Smart Propellant

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ABSTRACT

It costs thousands of dollars to put a kilogram of anything into orbit, including propellant. For many missions, one can significantly reduce the required on-orbit propellant mass by replacing cheap, "dumb" propellant with more expensive "smart" propellant composed of individual pico-, or nanospacecraft. The key is to use controlled ejection velocities and orbital mechanics to put these spacecraft on precise trajectories that eventually return them back to the host spacecraft for re-use. Each "smart propellant" spacecraft has on-board navigation, attitude control, and propulsion systems that enable fine-tuning of their trajectories for recapture. The ejected spacecraft mass, minus the expended on-board propellant mass for trajectory modification, can be re-used again and again. Smart propellant applications include orbit rephasing, orbit raising and lowering, and landing (plus subsequent take-off) on airless bodies. Required smart propellant ejection velocities range from tens of meters per second for rephasing to ten's of kilometers per second for orbit raising in low Earth orbit. This paper presents results from orbital analyses of the above applications, their impact on smart propellant spacecraft design, and the potential use of mass-produced smart propellant pico- and nanospacecraft for human and robotic exploration of the Moon in the next decades.

I. INTRODUCTION

Rocket propulsion is based on the high-speed ejection of propellant mass. Propellant mass, once ejected, typically does not return and the total mass of the spacecraft plus propellant decreases with each propulsive maneuver. The change in spacecraft velocity ΔV (delta-V) is a function of how much propellant mass M_p was ejected, and the exit speed of that mass with respect to the spacecraft. The rocket equation, given by:

$$\Delta V = g_o I_{sp} \ln (M_i/M_f) \tag{1}$$

relates the change in spacecraft velocity to the specific impulse I_{sp} and the change in total spacecraft mass from an initial M_i to a final M_f. Ejected propellant mass M_p is the difference between M_i and M_f, g_o is the gravitational acceleration constant at the Earth's surface (9.8 meter/s^2), and I_{sp} is the ratio of thrust divided by the mass flow rate. Figure 1 shows propellant mass fractions M_p/M_i, calculated using Eq. 1, required to reach various velocity increments for several values of specific impulse. The curves are representative of cold gas thrusters (~50-s), small solid rockets or hydrazine thrusters (~200-s), bipropellant thrusters (~300-s), hydrogen/oxygen thrusters (~450-s), hot hydrogen thrusters (~900-s), and ion thrusters (~3000-s).

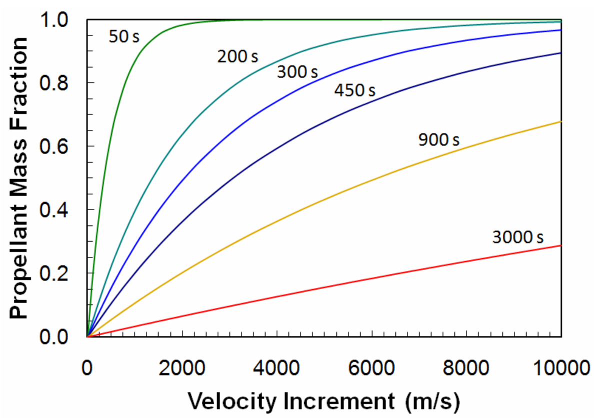

Figure 1. Propellant mass fractions vs. velocity increment for representative specific impulses.

Cold gas thrusters are the simplest, but are useful for velocity increments below about 300-m/s. Chemical thrusters with specific impulse between 200-s and 450-s are more complex, but they enable significantly lower propellant mass fractions. Chemical thrusters have been the primary workhorses of the Space Age; they regularly launch spacecraft into orbit and have propelled space probes beyond Pluto's orbit. Nuclear and solar thrusters can provide a 900-s I_{sp} with hydrogen propellant, but these have only been demonstrated in ground tests. Electric thrusters top the specific impulse range, but these are typically low-thrust (less than 1-N) devices.

The main reason electric thrusters provide low thrust is that the power required to produce a Newton of thrust increases proportionally with specific impulse. Note that the combination ($g_o I_{sp}$) is the directed exit speed V_e of the propellant mass; a 200-s I_{sp} thruster, for example, has a directed exit speed of 2.0-km/s while a 3000-s thruster has a directed exit speed of 30-km/s. The kinetic power P_{KE} required to maintain the exhaust plume is proportional to the mass flow rate *dm/dt* and the square of the directed exit speed V_e:

$$P_{KE} = 1/2\ dm/dt\ V_e^2. \quad (2)$$

Since thrust *T* is proportional to mass flow rate times velocity;

$$T = dm / dt\ \ V_e, \quad (3)$$

the power per unit thrust is proportional to V_e, and thus, specific impulse I_{sp}.

Figure 2 shows the power required to generate a Newton of thrust as a function of specific impulse, and the energy density of the propellant in the exhaust stream, assuming complete conversion of input power into directed plume power. Cold gas thrusters utilize propellant thermal energy densities at typical spacecraft temperatures that range from few tenths to ~2-MJ/kg. Chemical thrusters use propellants with chemical potential energy densities up to a few tens of MJ/kg. To get specific impulses beyond 500-s, addition of external energy (e.g., thermal or electric power from solar cells or nuclear reactors) to the propellant stream is currently required. Note that nuclear fuels have potential energy densities about a million times higher than chemical propellants; up to tens of TJ/kg. If these could be used directly as propellants, high thrust at 500,000-s I_{sp} or higher would be possible. For now, however, we must rely on low thrust electric propulsion for I_{sp} above 500-s, or develop hydrogen thermal thrusters for ~900-s.

Figure 2 shows that a megawatt of chemical power is generated by a kilonewton thruster (enough to barely lift the author at the Earth's surface) at 200-s I_{sp}. The U.S. Space Shuttle solid rocket motors generate 30 gigawatts of power to generate a total thrust of 25-MN at 242-s I_{sp}; this is equivalent to the average electrical power usage in the entire state of California. Rocket propulsion can require a lot of power!

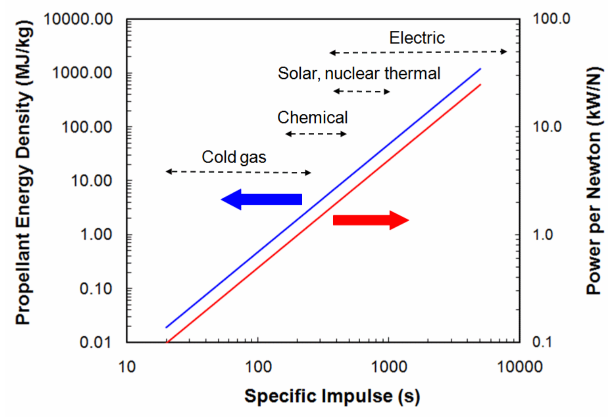

Figure 2. Propellant energy density and power per Newton for ideal thrusters as a function of I_{sp}.

