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The Case For Small Spacecraft: An Integrated Perspective on Electric Propulsion

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Abstract

As economies of scale are reduced in commercial and military endeavors, and with the international climate trending toward cooperation and stabilization, the aerospace industry today has taken a decided turn toward decreasing expenditures. This reduction in funding has challenged project managers and engineers to reduce the overall size of spacecraft while still accomplishing the same or similar tasks as larger spacecraft. Solutions to reconnaissance, environmental monitoring, and ground imaging are now currently being demonstrated by small satellites. Physical reductions in the size of overall spacecraft have been accompanied by reductions in subsystems and components. All manor of subsystems including propulsion, command and data handling, telemetry and electrical power must respond to the size challenge of smaller mass and volumetric requirements. This applies equally to electric propulsion. Overall power requirements for a small satellite can be considered between 100-300 Watts for LEO/GEO missions, and between 300-650 Watts for long duration missions. This paper will discuss the small satellite paradigm shift, unique approaches to electric propulsion integration into small satellite architectures, and develop a sample mission that focuses on enabling one of the "emerging markets" for small satellites using electric propulsion as the performance merit enhancer.

Trends

The reduction in aerospace funding levels have forced technologists to lower space mission costs. This realization has resulted in mass, power and volume reductions in most spacecraft subsystem elements. Current nomenclature designates three separate categories of satellites: large, small and micro; although the specifications on each of these catagories fluctuates depending upon the mission and the perspective of the definer. Figure 1 shows some of the characteristics of these satellite classes. The nanosat, or satellite on a chip, is another class of satellite which has been proposed, although this may be several years in the making.[1]

Of particular importance in today's environment is the small and microsatellite classes (termed small satellites for the remaining portion of this paper). A large body of information has been generated over the past years on the merits of small satellites for variety of missions.[2] Historically, small satellites began their development back in the late '50s[3]. The Utah State Small Satellite Conference, held now for the past 9 years, is a forum for the development and exchange of information specifically in this field. Many previously known & unknown international programs have been launching small satellites for a variety of LEO missions for the past 10 years[4,5,6,7], and universities have used small satellites as a means to demonstrate student-run initiatives in spacecraft education.[8,9,10] Small satellites have a number of advantages that are driving both large and small companies to invest into their development. Developing markets for small satellites include remote sensing, ground-space-ground communications, store-forward communications, synthetic aperture radar imaging, and more. Multiple spacecraft constellations have risen in importance over the past several years, as firms vying for global communication market dominance gain in momentum. Iridium, Odyssey, Globalstar, and Orbcomm are all examples of multi-satellite constellations that have identified a specific market niche in LEO space-based operation for ground-based applications[11].

The scientific community has engendered the support of small satellites to accomodate both much stiffer fiscal restrictions and shorter development times. A successful example of the small satellite philosophy was demonstrated by the Clementine mission successfully demonstrated by BMDO.[12] NASA's New Millennium and Discovery series are both examples of a trend toward smaller and faster missions for scientific benefit[13]. The Air Force Phillips Lab has a number of active interests and

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driving developments in spacecraft technology and their application. MSTI, Clementine, MightySat, and ISTD are all initiatives to develop techniques and obtain data in specific arenas relating to imaging, data fusion, communications and operations.

All of the above activities suggest that there is a growing market for small satellite propulsion. This year several flight experiments and commercial systems will deploy with electric propulsion (EP) devices.[14] A trend has taken place over the last few years that has seen a much wider audience to electric propulsion developments within the commercial arena rather than the military and government arenas. This significance of status that electric propulsion has now reached should be maintained with the growing small satellite market arena.

In some ways, small satellites are at a stage of relevance to the space industry as EP was several years ago. Mission of merit traditionally set aside for large satellites can be approached through multiple small satellite constellations. This presents an interesting and immediate opportunity for the EP community to emphasize system-level integration of the electric propulsion subsystem elements into the spacecraft architecture to increase the acceptance to levels of full reliance.

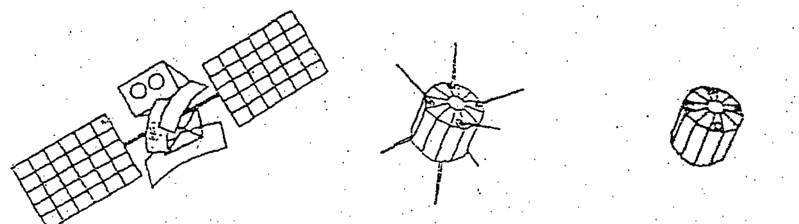

Figure 1: Characteristics of the small satellite progression.

