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Life Extension Strategies for Shuttle-Deployed Small Satellites Using Pulsed Plasma Thrusters

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At Space Shuttle altitudes, thermospheric drag is the dominant force limiting satellite on-orbit life (typically <100 days). The pulsed plasma thruster is ideally suited to extend the life of small satellites deployed from the Shuttle due to its low system mass and volume, high specific impulse, and inert solid propellant (Teflon®). The objective was to identify and analyze life extension strategies for Space Shuttle-deployed small satellites using the pulsed plasma thruster. A generalized analysis is presented that is applicable to a broad range of satellite and pulsed plasma thruster performance/life characteristics. Within the limits of typical small satellite power to mass ratios, the most capable of these strategies, designated lift and coast, requires the smallest amount of propellant and can extend mission life to one to two years with pulsed plasma thruster technology currently in development.

Nomenclature

A = instantaneous satellite cross-sectional area, m^2
Ap = geomagnetic activity parameter
a = orbit-averaged spacecraft acceleration due to pulsed plasma thruster (PPT) thrust, μg (note 1 μg is 9.81×10^{-6} m/s^2)
a^* = value of a required for orbit-averaged PPT thrust to equal the drag force at the Shuttle-deployed altitude [see Eq. (9)], m/s^2
C_d = instantaneous satellite drag coefficient
$F_{10.7}$ = 81-day centered average of the solar radiative flux at 10.7 cm, Wm^{-2}Hz^{-1}
f_s = fraction of the orbit in sunlight
g = gravitational acceleration at the Earth's surface, 9.81 m/s^2
g_l = gravitational acceleration at the local altitude, m/s^2
I_{sp} = PPT specific impulse, s
i = orbit inclination, deg
j = summation index
m = spacecraft mass, kg
$\dot{m}$ = PPT average mass flow rate, kg/s
m_{prop} = propellant mass expelled by PPT, kg
n = number of orbits from start of mission
P = average power input to PPT, W
R_e = Earth's mean radius, m
r_n = orbit radius after n orbits from start of mission, m
r_0 = initial orbit radius, m
$T_{s,d}$ = time-averaged PPT thrust in sunlight s and in shadow d, N
t = time, s
t_e = Earth rotational period, s
t_n = time after n orbits from start of mission, s
t_p = orbital period, s
t_{pn} = orbital period during orbit n, s
β = orbit-averaged satellite ballistic coefficient, identical to m/C_dA, kg/m^2
η = PPT system efficiency
μ = Earth's gravitational parameter, m^3/s^2
ρ = globally averaged thermospheric mass density, kg/m^3
σ = standard deviation

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Introduction

THE Space Shuttle Hitchhiker Eject System (HES) represents a reliable and cost-efficient means for deploying small satellites into low Earth orbit.[1] Unfortunately, at Space Shuttle altitudes (140–240 n mile), orbital lifetimes are typically less than 100 days, which is too short to be useful for most missions. The objective of this study was to identify and analyze life extension strategies for Space Shuttle-deployed small satellites using a pulsed plasma thruster (PPT). Previous work investigating the use of PPTs for drag compensation is very sparse and illustrative in nature.[2,3] This analysis is an extension of previous work and considers typical small satellite power to mass ratios and next-generation PPT performance in the assessment of the PPT's ability for accomplishing such a mission.

The primary motivation for this study is that the Phillips Laboratory's MightySat Program has identified the Space Shuttle as a possible launcher for its small satellites[4] and will use a PPT to extend the on-orbit life to one year.[5] Elements of this analysis were used to determine the life extension strategy for the 125-kg MightySat II.1 spacecraft to be launched in 1999 (Ref. 5). For this reason, the focus of this paper was satellite life extension to one to two years. Although not discussed, optimum launch windows and satellite drag reduction methods have also been shown to assist the PPT in extending mission life.[6]

The PPT is an electric propulsion device that uses electrical power to ionize and electromagnetically accelerate a plasma to high exhaust velocities (10–20 km/s) (Refs. 3 and 7–11). Its high specific impulse (1000–2000 s) enables significant reduction in propellant mass requirements compared to monopropellant and cold gas systems. The PPT (shown schematically in Fig. 1) is ideally suited to the propulsion needs of small satellites because it is compact, uses an inert solid propellant, is easily integrated to a spacecraft, and has a low system wet mass (<5 kg). It essentially consists of a bar of Teflon®, which is the propellant source, pressed firmly between two electrodes by a negator spring, which is the only moving part. Because of its inert nature when unpowered, the PPT is well suited for minimizing Shuttle-safety-related test and documentation requirements. When provided unregulated power from the spacecraft bus, the power processing unit (PPU) charges a capacitor to voltages in the 1000–2000-V range. The PPU also supplies a high-voltage pulse to a spark plug, which is used to ignite the discharge. Once the discharge is ignited, the energy stored in the capacitor (~20–50 J) powers a high-current/low-duration plasma discharge (~20 kA, ~5–10 μs), which ablates a small amount of Teflon from the face of the propellant bar and electromagnetically accelerates it to high exhaust velocities. The pulsed operation of the PPT allows it to operate over an extremely wide range of power levels at the