One way to reduce the power requirements for propulsion is to recycle propellant. If some fraction of the ejected propellant could be returned to the thruster for reuse, the same thrust could be maintained by using a higher mass flow rate through the thruster with a lower exit velocity and correspondingly lower power requirement. A 90% recycling fraction, for example, would enable a 10 X reduction in exhaust velocity and a 10 X reduction in thruster power for the same thrust and overall effective specific impulse. The key is to get ejected propellant to return to a spacecraft.

2. THE SMART PROPELLANT CONCEPT

Unlike gases, microscopic solid particles, and/or plasmas typically ejected by thrusters, macroscopic solid particles confine ejected propellant mass so that it can be aimed at a target. Solid projectiles, ejected by an electromagnetic accelerator located on a celestial body, have been proposed as momentum transfer agents that impact a target spacecraft to create thrust.[1,2] Robert C. Willis, in particular, has a patent based on "smart" projectiles that fine tune their trajectories in order to enable capture by the target spacecraft.[3] Two mass accelerator/decelerators are required; one on a celestial body like the Moon or Earth that provides the reaction mass, and another on the target spacecraft.

Through intelligent use of orbital mechanics, to be illustrated in the next section, one can return ejected solid propellant units to a spacecraft. Instead of "smart" projectiles, we use "smart propellant"; reaction mass is launched with the spacecraft and not at a particular spacecraft, and is reused multiple times. Ejection and return both generate an impulse, and in many cases, this amplifies the effect of propellant recycling. Each smart propellant unit is actually a small spacecraft with communications, attitude determination and control, position determination, and thrusters for fine-tuning trajectories.

It costs thousands of dollars to put a pound of anything into orbit; why not replace cheap "dumb" on-orbit propellant with more expensive but reusable, engineered "smart" propellant? This approach does not require an established in-space mass accelerator infrastructure for operation, and can be used throughout the solar system. It could make human and robotic exploration of the moon and Mars in the next decades more economical by drastically reducing the amount of initial propellant required on orbit.

3. ORBITAL MECHANICS

3.A. Rephasing

Rephasing is a maneuver that changes the true anomaly of a spacecraft in orbit. In practical terms, it changes where a spacecraft is along its orbit, without changing the other orbital parameters. Rephasing is typically used to change when a spacecraft flies over a given part of the Earth, or for geosynchronous satellites, to move a spacecraft over different regions of the Earth. The true anomaly of an individual satellite in circular orbit is typically changed by temporarily moving to a different altitude with a different orbital period, remaining at that altitude until the appropriate angular phase change has accumulated, followed by a return to the original altitude. For orbiting satellites, the orbital period τ is given by:

$$\tau = 2\pi (a^3/\mu)^{1/2} \qquad (4)$$

where a is the magnitude of the semi-major axis and μ is the gravitational constant G times the mass of the primary body; for Earth satellites, this is numerically equal to 398600.44 km^3/s^2. The phase (true anomaly) change $\Delta\theta_d$ that occurs while occupying a different altitude drift orbit is given by:

$$\Delta\theta_d = 2\pi t_d (\tau_0 - \tau_1)/\tau_0 \tau_1 \qquad (5)$$

where t_d is the time at new altitude, τ_0 is the original orbit period and τ_1 is the orbit period at the new altitude. A higher temporary altitude results in an *increased* orbit period and a *negative rate of change* in true anomaly.

Figure 3 shows the propellant mass fraction required to produce a 180^o phase change for a spacecraft in a 700-km altitude circular orbit using a 220-s I_{sp} thruster as a function of maneuver time. Under these conditions, a 17-day maneuver time consumes 0.5% of the initial total spacecraft mass as propellant. Faster maneuvers require higher velocity increments and higher propellant mass fractions. If 10% of the initial spacecraft mass were allocated to propellant, only 20 of these 17-day maneuvers could be performed over the entire life of the satellite.

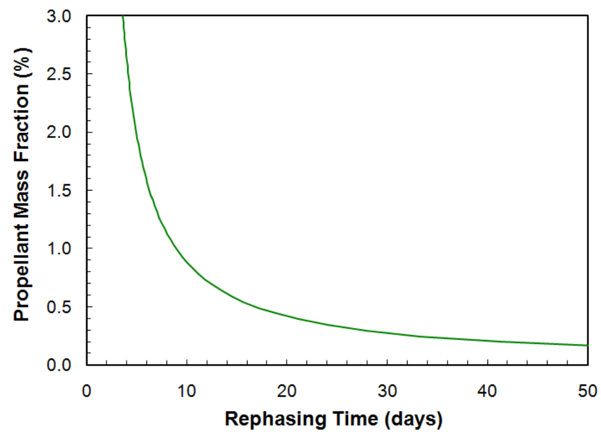

Figure 3. Propellant mass fraction required to rephase a satellite in a 700-km altitude circular orbit by 180^o using a 220-s I_{sp} thruster.

For rephasing maneuvers using smart propellant, smart propellant masses between 0.1 and 10% of the host spacecraft would be ejected at speeds less than a few hundred meters per second. Since smart propellant can be reused, a single spacecraft could perform thousands of rapid, large angle rephasing maneuvers over its lifetime. Just how can ejected propellant mass be returned to a spacecraft for a rephasing maneuver?

Figure 4 shows a schematic drawing of spacecraft and smart propellant orbits in an Earth-centered inertial reference frame before and after smart propellant ejection. In this case, the spacecraft of mass M_s starts in a 700-km altitude orbit and ejects a smart propellant mass M_p at relative speed V_e in the forward flight direction. The spacecraft gets a velocity increment ΔV_s of magnitude

$$\Delta V_s = V_e / [1+ (M_s/M_p)] \qquad (6)$$

in the retrograde direction, and thus enters an elliptical orbit with a perigee that is lower than the original orbit altitude, resulting in a shorter orbit period. The smart propellant gets a velocity increment ΔV_p of magnitude

$$\Delta V_p = V_e / [1+ (M_p/M_s)] \qquad (7)$$

in the posigrade direction, and thus enters an elliptical orbit with an apogee that is higher than the original orbit altitude, resulting in a longer orbit period. Note that 35 minutes after ejection, the spacecraft has traveled further in angle around Earth than the propellant.