Mission Study

Although the use of electric propulsion for small satellite missions has been discussed in a few papers[15,16,17,18], very little analysis has been published concerning the use of EP for LEO small satellites applied to specified missions. The objective in this analysis will be to demonstrate performance merit of electric propulsion applied to one such mission: LEO remote sensing satellites. An apparent application of electric propulsion is to significantly increase ground resolution by reducing the altitude of a remote sensing satellite (of specified optical train) and by using high-specific impulse EP for drag compensation.[19]

Remote sensing has rapidly become a commercially-viable endeavor with obvious strategic significance. In the commercial sector, increased resolution is a trend that has been on-going, and is expected to continue for many years to come. Table I shows the characteristics of three LEO remote-

sensing satellites currently in orbit. These satellites are typically in sun-synchronous orbits and have a design life in the 2-5 year range (with state-of-the-art satellites trending toward the 5-year range).

Satellite	BOL Power (W)	BOL Mass(kg)	Typical Wavelength (μm)	Ground Resol. (m)	Alt. (km)	Specific Power (W/kg)	α_{pl} (kg/m^3)	β_{pl} (W/m^3)
LANDSAT4	990	1941	0.6	30	705	~0.51	~7×10^8	~4×10^8
SPOT3	1382	~1900	0.6	10	820	~0.73	~2×10^7	~1×10^7
IRS-1A	~700	~975	0.6	72.5	900	~0.72	~2×10^9	~2×10^9

Table 1: Examples of LEO remote sensing (visible & IR) satellites currently in orbit[20]

Within the next five years, even more capable commercial remote sensing satellites will be placed into service, including SPOT5 (5m resolution), World View (3m), and ORBIMAGE (1m). The preoccupation with resolution is understandable considering that a factor of two increase in resolution can greatly enhance identification capabilities.

As with most missions, the enhanced capability provided by the use of EP can be realized in many ways:

1) Significant mass, power, and cost savings associated with the reduction in resolution requirements for the optical payload. The scaling parameter for optical payload mass ($\sim d^3$), power ($\sim d^3$), and cost ($\sim d^{1/2}$) is the aperture diameter (d). Furthermore, the cost associated with the payload is generally a significant fraction of the total cost of the satellite. A nice illustration of the magnitude of this cost is shown in the *FireSat* preliminary design in the text by Larson and Wertz[2]. Preliminary cost estimates showed that the IR payload accounted for about 90% of the total hardware costs associated with this remote sensing satellite concept[2].
2) Significant increase in payload ground resolution and satellite life.
3) Significant increase in payload capability on features other than resolution, such as the number of spectral bands, dynamic range, stereo imaging, swath width, number of sensors, increased memory and data transmission capabilities, etc.
4) Significant reduction in launch costs by downsizing launch vehicles for a given resolution requirement.

Of course these benefits need to be traded with the added complexity, mass, and cost associated with using electric propulsion: additional solar arrays, batteries, dry mass, contamination issues, and as yet unidentified impacts on the spacecraft. Note also that such penalties are usually compensated somewhat by the fact that the launcher can throw more mass into lower orbits. We plan to focus on these benefits by first examining the most tractable of them: increased resolution, in particular for satellites launched from a Pegasus-class launch vehicle.

Analysis

The following is a first-order mission analysis to quantify the benefits of using electric propulsion to increase the resolution of a Pegasus-class remote-sensing (visible and IR) satellite. To this end, we assumed that the satellite will be designed from the ground-up, fully capable to produce marketable images and fully implementing EP to maximize ground resolution, and with no regard to cost. In this section, the equations used in this analysis are reviewed.