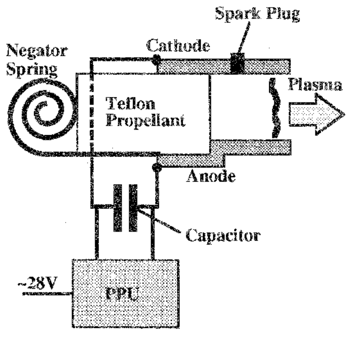

Fig. 1 PPT.

same performance level. Average spacecraft bus power supplied to the PPT is dictated by the pulse rate (typically on the order of 1 Hz maximum).

Although the described flight-qualified PPTs have performed flawlessly on several satellites,[12,13] unfortunately for small satellite designers, these models are no longer available for purchase. Furthermore, the performance of previous flight-qualified models are not well suited for the more ambitious life extension missions discussed in this paper, especially for >100-kg satellites. The absence of an off-the-shelf, flight-qualified PPT has recently spurred PPT research and development efforts with goals to significantly increase performance and decrease system wet mass, while maintaining as much as possible the flight heritage of previous designs.[3,9–11,14] The PPT to be demonstrated on MightySat Flight II.1 will provide a dramatic leap in capability compared to previous flight-qualified models.[3,5,9] The system wet mass goal is ≤5 kg, with a total impulse and power handling capability of >15,000 N-s and ≤125 W, respectively.[5]

Orbital Analysis Model and Parameter Space Definition

Model

The orbital analysis model used is simple to implement and is quite accurate when modeling low-thrust orbit raising missions and orbital decay.[6] It is derived in detail in Ref. 6; only a summary of the assumptions and final equations will be presented in this paper. The primary assumption in the model is that the Earth's gravitational force, the PPT thrust, and thermospheric drag are the dominant orbit-averaged forces on the spacecraft and that the net effects of all other perturbation forces, e.g., radiation pressure, Earth oblateness effects, lift forces, and moon and sun gravitational forces, are negligible when averaged over an orbit. The initial orbit is also assumed to be near circular ($e \ll 1$) and remain near circular during the low-thrust orbit raising maneuver and during orbital decay. In addition, it is assumed that the thrust and drag on the satellite are much smaller than the local gravitational force (the low-thrust assumption), such that higher-order terms in the equations of motion, with magnitudes on the order of $(T - F_D)/mg_l$, can be neglected. This assumption is easily satisfied for this study; for instance, using the MightySat mission as an example, $T - F_D \sim 2$ mN, $m \sim 125$ kg, and $g_l = 8.7$ m/s^2 for a 400 km orbit gives $(T - F_D)/mg_l \sim 10^{-6}$.

With these general assumptions, a model can be derived for the spacecraft orbit radius r_n and time t_n after the nth orbit[6]:

$$r_n = r_0 + \frac{1}{\pi}\sum_{j=0}^{n-1} t_{pj}^2 \left\{ a - \frac{\rho(r_j)}{2\beta}\left[\sqrt{\frac{\mu}{r_j}} - \frac{2\pi R_e}{t_e}\cos i\right]^2 \right\} \tag{1}$$

$$t_n = \sum_{j=0}^{n-1} t_{pj} \tag{2}$$

where

$$t_{pj} = 2\pi\sqrt{r_j^3/\mu} \tag{3}$$

Generally, this numerical scheme provides sufficient spatial and temporal resolution because a typical orbit raising maneuver involves on the order of 100–1000 orbits and because r_n does not change much from orbit to orbit. Equation (1), which is discussed in more detail subsequently, is an expression for the orbit radius after the nth orbit, starting from an initial orbit r_0 and accounting for the change in radius due to PPT thrust and drag. Equation (2) is an expression for the time expired during the mission after the nth orbit, obtained by summing the orbital period for all previous orbits. Note that, although t_{pj} changes slightly during one orbit, the effect is second order with a correction term on the order of $(T - F_D)/mg_l \ll 1$.

The parameter a in Eq. (1) is equal to the orbit-averaged PPT thrust divided by the spacecraft mass. A common expression for a is

$$a = \frac{f_s T_s + (1 - f_s)T_d}{m} \tag{4}$$

Using the parameter a, which is assumed constant for reasons to be discussed, allows for the results presented to be applicable to a wide range of spacecraft mass, flight operation schemes, and PPT performance and life characteristics. The explicit dependence of the average thrust on power input to the PPT, where η is the efficiency of the PPT system, accounting for inefficiencies in the power processing unit and in the thruster's ability to transform energy stored in the capacitor into useful thrust, is

$$T = 2\eta P/I_{sp}g \tag{5}$$

Typical performance numbers for a next-generation PPT are $\eta = 0.1$ and $I_{sp} = 1000$ s (Ref. 9).