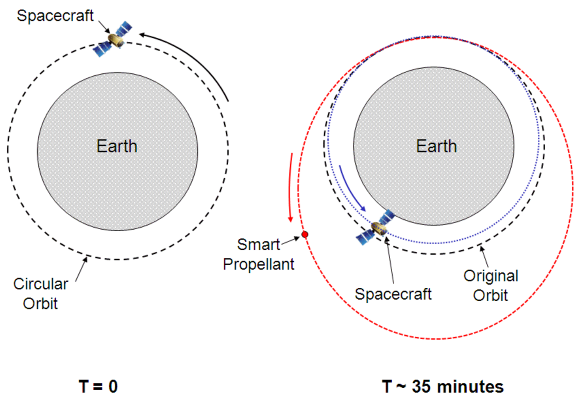

Figure 4. Schematic drawing of spacecraft and smart propellant orbits before and after smart propellant ejection in an Earth-centered inertial frame. (drawing not to scale)

In a reference frame centered on the original spacecraft and rotating at the original orbital rate, the post-ejection spacecraft drifts predominantly forward while the smart propellant mass drifts predominantly rearward. Figures 5 and 6 show the initial trajectories of the smart propellant, and spacecraft, in this reference frame for the first 150 minutes after ejection. In this case, the propellant mass is 1% of the spacecraft mass, the propellant was ejected at 100 m/s, and each data point is 90 seconds apart. The spacecraft and smart propellant return to their initial altitude (zero radial displacement) once per orbit, but their orbit periods are different. Note that the spacecraft moves about 17.5-km forward per initial orbit period while the propellant moves 1750-km backward.

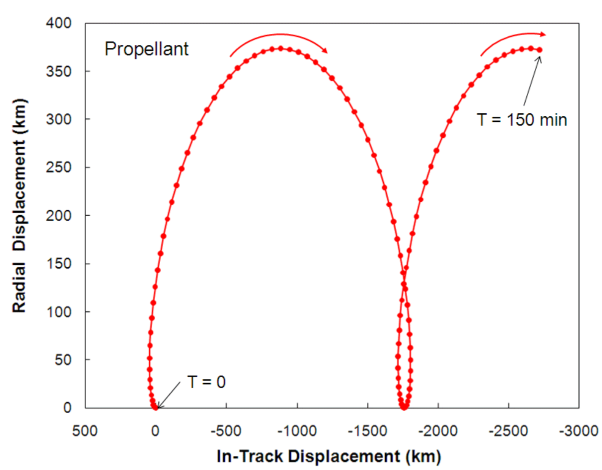

Figure 5. Propellant trajectory as viewed in a co-orbital reference frame rotating at the initial spacecraft angular rate.

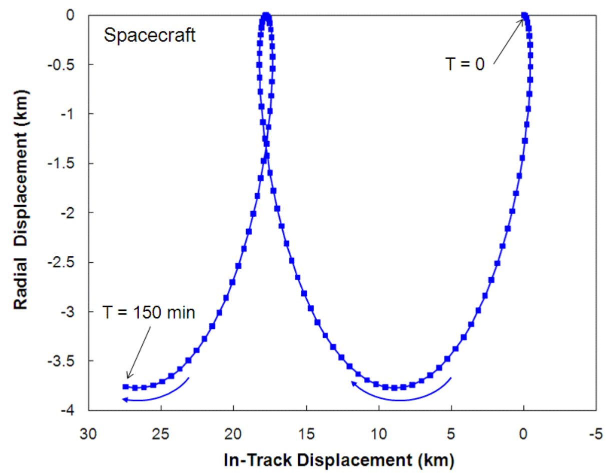

Figure 6. Spacecraft trajectory as viewed in a co-orbital reference frame rotating at the initial spacecraft angular rate.

Figure 7 shows three schematic snapshots of trajectory evolution in this rotating reference frame on a larger scale. Figure 7A shows propellant ejection in the forward flight direction, Fig. 7B shows a snapshot in the rotating reference frame 4 propellant orbits after ejection. The propellant has a longer orbital period than the original spacecraft orbital period, so it moves generally clockwise in this rotating reference frame. The spacecraft has a shorter period due to the impulse at ejection, so it moves counterclockwise in this reference frame. If the initial ejection velocity was adjusted properly, the spacecraft and smart propellant mass come together at the original orbit altitude N propellant mass orbits later, as shown in Fig. 7C. Note that the smart propellant mass impacts the satellite at relative speed V_e from the retrograde direction, thus imparting a positive impulse to the spacecraft. The magnitude of the recapture impulse is equal to the initial ejection impulse, thus leaving the spacecraft plus smart propellant mass system in the initial circular orbit, but with a different true anomaly. The key is to choose initial ejection conditions to assure that the spacecraft and smart propellant mass meet N propellant orbits later.

Figures 8 and 9 show rephase maneuver time as a function of ejection velocity for a 700-km altitude circular orbit with smart propellant mass ratios (M_p/M_i) of 1% and 10%, respectively. The phase change is 3.60° for the 1% mass fraction and 36.0° for the 10% mass fraction. Note that the ejection velocities are quantized; specific velocities are required to ensure spacecraft and smart propellant convergence at the appropriate time.

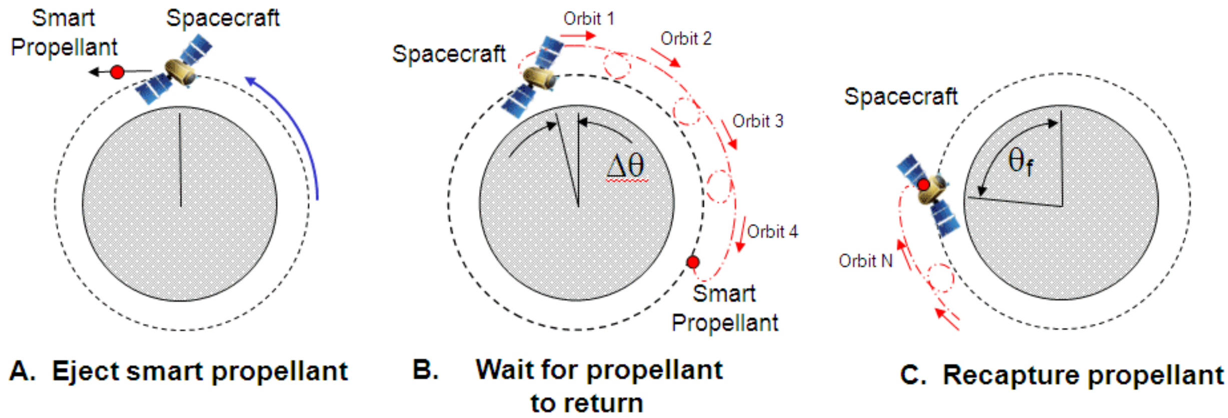

Figure 7. Schematic sequence of a smart propellant rephasing maneuver. (Not to scale)