First, we write expressions for the mass (M_{sc}) and power (P_{sc}) associated with the spacecraft:

$$M_{sc} = M_{pl} + M_{sa} + M_{bat} + M_{dry} + M_p = M_{pegasus}(h) \tag{1}$$

$$P_{sc} = P_{pl} + P_{ep} \tag{2}$$

where M_{pl} and P_{pl} are the mass and power associated with the optical payload and all supporting components of the spacecraft such as ADCS, thermal, structure, C&DH, communications, and the power subsystem (other than the solar arrays for the payload and EP system, and batteries for the EP system). In other words, M_{pl} and P_{pl} account for the mass and power associated with all subsystems except the solar arrays and the electric propulsion system. M_{sa} is the mass of the solar array, M_{bat} is additional battery mass required for the EP system, M_{dry} is the dry mass of the propulsion system, M_p is the propellant mass, and P_{ep} is the power associated with the EP system. Equation 1 also shows that the total spacecraft mass is equal to the sun-synchronous launching capability of the Pegasus XL, which is obtained from the Pegasus Payload User's Guide[21].

Remote sensing satellites, such as LANDSAT, typically have many instruments, with multiple capabilities optimized for various missions. A mission analysis which includes the details of the payload design is obviously beyond the scope of this study. To make the analysis tractable, it is assumed that the payload mass and power are proportional to the cube of the aperture diameter. Such an expression is good for first-cut estimates of optical payload characteristics[2]; furthermore, we have also assumed that the mass of the supporting subsystems are proportional to the optical payload mass. The aperture diameter can be related to the ground resolution of the imaging system using the standard expression for diffraction-limited ground resolution at nadir[2]:

$$\frac{R}{\lambda} \approx \frac{h}{d} \tag{3}$$

where R is the ground resolution at a given wavelength, λ, h is the altitude, and d is the aperture diameter of the imaging payload. With equation 3, and the above assumptions, the following expressions are used to relate M_{pl} and P_{pl} to R:

$$M_{pl} \sim d^3 \sim \left(\frac{h}{R/\lambda}\right)^3 \rightarrow M_{pl} = \alpha_{pl}\left(\frac{h}{R/\lambda}\right)^3 \tag{4}$$

$$P_{pl} \sim d^3 \sim \left(\frac{h}{R/\lambda}\right)^3 \rightarrow P_{pl} = \beta_{pl}\left(\frac{h}{R/\lambda}\right)^3 \tag{5}$$

Thus α_{pl} and β_{pl} reflect the capabilities of the optical payload and supporting hardware; typical values are estimated in Table 1. The functional form of equations 4 and 5 are important assumptions of the model, and are essential for accounting for unknown details associated with the payload. Also note that the values tabulated in Table 1 vary by as much as two orders of magnitude among the different satellites. As will be shown later, the results of this study are not dependent on α_{pl} and β_{pl}, but on their ratio: β_{pl}/α_{pl}. This ratio is equal to the specific mass of a standard remote-sensing satellite (without EP), which is typically on the order of one.

The solar array mass, M_{sa}, and area, A, are determined from the following standard relations and by assuming no degradation (this assumption can easily be relaxed).

$$M_{sa} = \frac{P_{sc}}{\alpha_{sa}} \tag{6}$$

$$A = \frac{P_{sc}}{\beta_{sa}} + A_0 \tag{7}$$

where α_{sa} is the specific power and β_{sa} is the specific area of the solar array, and A_0 is the average frontal area of the spacecraft, assumed to be 1 m^2. For this study, α_{sa}=40 W/kg and β_{sa}=140 W/m^2, which are typical BOL performance figures for a conventional planer silicon array[2]. The power required and mass of the electric propulsion system can be obtained from the following expressions:

$$M_p = \frac{Tf_t}{gI_{sp}}\tau_L \tag{8}$$

$$M_{dry} = \alpha_0 + \alpha_p M_p \tag{9}$$

$$P_{ep} = \frac{Tf_t gI_{sp}}{2\eta_{ppu}\eta_t} \tag{10}$$

where g is the gravitational constant equal to 9.81 m/sec^2. I_{sp} and T are the specific impulse and thrust of the propulsion device, α_0 and α_p are constants. The efficiency of the thruster and power processing unit (PPU) are represented by: η_t and η_{ppu}. τ_L is the design life of the satellite, and f_t is the fraction of the orbit that the thruster is operating, ranging from 1 at the minimum altitude (continuous thrusting) to very small values at large altitudes. f_t' is used to account for the use of batteries to store the energy required to operate the thruster (more will be discussed on this topic later).