The average-thrust distinction is required because the PPT is a pulsed device, where the average thrust is equal to the product of the impulse bit per pulse and the pulse frequency. Average thrust has meaning and is useful when the timescale of the life extension maneuver is much longer than the inverse pulse frequency. This criterion is easily satisfied in this analysis where transfer times are ∼10–100 days and the inverse pulse frequency is ∼100 ms.

The parameter f_s generally varies throughout a thrusting maneuver and is a function of the orbit inclination, altitude, right ascension of the ascending node, and time of year. Because of the desire to limit the parameter space involved and because the variation of f_s can be well represented by an average value, f_s is assumed constant.

An expression for the orbit-averaged satellite ballistic coefficient is

$$\beta = \frac{m}{t_p}\left[\int_0^{t_p} C_d(t)A(t)\,\mathrm{d}t\right]^{-1} = \frac{m}{\overline{\overline{C_d A}}} \tag{6}$$

This parameter is used to characterize the susceptibility of the satellite to thermospheric drag. Because the PPT generally does not use much propellant over an entire mission (mass fractions are typically >98%), it is assumed that the spacecraft mass and, thus, a and β are constant throughout the mission. This assumption eliminates I_{sp} from the parameter space associated with this study. The specific impulse is used to determine the propellant mass and average power required for the mission, both of which are used to compare life extension strategies:

$$\frac{m_{prop}}{m} = \frac{at_n}{I_{sp}g} \ll 1 \tag{7}$$

$$P/m = a(I_{sp}g/2\eta) \tag{8}$$

The thermospheric density ρ was determined by using the MSIS-86 model.[15] When globally averaged and annual averaged, the thermospheric model provides the density as a function of solar activity (as characterized by $F_{10.7}$). $F_{10.7}$ is the 81-day centered average of the solar radiative flux at the Earth's surface at 10.7 cm, which is commonly used as a proxy for solar activity in the ultraviolet spectrum. All other terms (diurnal, semidiurnal, longitudinal, etc.) are small when averaged over a single orbit, except for those accounting for geomagnetic activity. No effort was made to account for geomagnetic storms ($Ap = 4$) on PPT life extension performance, although their effects will be discussed later. The MSIS-86 model is state of the art and has compared very well with in-flight measurements (∼10%) (Refs. 16 and 17).

The last two terms in Eq. (1) are used to model the relative velocity of the thermosphere and the spacecraft. The first term, $(\mu/r_j)^{1/2}$, is the geocentric velocity of the spacecraft. The second term, $(2\pi R_e/t_e)\cos i$, accounts for the fact that the thermosphere rotates with the Earth, thus reducing the drag force for direct orbits.[17] For this study, it is assumed that the Shuttle was placed in a direct orbit with an inclination of 44.4 deg (which corresponds to the average value of cos i when i is between 28 and 57 deg). This assumption is very good considering that the Earth's rotational velocity has a small effect on the drag force and allows for the elimination of i from the parameter space investigated.

Parameter Space Definition

Using the given model and assumptions, there are essentially four parameters that are required to define the parameter space for this study: initial altitude, where the Shuttle first deploys the satellite; β, the orbit-averaged satellite ballistic coefficient; a, the orbit-averaged spacecraft acceleration due to PPT thrust alone; and $F_{10.7}$, the 81-day centered average of the solar radiative flux at the Earth's surface at 10.7 cm.

To further limit the trade space involved, typical ranges of these four parameters are first reviewed. First, an initial altitude range of 140–240 n mile was considered typical of the Space Shuttle.[18] Although the Shuttle can reach altitudes lower and higher than this range, it is shown later that 140–240 n mile bounds the region of practical interest in using the PPT for satellite life extension. From a survey of small-satellite designs, a range of 10–50 kg/m^2 is assumed to span values of β for small and microsatellites to be deployed from the HES. For the MightySat spacecraft, $m \sim 125$ kg, $A \sim 1.5$ m^2, and $C_d \sim 2$ results in a $\beta \sim 40$ kg/m^2.

Two extreme values of $F_{10.7}$ were assumed: one to represent solar minimum plus 2σ levels ($F_{10.7} = 80 \times 10^{-22}$ Wm^{-2}Hz^{-1}) and the other to represent solar maximum conditions plus 2σ levels ($F_{10.7} = 240 \times 10^{-22}$ Wm^{-2}Hz^{-1}) (Ref. 19). To compensate for the highly random nature of the thermosphere, 2σ levels were used because these are the minimum levels to which a typical spacecraft/PPT system would be designed.

Using Eqs. (4) and (5), next-generation PPT performance values ($\eta = 0.1$ and $I_{sp} = 1000$ s), and typical small-satellite payload (not total) power to mass ratios (0.3–1.0 W/kg), the acceleration was taken to range from $a = 0.7$ to 2.1 μg. Note that this range of a corresponds to the case where all payload power is devoted to the PPT for raising the satellite's orbit. If a satellite is designed for simultaneous operation of the payload and PPT, considerably less power will be available for PPT use. For this case, the power (to the PPT) to mass ratio is probably no greater than $\frac{1}{10}$th of typical small-satellite payload power to mass ratios, which correspond to an acceleration in the 0.07–0.21-μg range. The use of typical small-satellite payload power to mass ratios, which again were determined from surveying small-satellite designs, is extremely useful for assessing the practicality of various PPT life extension strategies.

