxiliary chemical rocket engines is considered to be part of the vehicle's basic dry mass since these engines are not normally used during any of the propulsive maneuvers.) The net output of the power plant was assumed to be 3500 MW. This electric power is converted into propulsive power by the ECR engine with an overall efficiency of 86%. Assuming that the ECR engine operates with a specific impulse of 20,000 s, and that the total amount of N_2 expelled from the vehicle while taking the payload from LEO to GEO is completely replenished during the refueling retromaneuvers back to LEO, the payload mass turns out to be 4,292,163 kg. The flight time from LEO to GEO is 11.1 days. Hence, the amount of fissionable material consumed by the nuclear-electric power plant will only be a few kilograms—which essentially represents the basic operating cost of the mission. (For small payloads on the order of 100,000 kg, the flight time to GEO would only be a few hours.)

Table 1 is a list of 22 flight parameters describing the four successive refueling retromaneuvers from GEO to LEO. They were obtained by integrating the vehicle's equation of motion while traversing the atmosphere. The diffuser's inlet diameter is assumed to be 24 m. The atmospheric entry and the exit altitudes are 110 km (360,892 ft). The parameters are defined in Table 2.

The inclination angle ϕ is measured relative to the local horizon and the position angle θ (which is the true anomaly) relative to the trajectory's periapsis vector. These angles have negative or positive values depending upon whether the vehicle is entering or leaving the atmosphere, respectively.

After the fourth refueling retromaneuver, the vehicle leaves the atmosphere with an apogee altitude of 237 km. It then executes a small accelerating ΔV maneuver of 60 m/s while passing through apogee, thereby circularizing its orbit in LEO.

It should be pointed out and emphasized that the refueling retromaneuvers not only replenish the vehicle's initial supply of N_2 that was used to transport the 4,292,163 kg payload from LEO to GEO, but also generate 54,146 kg of O_2—which it did not have before the mission. This oxygen production could more than pay for the total cost of transporting this enormous payload to GEO.

Although self-refueling space vehicles with much lower mass could be constructed, the overall performance would decrease because the vehicle's accelerating thrust-to-weight ratio would be lower. This is because decreasing the vehicle mass results in increasing the specific mass of the nuclear-electric power plant and because self-refueling vehicles require a high specific impulse for the accelerating system that is independent of vehicle mass. Low thrust-to-weight ratios will result in long flight times that are impractical for manned missions.

One of the most important design characteristics of space vehicles operating under the generalized theory of rocket propulsion is that their overall performance increases with inertial mass with relatively little increase in operating cost. This is because greater vehicle mass results in higher accelerating thrust-to-weight ratios, and —most important of all—because the vehicle's inertial mass itself is utilized to generate all of the propellant required to operate it via refueling retropropulsion. Under the theory of classical rocket propulsion, a vehicle's inertial mass is inherently the main culprit impeding performance that must be overcome by expelling huge amounts of propellant, which has to be replaced at very high cost.

Since the specific impulse of the ECR system is well above the minimum required for self-refueling operation with O_2 generation (which is 2820 s), there will always be much more

Table 2 Flight parameters

a_1	= semimajor axis of approach trajectory, km
e_1	= eccentricity of approach trajectory
p_1	= perigee altitude of approach trajectory, km
M_1	= vehicle mass before entering atmosphere, kg
ϕ_1	= trajectory inclination at entry point, deg
θ_1	= position angle of entry point, deg
V_1	= vehicle's entry velocity, km/s
h_{min}	= vehicle's minimum altitude, km
d_{max}	= vehicle's maximum deceleration, g
P_{max}	= vehicle's maximum retropropulsive power, MW
V_2	= vehicle's exit velocity, km/s
θ_2	= position angle of exit point, deg
ϕ_2	= trajectory inclination at exit point, deg
θ_{12}	= sweep angle of retromaneuver around planet
T_{12}	= total time of retromaneuver, s
m_p	= total mass of propellant ingested, kg
ΔV	= effective ΔV (impulsive at perigee), km/s
M_2	= vehicle mass after leaving atmosphere, kg
A_2	= apogee altitude after retro, km
e_2	= eccentricity of exit trajectory
a_2	= semimajor axis of exit trajectory, km
T_{23}	= flight time to apogee after retro, h