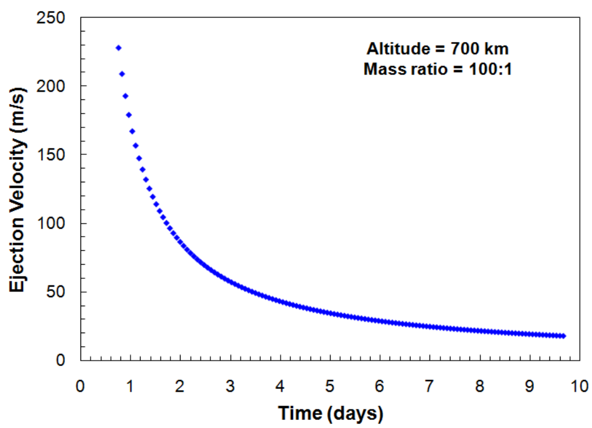

Figure 8. Ejection velocity vs. rephase time for a smart propellant rephasing maneuver with smart propellant mass ratio of 1%.

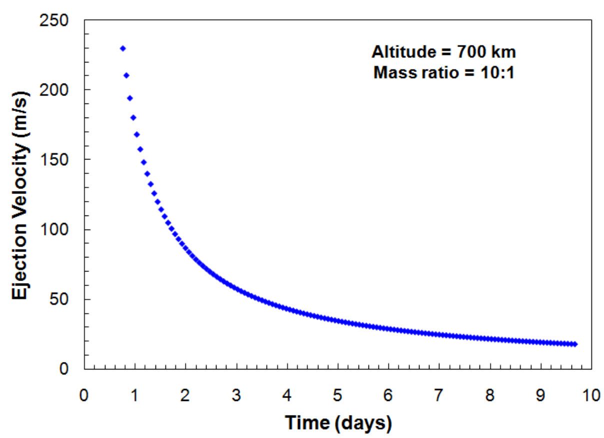

Figure 9. Ejection velocity vs. rephase time for a smart propellant rephasing maneuver with smart propellant mass ratio of 10%.

The data points in Figures 8 and 9 correspond to an integer number *N* propellant orbits and *N+1* spacecraft orbits where *N* ranges from 10 to 140. Note that Figs. 8 and 9 look almost identical, but the 36^o phase change in Fig. 9 is ten times larger than the 3.6^o phase change in Fig. 8. A 17-day, 180^o rephase maneuver, like the chemical thruster maneuver mentioned on page 3, could be performed using 5 successive 36^o steps, each lasting 3.4 days. From Figure 9, we see that the ejection velocity for this case would be 51.2 m/s. Based on the mass fraction for the chemical thruster maneuver (0.5%), the exit velocity of the chemical thruster (2.2-km/s), and the mass fraction for the smart propellant (10%), one would estimate a required smart propellant exit velocity of:

$$(0.5\% / 10\%) * 2.2\text{-km/s} = 110\text{-m/s}. \qquad (8)$$

The actual required exit velocity is about half of this.

Orbit rephasing is a maneuver that does not change the total energy of the spacecraft; the semi-major axis of the orbit is not affected. In the traditional rephase maneuver, thrusting is performed both parallel and anti-parallel to the flight direction with equal magnitudes, thus resulting in a zero net change in spacecraft velocity at the end of the maneuver. With smart propellant, the reversing impulse is free; it occurs when the propellant recontacts the spacecraft. The smart propellant maneuver is therefore twice as efficient as the conventional thrusting maneuver. In addition, if the kinetic energy of the returning propellant can be stored for reuse during the next ejection, the net energy usage is ideally zero.

Based on Figures 8 and 9, one sees that smart propellant must be ejected from the host spacecraft at velocities between 20 and 200 m/s for rephasing times between 1 and 9 days at 700-km altitude. The phase change per jump is proportional to the smart propellant mass fraction, with phase changes ranging from a few degrees to almost 40 degrees for smart propellant mass fractions between 1% and 10%. Unlike the chemical thruster rephasing maneuver, the smart propellant rephasing maneuver can be done over and over again, potentially enabling thousands of rephasing maneuvers.

Ideally, no propulsion is required other than the ejection of smart propellant. In practice, a number of effects such as ejection velocity errors, differential drag, and orbit perturbations due to higher geopotential terms need to be counteracted using propulsion on board the smart propellant unit.

3.B. Temporary Apogee/Perigee Modification

As shown in Figure 4, smart propellant rephasing temporarily decreases spacecraft perigee when the desired phase change is positive. When the desired phase change is negative, spacecraft apogee temporarily increases. Space and Earth environmental sensing missions can benefit from this ability to change altitude ranges, particularly if the range can be changed many times.

Figure 10 shows spacecraft perigee and smart propellant apogee altitudes for the rephasing conditions used to generate Fig. 9 (10% smart propellant mass fraction, 700-km circular orbit). Spacecraft perigee can be reduced by almost 100-km over 11 orbits (0.75 days) using a smart propellant ejection speed of 230-m/s (see Fig. 9).

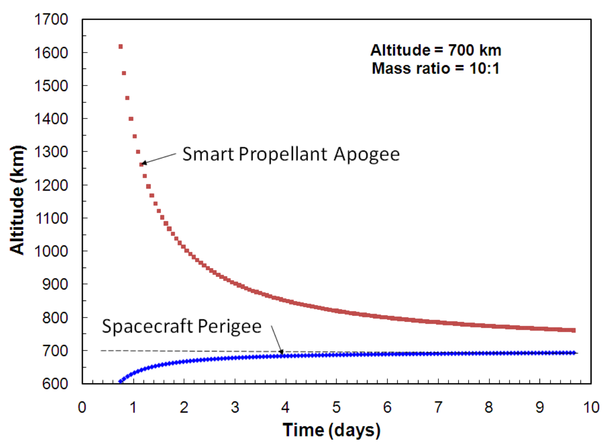

Figure 10. Spacecraft perigee and smart propellant apogee for a smart propellant rephasing maneuver with smart propellant mass ratio of 10%.

If space environmental sensing is desired, and the sensors can fit within the desired smart propellant mass limit, smart propellant units can function as smart environmental sensors. These smart sensors return to the spacecraft for data download and re-ejection into new orbits. This approach is very favorable for high data rate sensors; hundreds of gigabytes can be integrated into a sub-kilogram mass module and downloaded within hours to the host satellite once docked. Figure 11 shows spacecraft perigee and smart propellant apogee altitudes for the rephasing conditions used to generate Fig. 8 (1% smart propellant mass fraction, 700-km circular orbit). There is a small change in spacecraft perigee, but the smart propellant can access altitudes above 1600-km.