Before reviewing the various life extension strategies, it is useful to examine the orbital life of typical small satellites deployed from the Shuttle without propulsion ($a = 0$). Shown in Fig. 2 is a plot of satellite life for the described parameters and with end of life defined as when the spacecraft falls below 130 n mile. Depending on solar conditions and the satellite ballistic coefficient, orbital decay times range from as little as a few days to over three years. For deployment above 215 n mile, with $\beta > 30$ kg/m^2 at solar minimum, a propulsion system is not required to extend satellite life to one to two years, although the PPT is well suited to extend life beyond two years.

Life Extension Strategies

Hold

The first life extension strategy to be investigated is the use of PPT, at the Shuttle-deployed altitude, to provide an orbit-averaged force that exactly compensates the drag force. This life extension strategy is designated hold, as inspired by the use of this term in Ref. 20. This PPT operational scheme is not new and is exactly that used on the NOVA satellites to precisely compensate for atmospheric and solar drag.[12,13] However, the NOVA satellites were operated at an altitude of 634 n mile, where on-orbit life is not an issue.

From Eq. (1), the value of a required to maintain the spacecraft orbit is given by the following expression:

$$a^* = \frac{\rho(r_0)}{2\beta}\left[\sqrt{\frac{\mu}{r_0}} - \frac{2\pi R_e}{t_e}\cos i\right]^2 \tag{9}$$

where a^* corresponds to the value of a required to compensate exactly for the drag force at the Shuttle-deployed altitude. The parameter a^* is very significant and will often dictate whether a particular strategy is feasible. The propellant mass and power required for hold are determined from Eqs. (7) and (8).

Shown in Fig. 3 is the 2σ worse-case drag force as a function of altitude, the orbit-averaged ballistic coefficient, and the $F_{10.7}$ index. The right-hand scale of Fig. 3 shows the orbit-averaged power divided by the total spacecraft mass (the specific power) required for a PPT with next-generation performance ($\eta = 10\%$ and $I_{sp} = 1000$ s) to perform the hold mission. For example, at 1 W/kg, a 50-kg satellite requires 50 W of orbit-averaged power to maintain the satellite's altitude.

The upper shaded region in Fig. 3 corresponds to the range of specific power (payload power divided by spacecraft mass) for typical small satellites. The hold mission, by definition, requires the PPT and payload to operate concurrently on-average during each orbit. The lower shaded region corresponds to the assumption that 10% of the orbit-averaged payload power (at best) is available to the PPT for the hold mission. Based on this assumption, Fig. 3 shows that the hold strategy requires too much power to be practical for most satellite designs and will not work at solar maximum. Even at solar minimum, the hold strategy is only practical at the highest Shuttle orbits, where orbital life is already quite long (cf. Fig. 2). As will be discussed later, the propellant mass requirements are also very high for the hold strategy.

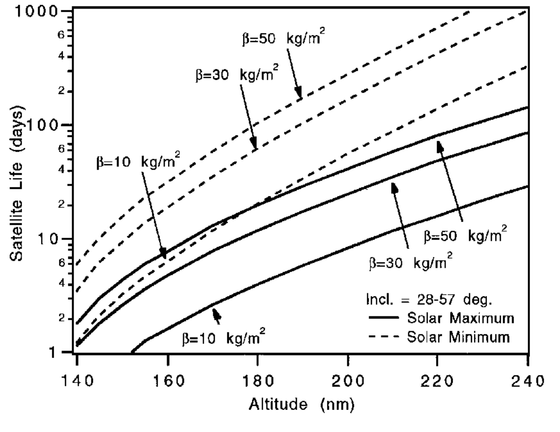

Fig. 2 Characteristic small-satellite on-orbit life (2σ worse case) when deployed from the Space Shuttle.

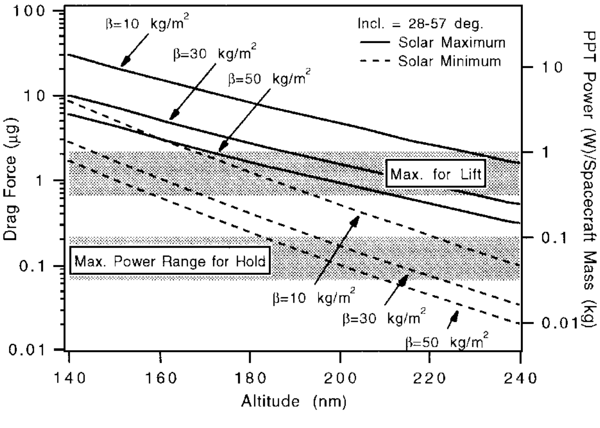

Fig. 3 Drag force and PPT power required to maintain the orbit at typical Shuttle altitudes (2σ worse case).