Table 3 Flight parameters corresponding to two refueling retromaneuvers at Mars ($V_\infty = 3$ km/s)

Parameter	1st retro	2nd retro
a_1	−4759	16,122
e_1	1.7178	0.7881
p_1	30.00	30.00
M_1	950,000	1,171,662
ϕ_1	−9.786	−8.166
θ_1	−15.465	−18.549
V_1	5.788	4.674
h_{min}	28.913	27,838
d_{max}	1.076	0.627
P_{max}	29,367	17,106
V_2	4.673	3.500
θ_2	18.549	46.770
ϕ_2	8.337	2.156
θ_{12}	34.014	65.319
T_{12}	396	1018
m_p	221,662	378,019
ΔV	1.115	1.174
M_2	1,171,622	1,549,682
A_2	25,442	242
e_2	0.7883	0.0376
a_2	16,121	3496.6
T_{23}	8.631	0.872

Summation of refueling retromaneuvers at Mars

Total retro ΔV	2.289 km/s
Total atmosphere ingested	599,681 kg

N_2 ingested during the refueling retromaneuvers returning from GEO than is expelled by the ECR engine going to GEO. This excess propellant production allows the vehicle to be used as an ultra-high-speed interplanetary transfer vehicle. The payload could be a reusable chemically propelled landing module using the oxygen accumulated during Earth orbital operations.

Table 3 describes two refueling retromaneuvers at Mars after an ultra-high-speed, 40-day interplanetary journey from Earth with a 200,000 kg landing module. In order to avoid entering the Martian atmosphere with excessive speed, the vehicle is decelerated (using its ECR engine) to an asymptotic approach velocity of 3 km/s.

After the second retromaneuver, the vehicle leaves the atmosphere with an apogee altitude of 242 km and circularizes its orbit at this altitude, which becomes the parking orbit. After the vehicle is inserted into the parking orbit, the landing module is withdrawn from the cargo bay and used for carrying out a manned landing on the Martian surface. The landing module is sufficiently large to carry a crew of ten or more and a great array of exploration equipment. After spending about twenty days exploring Mars, the crew returns to the mother ship inside the landing module. The self-refueling mother ship is then accelerated out of the parking orbit for an ultra-high-speed journey back to Earth, using the 600,000 kg of Martian atmospheric gas as propellant. The landing module is left behind orbiting Mars in the parking orbit for reuse on future exploration missions to the Martian surface.

Upon approaching Earth, the self-refueling transfer vehicle executes a decelerating retromaneuver using its ECR engine to reduce the asymptotic approach velocity to 3.00 km/s. The retrocapture maneuver at Earth and all subsequent retromaneuvers are executed by refueling retropropulsion. Table 4 describes five successive refueling retromaneuvers designed to bring the vehicle back to LEO while simultaneously refueling it for the next mission.

Figures 3 and 4 are graphs describing the vehicle's deceleration and accumulated ingested propellant as functions of time, respectively, corresponding to three refueling retromaneuvers. These maneuvers are identified on the figures by A, B, and C and correspond to the first refueling retromaneuver given in Tables 1, 3, and 4, respectively. Notice that the vehicle's deceleration has smooth buildup and decline profiles that can be easily tolerated by human passengers. Since the deceleration loads are always below 2 g and always take place along the vehicle's longitudinal axis, it will be relatively easy to design the vehicle to withstand these decelerations. Detailed computer simulations of the flight dynamics of a self-refueling vehicle during these refueling retromaneuvers are given in Ref. 1.

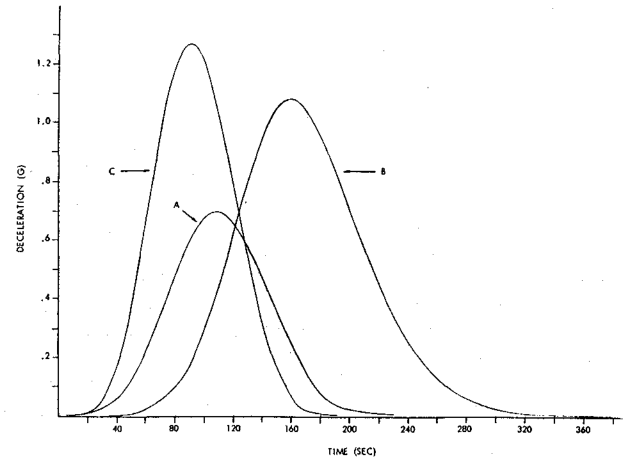


Fig. 3 Vehicle deceleration vs time during refueling retromaneuvers.