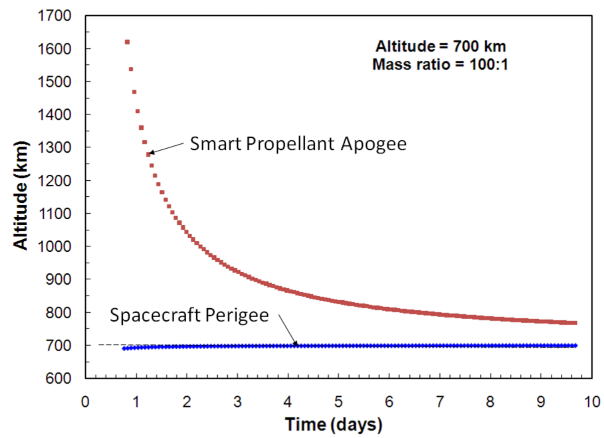

Figure 11. Spacecraft perigee and smart propellant apogee for a smart propellant rephasing maneuver with smart propellant mass ratio of 1%.

3.C. Lunar Surface Shuttle

During the U.S. Apollo program, the Lunar Excursion Module (LEM) took astronauts from the Command and Service modules in Low Lunar Orbit (LLO) to the Moon's surface and back. The LEM used space-storable bipropellants with a specific impulse of 311-s. It had two stages to minimize initial mass, both of which could be used only once.

The lowest velocity increment to go from a circular orbit to surface landing results from an orbit slightly above the surface, with an impulse large enough to cancel the circular orbit velocity V_o. The vehicle then drops to the surface. Launch back into orbit requires the reverse process with a short vertical ascent to orbit altitude, followed by another horizontal impulse of equal magnitude to the original deorbit maneuver. The total minimum velocity increment for landing and return to orbit is therefore $2V_o$.

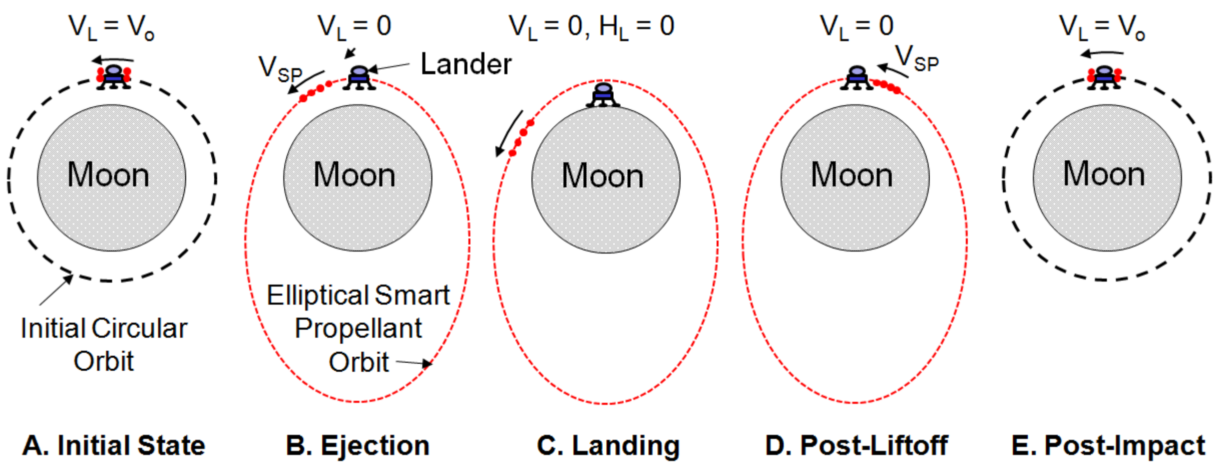

Figure 12. Schematic sequence of a smart propellant lunar landing and re-orbit maneuver. (Not to scale)

A spacecraft in circular orbit about a primary body has an orbital velocity V_o given by

$$V_o = (\mu / a)^{1/2}. \quad (9)$$

For an orbit just meters the lunar surface, $V_o = 1681$ m/s. Table 1 lists the minimum propellant mass fractions required to perform the surface landing and return mission once, twice, and three times using a single vehicle with a specific impulse of 311-s. Performing the round trip once is challenging, but repeating it more than once using standard rocket propulsion becomes impractical due to the vanishingly small (<10% at best) payload mass fractions.

Table 1: Minimum Velocity Increment and Propellant Mass Fraction at 311-s I_{sp} for Lunar Landing and Return to LLO.

Number of Missions	Velocity Increment	Propellant Mass Fraction
1	3362 m/s	0.668
2	6724 m/s	0.890
3	10086 m/s	0.963

On-orbit refueling is one option for a reusable lunar surface shuttle, but the propellant for each landing and return will have to be brought from the Earth in the foreseeable future. Smart propellant offers significant propellant recycling, thus enabling part of a sustainable lunar transportation architecture without constantly launching propellant. Figure 12 shows a schematic sequence of the lunar landing and return mission using smart propellant. In Fig. 12A, a lunar landing vehicle with smart propellant is in a LLO with $V = V_o$. In Fig. 12B, smart propellant is ejected in the forward flight direction into a higher-energy elliptical orbit with $V = V_{SP}$ while the lander orbital velocity V_L is reduced to zero. In Fig. 12C, the lander has dropped to the lunar surface, using some on-board propellant for a soft landing. For a 1-km drop, the soft landing delta-V is about 60-m/s; much less than the original 1681-m/s orbital velocity. In Fig. 12D, at the appropriate time, the lander rises from the surface, again using a small amount of on-board propulsion. In Fig. 12E, the returning smart propellant impacts the lander from the original anti-flight direction, putting the entire lander and smart propellant system back into the initial circular orbit. Note that the initial orbital energy and mass of the smart propellant plus spacecraft system is relatively unchanged by the mission (zero net energy usage), and can therefore be repeated many times.

The post-ejection smart propellant velocity V_{SP} in this application ranges from V_o to V_e where V_e is the local escape velocity. Escape velocity is equal to the local circular orbit velocity times $\sqrt{2}$. Figure 13 shows the smart propellant mass fraction, as a function of V_{SP} / V_o, to produce a lander orbital velocity of zero. The lowest smart propellant mass fraction of 71% occurs near $V_{sp} = 1.4\ V_o$.; about 2350-m/s for LLO. This seems rather high, but it is only slightly higher than the minimum propellant mass fraction from Table 1 for a single mission with conventional thrusters at 311-s I_{sp}.