Lift and Coast

After the Shuttle deploys the satellite, an alternative life extension option is to use all of the power available to the payload to boost the satellite to a higher altitude. This strategy is designated lift and coast, where lift is again an appellation used in Ref. 20. The obvious tradeoff between this strategy and hold is the constraint of operating the payload on standby power during the transfer, which may have a duration of a few months.

With the payload operating in standby mode, the assumed upper limit of 0.3–1 W/kg is available for propulsion system use (see the upper shaded region in Fig. 3). Note that the PPT is of limited use during solar maximum conditions at altitudes below 170 n mile due to inadequate thrust to overcome drag. At solar minimum, the PPT is also of limited use at altitudes below 140 n mile and is not necessarily needed for altitudes above 210 n mile with $\beta > 30$ kg/m^2 (where orbital decay times without propulsion are in the one- to two-year range).

Within these PPT applicability limits, a large region of parameter space is accessible for using the PPT to extend satellite life using the lift and coast strategy. Figures 4–8 illustrate the benefits of the lift and coast strategy and span most of the parameter space involved. Plotted is the satellite life after the PPT orbit raising maneuver vs the PPT thrusting duration. To assess total on-orbit satellite life, the PPT thrusting time must be added to the postfiring life of the satellite. Each plot spans a range of a and initial altitude for a given level of solar activity and β. As the thrusting duration is reduced to zero, the satellite life is reduced to its natural orbital decay life at the given initial altitude and β.

As expected, for a given transfer time, increasing the value of a significantly increases satellite life; conversely, for a fixed satellite life requirement, the transfer time is significantly reduced by increasing a. Enhancing PPT efficiency, maximizing its operating time during each orbit, and boosting the power input to the PPT all act to increase a. In fact, increasing the solar array area to provide

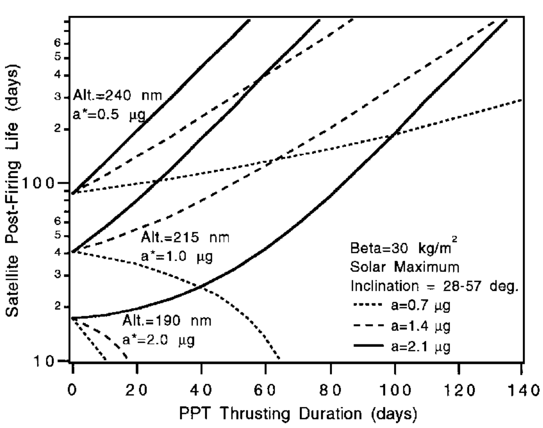

Fig. 4 Satellite life extension using the lift and coast strategy (solar maximum, β = 30 kg/m^2).

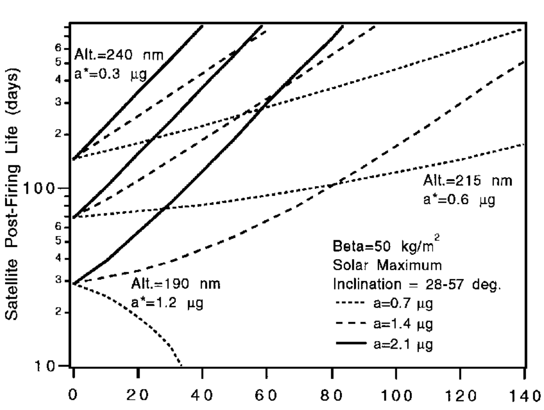

Fig. 5 Satellite life extension using the lift and coast strategy (solar maximum, β = 50 kg/m^2).

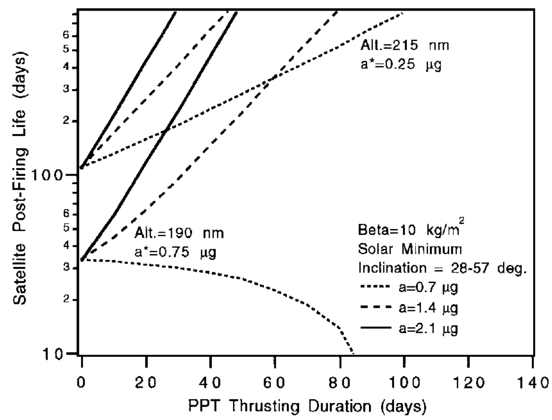

Fig. 6 Satellite life extension using the lift and coast strategy (solar minimum, β =10 kg/m^2).