Table 4 Flight parameters corresponding to five refueling retromaneuvers at Earth ($V_\infty = 3$ km/s)

Parameter	1st retro	2nd retro	3rd retro	4th retro	5th retro
a_1	−44,289	39,356	14,619	10,054	7,497
e_1	1.1454	0.8364	0.5594	0.3593	0.1400
p_1	66.00	66.00	68.00	68.00	74.80
M_1	800,000	864,383	938,641	1,005,528	1,098,017
ϕ_1	−4.883	−4.510	−3.908	−3.350	−2.079
θ_1	−9.146	−9.904	−10.906	−12.711	−17.102
V_1	11.488	10.623	9.783	9.128	8.355
h_{min}	64.915	64.665	66.404	65.738	71.527
d_{max}	1.268	1.031	0.660	0.571	0.213
P_{max}	57,412	46,666	29,855	25,858	9629
V_2	10.621	9.782	9.127	8.352	7.854
θ_2	10.262	11.452	13.097	18.251	39.400
ϕ_2	4.701	4.155	3.519	2.444	0.769
θ_{12}	19.408	21.356	24.003	30.962	56.502
T_{12}	198	236	286	402	796
m_p	64,838	73,813	66,906	92,575	69,394
ΔV	0.862	0.837	0.652	0.772	0.499
M_2	864,838	938,651	1,005,547	1,098,103	1,167,411
A_2	65,901	16,424	7293	2174	220
e_2	0.8364	0.5597	0.3596	0.1410	0.0138
a_2	39,355	14,616	10,052	7490	6504
T_{23}	10.791	2.442	1.393	0.896	0.725

Summation of refueling retromaneuvers at Earth

Total retro ΔV	3.622 km/s
Total atmosphere ingested	367,526 kg
N_2	282,995 kg
O_2	84,531 kg

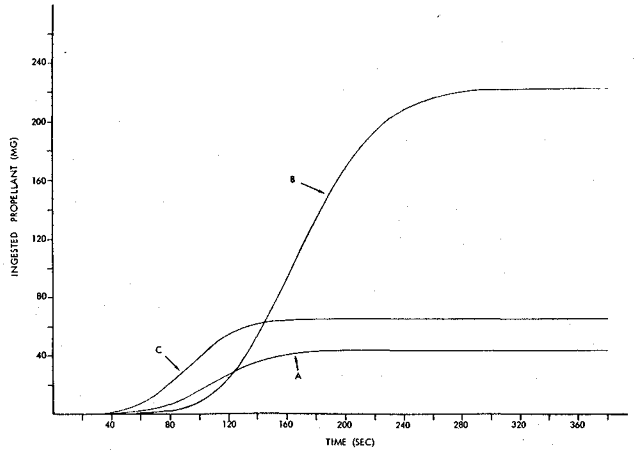


Fig. 4 Accumulated ingested propellant vs time during refueling retromaneuvers.

It should be pointed out that in the round-trip mission to Mars previously described, the self-refueling transfer vehicle does not have to return to Earth after arriving at Mars. It could proceed to travel deeper into interplanetary space taking the landing module with it. For example, after leaving Mars, the vehicle could proceed to one of the many satellites of Jupiter. The Mars-Jupiter flight time would be only about four months. The vehicle would approach Jupiter on a hyperbolic trajectory and execute a long refueling retromaneuver by passing through a portion of its atmosphere designed to place the vehicle on a low-energy elliptical trajectory that intercepts the target satellite. After landing on the satellite's surface with the landing module and conducting several days of exploration, the module could return to the self-refueling mother ship and the ship could then proceed to some other celestial body.