Figure 13 is valid for any airless primary body such as an asteroid, moon, dwarf planet, or planet. Post-ejection smart propellant velocities are proportional to the circular orbit velocity, but the mass ratios remain the same. Table 2 lists the minimum and maximum post-ejection smart propellant velocities for landing and re-orbit about various bodies in our solar system. For the smaller bodies with low orbit velocities, conventional rocket propulsion offers a lower propellant mass fraction even for multiple landings. The smart propellant approach is beneficial primarily for the larger moons in the solar system.

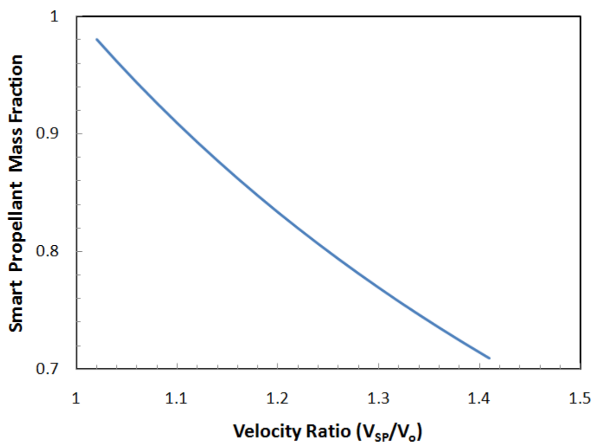

Figure 13. Smart propellant mass fraction as a function of velocity ratio for landing and re-orbit.

Table 2: Surface orbit and escape velocities, and minimum propellant mass fractions using 311-s Isp thrusters, for representative solar system bodies.

Body	Surface Orbit Velocity	Surface Escape Velocity	Prop. Mass Fraction 1 Trip	Prop. Mass Fraction 3 Trips
Phobos	7.3-m/s	10.3-m/s	0.48%	1.43%
6-Hebe	91-m/s	130-m/s	5.8%	16.4%
2-Pallas	220-m/s	311-m/s	13.4%	35.2%
4-Vesta	248-m/s	351-m/s	15.0%	38.6%
1-Ceres	359-m/s	508-m/s	21.0%	50.7%
Europa	1430-m/s	2020-m/s	60.9%	94.0%
Moon	1681-m/s	2377-m/s	66.8%	96.3%
Callisto	1730-m/s	2440-m/s	67.9%	96.7%
Mercury	3000-m/s	4250-m/s	86.0%	99.7%

3.D. Orbit Raising or Lowering

Orbit raising (or lowering) starts with ejecting propellant in the anti-flight (or flight) direction. The propellant should re-approach the spacecraft from the same direction in order to impart additional momentum, and hence impulse. Figure 14 shows a schematic sequence of one technique, called apoapsis reflection, discussed in reference 4 , that uses this approach to provide orbit raising using smart propellant. Figure 14A shows a spacecraft in circular orbit about a primary body with orbit velocity V_o. In Fig. 14B, smart propellant is ejected in the anti-flight direction with enough speed to put it into a retrograde orbit with an apoapsis higher than the original orbit altitude. The relative ejection speed can be between $2V_o$ and $2.414V_o$, thus yielding V_{sp} in the primary body-centered inertial frame between V_o and 1.414 V_o. The spacecraft receives an increase in velocity, thus injecting it into an elliptical orbit with higher energy as shown in Fig. 14B. When the smart propellant reaches apoapsis at high altitude as shown in Fig. 14C, it's speed will be significantly lower than V_o due to conservation of orbital angular momentum. Figure 14D shows the smart propellant after an impulsive burn large enough to maintain its orbital speed, but in the reverse flight direction. The smart propellant is now in a posigrade orbit with a periapsis equal to the original orbit altitude. If the initial ejection velocity was chosen carefully, the smart propellant and spacecraft will impact at periapsis as shown in Fig. 14E. The spacecraft will receive an additional impulse from the smart propellant, thus raising its apogee even further (see Final Elliptical Spacecraft Orbit in Fig. 14E). The relative impact velocity will range from 0 to $0.414\ V_o$. Some of the initial energy used to launch the smart propellant can be recovered by an appropriately-designed decelerator.

Apoapsis reflection requires conventional thrusters on the smart propellant, and thus does not completely conserve smart propellant mass. However, the velocity increment required for apoapsis reflection can be much smaller than the original orbital velocity, thus saving significant propellant mass. At infinite distance, for example, the required delta-V is zero. Unfortunately, this would take infinite time. Figure 15 shows the smart propellant orbit period and apolune velocity change required for an initial orbit 1-km above the lunar surface. For an apolune of 40,000-km, the smart propellant returns in 79.9 hours and the required delta-V at apolune is 194-m/s.

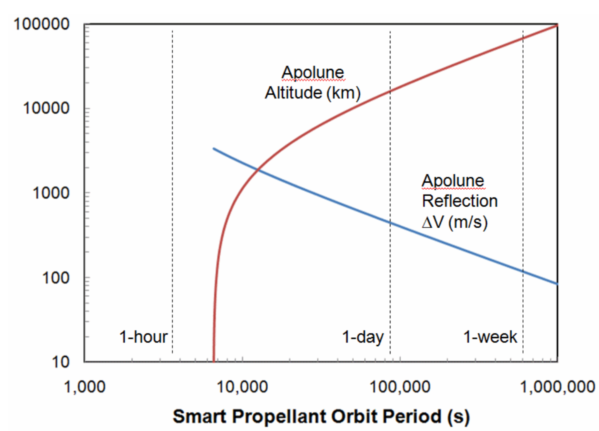

Figure 15. Smart propellant apolune altitude and apolune reflection delta-V as a function of smart propellant orbit period.