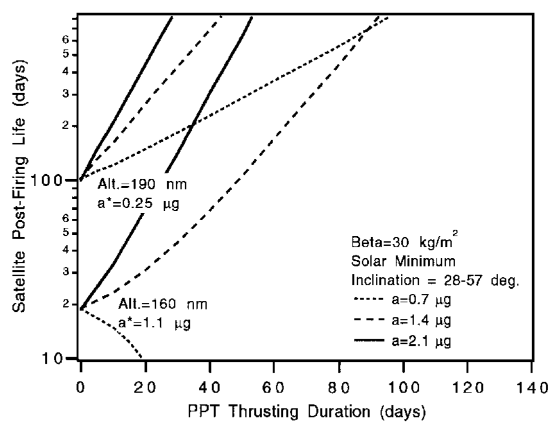

Fig. 7 Satellite life extension using the lift and coast strategy (solar minimum, β = 30 kg/m^2).

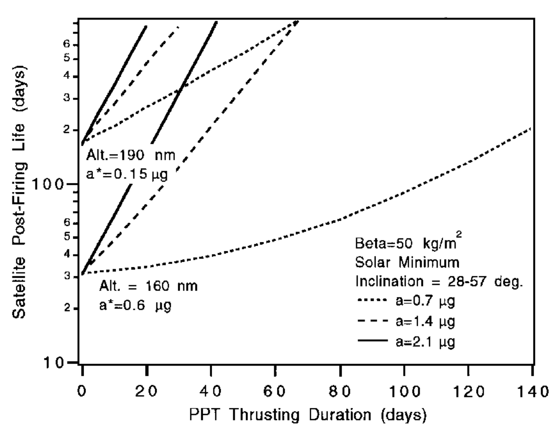

Fig. 8 Satellite life extension using the lift and coast strategy (solar minimum, β = 50 kg/m^2).

more PPT power compensates quite well for the drag increase and the need to raise the satellite to a higher altitude to achieve the same life (compare the $\beta = 30$ kg/m^2, $a = 0.7\,\mu g$ case to the $\beta = 10$ kg/m^2, $a = 2.1\,\mu g$ case at 190 n mile in Figs. 6 and 7). Such an approach is not worth the additional cost of the solar arrays when used for the hold mission because the additional thrust is canceled by drag from the larger array.

When PPT thrust is much greater than the initial drag force ($a \gg a^*$, where values of a^* are also shown in Figs. 4–8), transfer times of 10–30 days are typically required to achieve a 1-year life. When the thrust is only slightly greater than the initial drag force ($a \sim a^*$), transfer times are typically much longer (>100 days). When the thrust is less than the initial drag force ($a < a^*$), satellite life after PPT operation actually reduces with thrusting duration. However, because the PPT is reducing the rate of orbital decay, the total time in orbit (PPT thrusting time plus nonthrusting time) is increased.

Table 1 Comparison of lift and coast and hold

	$a \sim a^{*a}$		$a \gg a^{*b}$	
Life extension strategy	Lift and coast	Hold	Lift and coast	Hold
Transfer time, day	128.5	0	38.0	0
Propellant mass fraction	0.016	0.034	0.007	0.015
Hold power/satellite mass, W/kg	0	0.57	0	0.14

[a]Assumptions: initial altitude = 190 n mile, solar maximum, $\beta = 50$ kg/m^2, $a = 1.4\,\mu g$, $a^* = 1.2\,\mu g$, life = 1 year, $I_{sp} = 1000$ s, $\eta = 10\%$.
[b]Assumptions: initial altitude = 240 n mile, solar maximum, $\beta = 50$ kg/m^2, $a = 2.1\,\mu g$, $a^* = 0.3\,\mu g$, life = 2 years, $I_{sp} = 1000$ s, $\eta = 10\%$.

Although factors of two to three times life extension can be obtained in this case, it is generally not possible to extend life to one to two years when $a < a^*$.

Just as lift and coast characteristics can be separated into three cases ($a \gg a^*$, $a \sim a^*$, and $a < a^*$), so can the relevant characteristics of all other strategies considered. In the remaining portion of the paper, life extension strategies will be compared for two distinct cases: $a \gg a^*$ and $a \sim a^*$ (note that the expression $a \sim a^*$ implies $a^* < a < \sim 1.5a^*$).

Comparison of Hold and Lift and Coast

Hold and lift and coast represent the two basic strategies for extending satellite life; all other PPT thrusting schemes discussed represent compromises between these two. Hold, at its best, represents the ability to extend satellite life by operating the PPT throughout the mission at a very low-power level. Lift and coast does not require PPT operation during the payload's mission but does require the payload to operate on standby power during the orbit raising maneuver after Shuttle deployment. In this section, the relative merits of each strategy are discussed in the context of a small satellite deployed from the Shuttle.

Although the principal advantage of hold is the ability to use the PPT to extend satellite life and to operate the payload soon after Shuttle deployment, there are many disadvantages associated with this strategy when applied at Shuttle altitudes. First, the power and propellant mass requirements are generally too high to provide life extension to one to two years. Shown in Table 1 are the power and propellant mass requirements for both strategies for the two distinct cases: $a \gg a^*$ and $a \sim a^*$. For the $a \sim a^*$ case, a 50-kg satellite requires 1.7 kg of propellant and 28.5 W of orbit-averaged power to extend the life to one year. This power level is very high for this class of satellite, and considering that the PPT dry mass can be as high as eight times the propellant mass, the PPT system mass is also large. Even when $a \gg a^*$, the power required by the PPT is probably too much for nominal small-satellite designs.