The battery mass is calculated using the following expression[2]:

$$M_{bat} = \frac{TgI_{sp}t_p}{2\,\eta_{ppu}\eta_t\eta_{bat}n\alpha_{bat}DOD} \tag{11}$$

where η_{bat} is the transmission efficiency between the battery and PPU (assumed equal to 0.9), n is the what we have termed the battery loading factor (more to be discussed on this later), α_{bat} is the specific power of the battery (assumed to be conventional NiCd, 30 W-hr/kg), t_p is the orbital period (t_p=$2\pi\mu^{1/2}[R_e+h]^{1/2}$, but assumed to be equal to 90 min for this study), and DOD is the depth of discharge of the battery:

DOD = 1.457 - 0.122 ln(# of charging cycles) $\quad 500 \leq cycles \leq 300{,}000$

This expression was obtained from a line-fit of figure 11-11 of reference 2. For this study:

$$DOD = 1.457 - 0.122\ln\left(\frac{nf_t\tau_L}{t_p}\right) \tag{12}$$

When the battery loading factor, n, is equal to one, it corresponds to the case where the thruster is fired for 1.5 hours every 1.5/f_t hours. There is benefit to operating the thruster more often for shorter periods of time (to save battery mass): so n=2 corresponds to the case when the thruster is fired 45 minutes every 0.75/f_t hours. 45 minutes was assumed to be the minimum firing time to neglect the effects of start-up transients on thruster performance (for the SPT, ion engine, and arcjet). Battery mass can be neglected ($n \to \infty$) for thrusters which can be pulsed or operated over much shorter time periods, such as the resistojet and pulsed plasma thruster.

For this study, f_t' can be many values depending on the situation. When f_t' is set equal to 1, additional solar arrays are added to allow for thruster operation at full power at any time the solar arrays can produce power. This condition represents the worst-case scenario concerning the use of EP on the LEO satellite; in many cases the drag on the satellite can be significantly reduced by the use of batteries to store energy from smaller solar arrays. Note that for this case n is set equal to infinity to account for the fact that batteries are not used for this situation (M_{bat}=0). Setting f_t'=0, and $n=\infty$ represents the case where conventional chemical thrusters are used (such as low-thrust hydrazine thrusters). When batteries are used:

$$f_t' = \frac{f_t}{\eta_{bat}(1-f_e)} \tag{13}$$

where f_e is the fraction of the orbit in eclipse. Typically f_e is near its maximum (at a given altitude) for remote sensing satellites. We assumed f_e to be constant and equal to 0.37, which is a good average estimate of the maximum eclipse fraction for altitudes below 1000 km. Note also that the use of existing batteries and/or by allowing the thruster to fire while the payload is off (such as over oceans) suggests that f_t' could actually be smaller.

The design life of the satellite is determined from the following relation:

$$\tau_L = \frac{N\tau}{f_t} \tag{14}$$

where N is the number of thrusters used serially for drag makeup ($N=1$ for this study), and τ^* is their operating life (which is limited by their design life). The design life of the satellite is considered the time for expellation of all propellant. At that point, the spacecraft's orbit will begin to decay, which will alter the timing associated with the ground track, and cause other assorted problems. Note that for conventional remote-sensing satellites, it is not the loss of stationkeeping propellant which is the life-limiter, but typically the life of components in the payload or spacecraft bus.

The final expression to consider is that for the drag on the satellite. Assuming that the satellite maintains a circular orbit, the expression for the drag force F_D on a satellite is:

$$F_D = \frac{1}{2}\rho(h)\frac{\mu}{h+R_e}C_d A = Tf_t \qquad (15)$$

where $\rho(h)$ is the mass density at altitude h, μ and R_e are the gravitational parameter and radius of the earth respectively, and A is determined from equation 7. C_d is the drag coefficient, which is generally on the order of one[2], and we will assume to be 2.2. Although it neglects lateral drag effects, the use of equation 7 to determine A is a conservative assumption, considering that the orientations of the solar arrays and satellite with respect to the velocity vector are continually changing throughout the orbit, and equation 7 represents the maximum area.

The 1976 U.S. standard atmosphere was used to determine the mass density versus altitude, although the mission-averaged density can change considerably depending when the launch date is with respect to the 11-year solar cycle and on the satellite design life. Such a model represents the average mass density profile, and actually the results of this study are not sensitive to factor of ten increases in density because of the exponential nature of the profile (the satellite design orbit can be increased slightly, with a corresponding small drop in ground resolution).