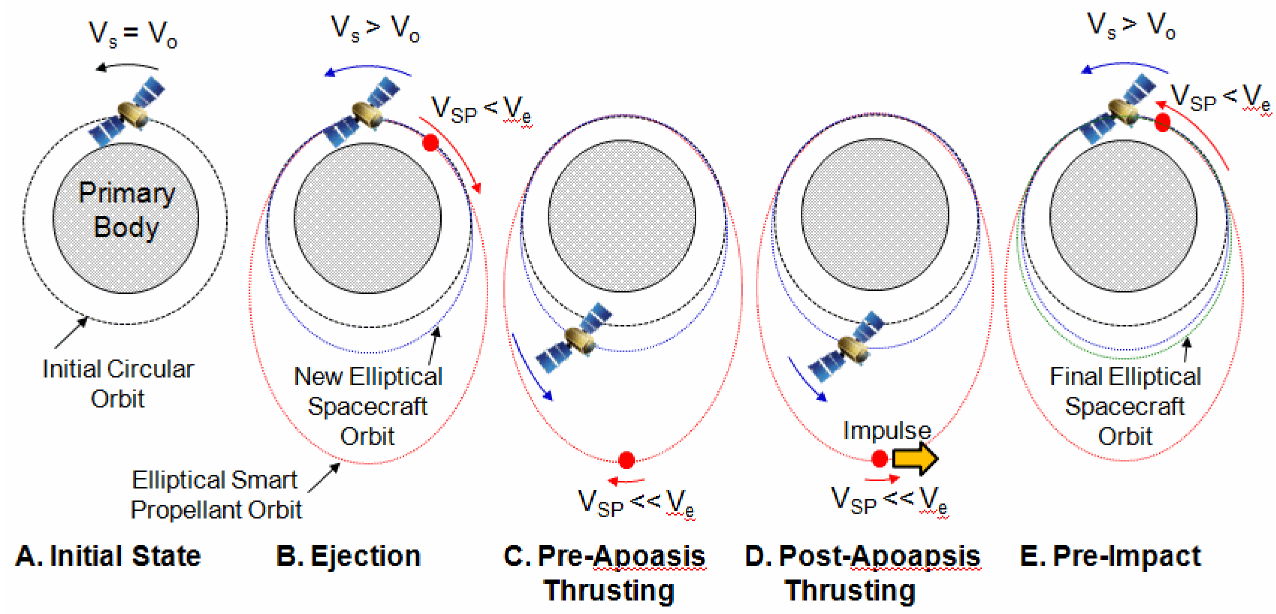

Figure 14. Schematic sequence of an apogee-raising maneuver using smart propellant. (Not to scale)

For a specific smart propellant mass fraction, ejection speeds are quantized. Smart propellant returns to the spacecraft after one smart propellant orbit period. The spacecraft, meanwhile, has to execute an integer number of orbits during this time in order to meet the returning smart propellant. Figure 16 shows the smart propellant orbit period as a function of ejection velocity for a spacecraft in an initial orbit 1-km above the surface of the moon with a 1% smart propellant mass fraction. The different N values indicate how many orbits the spacecraft has performed before the smart propellant returns. For a 1% smart propellant mass fraction and N=25, the smart propellant apolune altitude is 27,050-km, the relative smart propellant ejection speed is 4027-m/s, and the spacecraft apolune increases by 176-km after initial ejection. The spacecraft gains another 12.5-km in apolune altitude when the smart propellant returns with a relative speed of 286-m/s from the anti-flight direction.

This process can be repeated at apolune to boost perilune, thus increasing the overall orbit radius with time. For orbit-lowering, smart propellant would be ejected in the flight direction at speeds between 0 and $0.414\ V_o$, and it would return from the flight direction with speeds between $2\ V_o$ and $2.414\ V_o$. The smart propellant returns with additional kinetic energy due to the change in flight direction. Orbit-lowering with smart propellant converts spacecraft orbit energy into smart propellant kinetic energy. If smart propellant kinetic energy at recapture could be collected and stored at 100% efficiency, a spacecraft could start in high circular orbit, drop down to a lower circular orbit, and return to the original orbit altitude with no net energy usage.

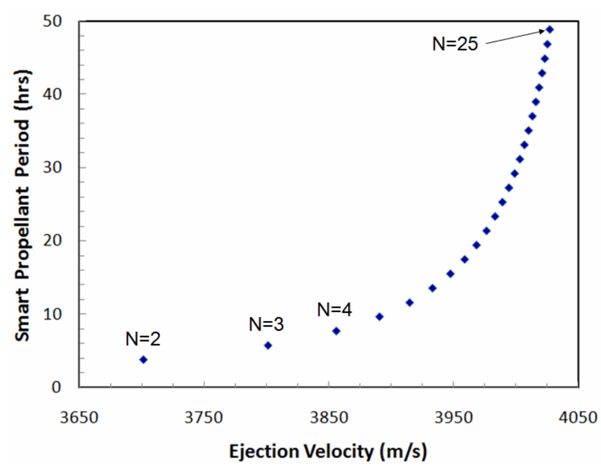

Figure 16. Smart propellant period vs. ejection velocity for orbit raising with a 1% smart propellant mass fraction in a 1-km altitude lunar orbit.

4. SMART PROPELLANT DESIGN

I've given examples of smart propellant orbital mechanics with increasing levels of difficulty. Smart propellant mass fractions have ranged from 1% to 67% of the total spacecraft mass, and these spacecraft have a potential mass range of 10-kg to well over 100,000-kg. A smart propellant unit could therefore have a minimum mass of about 0.1-kg, but 1-to-10-kg is more realistic. Smart propellant units require attitude determination and control, position determination, velocity determination, propulsion, host spacecraft position and velocity determination, and communications to fine-tune their trajectories in order to enable recapture. The Canadian CANX-2 spacecraft is one example of an on-orbit, 3.5-kg mass, "3U" CubeSat that has most of these systems.[5] Some systems like propulsion will need enhancement, and a terminal guidance sensor for relatively high-speed rendezvous is still needed. Unlike CubeSats, however, smart propellant units will be spherical in shape to minimize the effect of orientation errors during recapture. Try catching or hitting a cubic baseball travelling at typical baseball speeds between 60 and 90 miles per hour (27 to 40-meter/s). Note that this is approximately the velocity range for smart propellant orbit rephasing applications in low Earth orbit (LEO).

Precision position and velocity determination can be provided by GPS receivers for LEO applications like satellite rephasing. Commercially-available GPS receivers suitable for 1-to-10-kg class spacecraft are available from a number of vendors such as Surrey Satellite Technology Limited and SpaceQuest Limited.[6,7] The CANX-2 CubeSat used a modified NovAtel receiver for both position determination and GPS occultation measurements.[5] Position accuracy for these receivers are ~10-meters (95% of the time), and the velocity accuracies range from 3 to 15-cm/s. This level of accuracy is sufficient for general trajectory control over at least 99% of the smart propellant orbit.

During the last 100 seconds before rendezvous, during the terminal guidance phase, 1-cm or better accuracy is required to enable recapture of smart propellant without damage to the host spacecraft. Relative position accuracies of ~2-cm are possible using carrier-phase differential GPS.[8] New optical sensors that provide ~1-cm relative position determination for this terminal phase need to be developed.