A serious disadvantage of the hold strategy results because power-limited propulsion for drag makeup is inherently unstable to multi-orbit timescale fluctuations in thermospheric density. Consider the scenario where a multiday global rise in density reduces the satellite's altitude to the point where the PPT is unable to hold the satellite in orbit without more power. If, at that point, the spacecraft is unable to provide more power to the PPT, the spacecraft will quickly fall out of orbit. Potential sources of density fluctuations on this timescale include variations in solar uv flux and geomagnetic storms.[21–23] Density fluctuations from these phenomena are truly unpredictable in terms of when they occur, their duration, and the magnitude of the density increase. A design life of one to two years results in a high probability that a large multiday density fluctuation will occur during the hold mission. The average number of geomagnetic storms classified as severe ranges from zero to five per year, depending on the time with respect to the solar cycle.[22] Geomagnetic storms classified as major number about 5–20 per year.[22] Therefore, even though the results shown in Table 1 for hold are calculated assuming $a = a^*$, some margin and/or peak power requirements must be implemented into the PPT/spacecraft design before such a strategy can be implemented. Considering this issue, the values of the propellant mass fraction and hold power in Table 1 should be taken as the minimum required.

Lift and coast is much less sensitive to density fluctuations because more power is available to lift the satellite and because the satellite is at its lowest altitude for only a brief period of time at the beginning of the transfer. As shown in Table 1 and Figs. 4–8, the primary advantage of lift and coast is that this strategy is feasible for most small-satellite-specific powers and at next-generation PPT performance levels. In addition, lift and coast eliminates the need to operate the PPT concurrently with the payload, thus maximizing payload power during its operational phase of the mission.

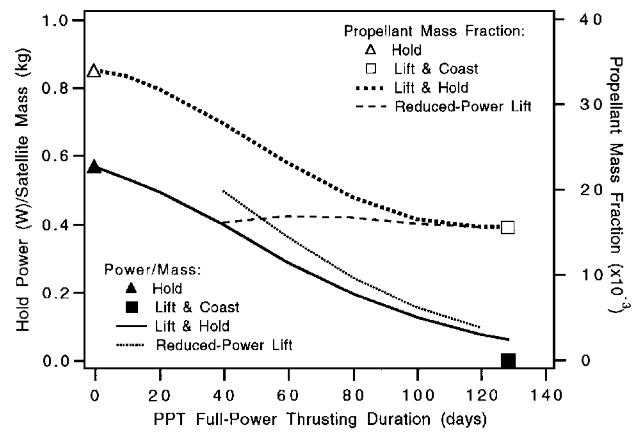

Fig. 9 Comparison of life extension strategies for a representative $a \sim a^*$ case: initial altitude = 190 n mile, solar maximum, β = 50 kg/m^2, a = 1.4 μg, a^* = 1.2 μg, I_{sp} = 1000 s, η = 10%, 1-year design life.

Lift and Hold and Related Strategies

There are various life extension strategies that can be used to provide a compromise between lift and coast and hold. The use of these strategies will typically result in lower power and propellant mass requirements than hold and a reduced transfer time compared to lift and coast. One strategy, which is designated lift and hold, is to use the full power available to the payload to boost the satellite's orbit up to a point where the PPT requires much less power and propellant to hold the satellite in orbit. Another strategy, reduced-power lift, is to use the full payload power to boost the satellite up to a point where much less power is required to perform the remaining lift mission. For this strategy, the PPT is operated at full power to reduce the time that the drag force has to act on the satellite; when the drag force is no longer excessive, the PPT is throttled down, and the payload becomes operational for the reduced-power-lift phase of the maneuver and during the subsequent coast period. The final strategy examined, lift/coast/reboost, requires the PPT to be used at full power to boost the satellite to a higher altitude; the PPT is then turned off, and the payload is allowed to operate in its nominal mode at full power until the satellite falls to the original altitude. At this point, the PPT is again used at full power to reboost the satellite to the same peak altitude. It is not difficult to conceive of many more combinations of the described life extension strategies.