If the vehicle is supplied with a few thousand kilograms of fissionable material and if the life-support system were regenerative, the vehicle and crew could travel around the entire solar system for many years without ever having to return to Earth to replenish the onboard propellant supply or to refurbish the life-support system. Every planet in the solar system, with the possible exception of Mercury and Pluto, has an atmosphere that can be utilized for carrying out refueling retromaneuvers. The gas collected from one planet's atmosphere during the refueling retropropulsive maneuvers will provide the propulsive working fluid to propel it away from the planet to another planet. Thus, a single self-refueling reconnaissance vehicle could carry out a whole series of high-speed interplanetary flights from one planet to another planet, represented by a sequence $P_1 - P_2 - P_3 - \ldots$, that includes going into actual stopover, parking orbits at each planet P_i (or satellite thereof) and conducting surface explorations using the landing module.[1] The basic cost would be represented by the cost of the fissionable fuel consumed by the nuclear power plant.

This method of interplanetary space travel based on a generalized theory of rocket propulsion utilizing a series of successive planetary encounters to generate propulsive thrust is similar, in many respects, to the concept of gravity-propelled space travel,[11-18] propularly called "gravity-assisted" or "swing-by" trajectories. In this concept, a vehicle can also travel around the solar system indefinitely, propelled from one planet to another planet with radical trajectory changes by also utilizing each planet as a means for generating propulsive thrust. But in this concept, the vehicle utilizes each planet's gravitational field as a propulsive thrust source instead of utilizing its atmosphere as propulsive working fluid. Actually, the invention[11] of gravity-propelled trajectories represented a partial solution to the mass ratio problem of classical rocket propulsion because, in principle, a gravity-propelled space vehicle does not require any rocket propulsion system after it is launched. Unfortunately, under this theory of gravitational propulsion it is not possible to decelerate a vehicle relative to an approaching planet in the series and go into orbit around it. It is therefore a theory of propulsion more suitable for unmanned interplanetary reconnaissance vehicles such as Pioneer and Voyager. However, in the generalized theory of rocket propulsion, such decelerating stopover maneuvers is one of its basic features. It is therefore ideally suited for manned space travel.

Summary and Conclusions

This paper introduces a fundamentally new principle and theory for propelling space vehicles. It is based upon utilizing a planet's atmosphere as propellant and generating both decelerating and accelerating propulsive thrust by ingesting propellant that is initially at rest and stored outside the vehicle or by expelling propellant that is initially at rest and stored inside the vehicle respectively. According to this theory, decelerating propulsive thrust is generated by allowing the vehicle to pass through the tenuous upper reaches of a planet's atmosphere in order to ingest portions of it at high speed by means of an inlet diffuser mounted on the front of the vehicle. The underlying thrust-generating mechanism that creates the decelerating retropropulsion is based on transferring momentum from the vehicle moving at orbital velocities to the ingested atmospheric gas that was initially at rest outside the vehicle in accordance with the principle of conservation of linear momentum. It is a reverse application of the basic theory of classical rocket propulsion.

This basically simple thrust-generating mechanism has important consequences because it provides a means for generating the required decelerating propulsive thrust by ingesting propellant instead of by expelling propellant as in classical rocket propulsion. The ingested decelerating propellant is cooled, liquefied, stored onboard the vehicle, and utilized later as accelerating propellant for generating accelerating propulsive thrust by expelling it out an exhaust system at high velocity according to the theory of classical rocket propulsion.

The symmetry of this generalized theory of rocket propulsion is important because it reduces the process of propelling a space vehicle to a process where the reactive working fluid (i.e., propellant) is simply transferred from one storage location (off the vehicle) to another storage location (in the vehicle). By expelling the propellant with an exhaust velocity exceeding the ingestion velocity, this generalized theory of rocket propulsion offers the possibility of conducting space travel without ever having to replenish the vehicle's supply of onboard propellant from any outside source—thus rendering the vehicle self-refueling. Moreover, by expelling the propellant with a velocity significantly above the ingestion velocity, it is possible to generate substantially more propellant than is needed to operate the vehicle. This excess propellant will enable the vehicle to transfer high-mass payloads from low-energy orbits to high-energy orbits and/or to conduct ultra-high-speed interplanetary space travel without exhausting the vehicle's supply of onboard propellant. The theory therefore has the potential of reducing the cost of space travel by many orders of magnitude.

It should be emphasized that this paper is not intended to present detailed solutions to all of the various technical problems that will inevitably arise when contemplating the design and construction of self-refueling space vehicles operating under this theory. Rather, the paper is intended to introduce the generalized theory of rocket propulsion, to demonstrate its potential for achieving economical space travel, and to suggest general solutions to the most difficult technical problems so that this theory can be put into practice.

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