Smart propellant will need on-board propulsion. The magnitude of the required propulsive delta-V will be a function of the number of potential ejections, accelerator velocity error, and orbit altitude (drag effects). A reasonable order-of-magnitude estimate is to assume a required delta-V of 1% of the ejection velocity for acceleration velocity error, plus an additional 1% for orbit corrections. The delta-V requirement for the final orbit corrections during terminal guidance can be quite small; a 10-m error (from GPS) at the beginning of the 100-s terminal phase requires a correction delta-V of only 0.1-m/s. The first-order orbital analyses presented in the previous sections do not include higher-order geopotential effects starting with J2 (oblateness coefficient). The remaining propellant allocated for orbit corrections is used to counteract these effects, plus air drag and solar pressure. For an orbit rephasing application with 200 ejections and returns, and an average ejection velocity of 50-m/s, one gets a required ΔV of ~200-m/s. From Fig. 1 and a desire to limit the expendable propellant mass to less than 10% of the smart propellant mass, we get a preferred specific impulse greater than ~190-s. While chemical monopropellant thrusters can provide this specific impulse, a better approach is to develop high specific impulse electric microthrusters with I_{sp} in excess of 1000-s for slow corrections over >99% of the smart propellant orbit, and use chemical microthrusters for terminal guidance.

Acceleration (and deceleration) levels will be an important factor in smart propellant design. Figure 17 shows average acceleration level in g's as a function of ejection (or incoming) velocity for accelerator lengths of 0.1, 1, 10, and 100-m. For the smart propellant rephasing mission in LEO with a 50-m/s relative ejection velocity, a 1-meter long accelerator will generate an average acceleration of 128-g's. This is high by an order-of-magnitude by spacecraft standards, but not difficult. The majority of mass in a smart propellant module can be used for structure, and electronic circuit boards can be encapsulated to provide several thousand-g resistance. For smart propellant ejection velocities between 1 and 4-km/s for the lunar applications mentioned in sections 3.C and 3.D, accelerator lengths of 100 to 300-meters would be required to obtain these acceleration levels.

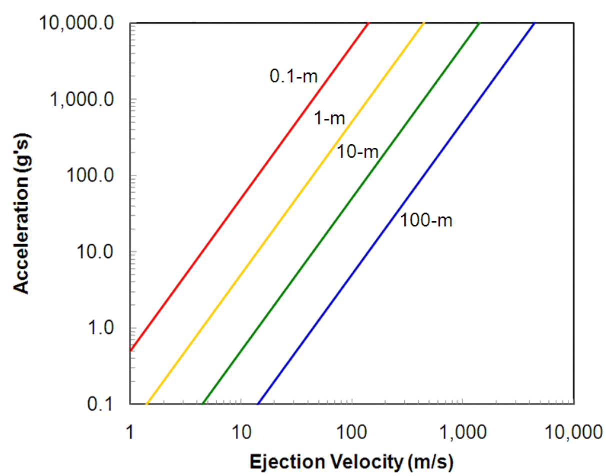

Figure 17. Smart propellant acceleration as a function of relative ejection velocity for accelerator lengths of 0.1, 1, 10, and 100-m.

5. EJECTION AND RECOVERY SYSTEMS

Smart propellant ejection and return velocities can range from a few tens of meters per second for orbit rephasing to a few tens of kilometers per second for apoapsis reflection maneuvers about the Earth or other planet. The ejection system should take electric, chemical, or mechanical energy to accelerate the smart propellant, and be able to reversibly extract and store energy from the incoming propellant. As a starting point, consider that a ~100-kg mass U.S. baseball pitcher is capable of throwing a 0.15-kg mass baseball at speeds up to 100-miles/hour (45-m/s), and a U.S. football player is capable of throwing a 0.42-kg mass football at speeds up to 70-miles/hour (31-m/s) using arm and body muscles within a ~ 2.5-meter length (body tilt plus twice the arm length). Based on the results shown in section 3.A, a baseball or football player stranded hundreds or thousands of kilometers from the International Space Station (ISS) in LEO, in the flight or anti-flight direction, could perform multiple rephasing operations using their appropriate ball to return to the ISS.

CubeSats use a mechanical ejection system based on a compressed spring to provide ejection velocities of 1 to 5-m/s. With some engineering, these springs can be increased in stiffness and length to get ejection velocities up to ~50-m/s. In this case, a 1-m long spring requires a spring constant of about 8,700 N/m (600 lb_f/ft). In principle, the returning smart propellant would recompress the spring, and the compressed spring would be mechanically latched in place to capture the strain energy. An electric motor and drive system would be used to further compress the spring, or readjust compression to set the correct next ejection velocity. The spring approach is simple, but maximum accelerations are about twice the average acceleration. For a 50-m/s ejection velocity and 1-meter long spring, the acceleration (or deceleration) is 255-g's and the acceleration (or deceleration) time is a mere 31-ms. Over 30 can be launched (or retrieved) in a second. A 100-kg mass spacecraft ejecting (or retrieving) 3.5-kg mass smart propellant units would experience a maximum instantaneous acceleration of 8.9-g's. To minimize instantaneous g-loads on the host spacecraft, smart propellant would ideally be broken down into the lightest possible units that are launched sequentially.

Higher relative ejection velocities can be achieved using electromagnetic accelerators. These devices are basically linear electric motors that use switched currents to generate moving magnetic fields that accelerate either a magnetic or electrically conductive object at accelerations up to several hundred thousand g's. Typical accelerations are in the several hundred to several thousand g range. In the case of electrically conductive objects, Eddy currents generated within the object generate magnetic fields that oppose the applied fields. Appropriately-designed electromagnetic launchers can be operated as motors or generators, thus providing the ability to recover kinetic energy from incoming smart propellant units during deceleration.

An early design for launch of raw materials in 20-kg units at a 1-Hz duty cycle off of the Lunar surface at velocities around 2-km/s had a mass of 3,500 tons.[9] Another design suitable for transferring 4,000 tons of cargo from LEO to geosynchronous orbit with ejection velocities in the 5 to 10-km/s range was presented in 1982.[10] Use of smart propellant instead of raw materials as reaction mass would greatly increase the maneuvering capability and utility of such a system. At the other extreme, a demonstration railgun built and fabricated at the Westinghouse R&D Center in Pittsburgh, Pennsylvania, achieved 4,200-m/s ejection using a 317-gram projectile.[11] This device had a length of only 5-m, resulting in a peak acceleration of 230,000-g's. Railguns typically do not operate reversibly in generator mode, but design modifications could enable this capability. Other concepts for electromagnetic accelerators such as the tubular linear electromagnetic launcher are currently being studied for use in space.[12]

6. SUMMARY

Smart propellant can be demonstrated in a LEO orbit rephasing mission in the near-to-mid term (within 10 years). Ejection velocities are modest, and most of the technologies for building nanosatellite-class smart propellant modules exist. Longer term (beyond 20 years) applications hold great promise, especially for manned missions within cis-lunar space, but significant research needs to done in demonstrating space-based electromagnetic launchers with exit velocities in the 1 to 10-km/s range. If successful, today's "throw away" in-space transportation architectures could be replaced by sustainable mass and energy-efficient space architectures based on recycling smart propellant mass.

Acknowledgments

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