Shown in Figs. 9 and 10 is a comparison of these strategies with hold and lift and coast for two representative conditions: $a \sim a^*$ (Fig. 9) and $a \gg a^*$ (Fig. 10). Plotted on the left-hand scale is the specific power required, while the payload is operational, to perform the particular life extension strategy. On the right-hand scale is the propellant mass fraction associated with providing one year (for $a \sim a^*$, Fig. 9) or two years ($a \gg a^*$, Fig. 10) of on-orbit life. The horizontal scale is the full-power (payload on standby)

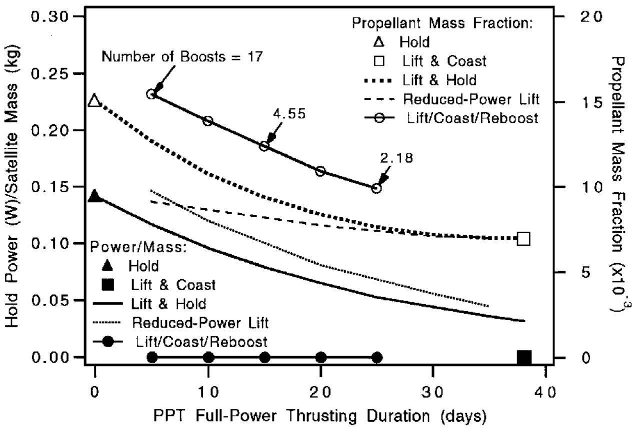

Fig. 10 Comparison of life extension strategies for a representative $a \gg a^*$ case: initial altitude = 240 n mile, solar maximum, β = 50 kg/m^2, a = 2.1 μg, a^* = 0.3 μg, I_{sp} = 1000 s, η = 10%, 2-year design life.

PPT thrusting duration in days. Note that the additional margin to account for density disturbances is not accounted for in these plots.

Starting with Fig. 9, it is seen that hold (PPT full-power thrusting duration = 0 days) has the largest power and propellant mass requirements of all strategies, and lift and coast requires the longest duration of PPT full-power operation (128.5 days). Lift and hold allows for a continuous distribution of power, mass fraction, and thrusting time between these two extremes. For instance, if the lift and coast PPT thrusting duration of 128.5 days is too long, lift and hold allows for considerable transfer time reduction, although at the cost of propellant mass and power required to hold the satellite in orbit. An 80-day transfer time increases the propellant mass fraction from 0.016 to 0.019 and hold power requirements from 0 to 0.19 W/kg. Although these values are high, they are much better than those corresponding to hold. When $a \gg a^*$, significant transfer time reductions can be achieved with relatively small increases in propellant and power (cf. Fig. 10). For example, using lift and hold to reduce the transfer time to 10 days from 38 days requires a specific power level of 0.1 W/kg (plus margin to account for density disturbances) and only a slight increase in propellant fraction.

Reduced-power lift is another strategy that may prove useful for certain missions. For Figs. 9 and 10, it was assumed that, after a specified period of full-power thrusting, the PPT power was reduced to the point where its orbit-averaged thrust was 25% greater than the drag force at the corresponding altitude. The PPT is then powered at the reduced level until it reaches an altitude where the decay time equals the desired life minus the reduced-power thrusting duration. As shown in Figs. 9 and 10, reduced-power lift requires less propellant and more power than lift and hold for all PPT full-power thrusting durations.

The performance of the lift/coast/reboost strategy was not plotted in Fig. 9 because it is not practical when $a \sim a^*$. Typically, when $a \sim a^*$, the full-power transfer time is greater than the peak to original altitude decay time. As shown in Fig. 10, even when $a \gg a^*$, short transfer times result in excessive propellant usage compared to the other strategies. By definition, there are no power requirements for this strategy when the payload is operational. Also shown in Fig. 10 is the number of boosts corresponding to each full-power thrusting duration. For instance, instead of a single 38-day boost, a 2-year life can also be obtained by performing 17 5-day boosts at about 1-month intervals.

Conclusions

At nominal Space Shuttle altitudes, thermospheric drag is the dominant force limiting satellite on-orbit life (typically <100 days). Because of its low system mass and volume, high specific impulse, and inert solid propellant, the PPT is ideally suited to extend the life of small satellites deployed from the Shuttle. The objective of this study was to identify and analyze life extension strategies for Space Shuttle-deployed small satellites using the PPT. The generalized analysis presented is applicable to a broad range of satellite and PPT performance/life characteristics and resulted in the following conclusions.

1) There are many strategies, enough to fit most operational scenarios, for significantly extending small-satellite on-orbit life. Within the limits of typical small-satellite power to mass ratios, the most capable of these strategies, designated lift and coast, requires the smallest amount of propellant and can extend life to one to two years with next-generation PPT technology. The lift and coast strategy consists of an initial orbit raising mission with the PPT utilizing all payload power. At the peak altitude of the transfer, which is determined from the satellite life requirement, the PPT is shut down for the remaining life of the satellite. The disadvantage of the lift and coast strategy is that the duration of the orbit raising maneuver can be as long as a few months.

2) The orbit raising transfer time is very sensitive to the orbit-averaged power provided to the PPT. When the transfer time is too long (usually due to power constraints), there are alternative strategies, which may prove useful for extending satellite life (lift and hold, lift/coast/reboost, reduced-power lift), while reducing (but not eliminating) the duration of the full-power lift phase.

3) The strategy designated as hold, which uses the PPT at the Shuttle-deployed altitude to provide an orbit-averaged force that exactly compensates the drag, is generally impractical for enhancing small satellite life to one to two years. The power and propellant requirements are too high to implement this strategy.

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