The above equations can easily be combined into two equations for two unknowns: h and R/λ. With some additional algebra, the following two equations can be derived:

for h:

$$1 = \xi_0(\beta_{pl}\xi_1 + \alpha_{sa}\xi_2 + \beta_{sa}A_0) \qquad (16)$$

where:

$$\xi_0 = \frac{\rho(h)\mu C_d}{2Tf_t\beta_{sa}(h+R_e)} \qquad (17)$$

$$\xi_1 = \left(M_{sc} - \xi_2 - \xi_3 - \alpha_0 - (\alpha_p + 1)\frac{N\tau^* T}{gI_{sp}}\right)\Bigg/\left(\alpha_{pl} + \frac{\beta_{pl}}{\alpha_{sa}}\right) \qquad (18)$$

$$\xi_2 = \frac{Tf_t gI_{sp}}{2\alpha_{sa}\eta_{ppu}\eta_t} \qquad (19) \qquad \xi_3 = \frac{TgI_{sp}t_p}{2\alpha_{bat}\eta_{ppu}\eta_t\eta_{bat}nDOD} \qquad (20)$$

where equation 16 is solved by iteration for h, and for R:

$$\frac{R}{\lambda} = \frac{h}{\xi_1^{1/3}} \qquad (21)$$

A more elegant way to showing the benefits of EP for remote-sensing satellites is to non-dimensionalize equation 21 with the resolution of a remote-sensing satellite which does not incorporate electric propulsion. From equation 4:

$$\left(\frac{R}{\lambda}\right)_0 = \left(\frac{\alpha_{pl}}{M_{sc}(h_0)}\right)^{1/3} h_0 \tag{22}$$

For instance, assuming h_0=800 km and α_{pl}=1x10^8 kg/m^3: M_{sc}(800 km)=206 kg[19], and $(R/\lambda)_0$=63 m/μm. These assumptions suggest that a Pegasus-class remote sensing satellite, launched to a long-life orbit of 800 km, has the capability to obtain a ground resolution of about 40 m at 0.6 μm. Using equation 22, we are now interested in maximizing (smaller R corresponds to higher ground resolution) the following ratio:

$$\frac{R_0}{R} = \left(\frac{\alpha_{pl}\xi_1}{M_{sc}(h_0)}\right)^{1/3} \frac{h_0}{h} \cong \left(\frac{M_{sc}(h) - M_{ep}}{M_{sc}(h_0)}\right)^{1/3} \frac{h_0}{h} \tag{23}$$

The second relation shows that there are two ways that EP is used to increase ground resolution. The first by reducing the altitude, and the second by the fact that at lower altitudes the launcher can put a more capable (and heavier) payload into orbit. Note that the ability to put a more capable payload into a lower orbit has only a secondary effect on resolution because the wet mass and additional solar array mass associated with the EP system reduces this benefit, and also by the fact that the resolution is impacted by only the cube root of this factor. Another benefit of using equation 23 to present the results of this study is that the resolution ratio is independent of the values of α_{pl} and β_{pl}, and dependent only on the ratio β_{pl}/α_{pl}. This ratio is much easier to estimate (values are shown in table 1), since it is equal to the specific mass of a remote sensing satellite that does not implement EP. For this study, β_{pl}/α_p was assumed to be equal to 0.75.

Results

The following thruster technologies were examined for the remote sensing application: the Hall thruster, the ion engine, the pulsed plasma thruster, the hydrazine arcjet, resistojet, and monopropellant engine. Shown in Table 2, are representative examples of each thruster, along with their nominal operating parameters (which were also used as inputs in the mission model). These parameters can be found in the following references[18,22,23,24,25,26,27,28]. The dry masses include everything associated with the propulsion system except the propellant: propellant system, PPU, gimbal, thruster, etc., plus 20% to account for structure, harness, plumbing, and margin. Note that α_0 is also calculated for each propulsion system implemented with a redundant thruster.

	SPT-50	SPT-70	XIPS 13-cm	Pulsed Plasma Thruster	MR-111 0.45-lbf N2H4	MR-501 N2H4 EHT	500W N2H4 Arcjet
Manufacturer	ISTI	ISTI	Hughes	Lab	OAC	OAC	Lab
Power to thruster (W)	300	650	439	150	----	350	500
Thrust, T (mN)	19	40	18	4.6	2000	180	79
Thrust Efficiency, η_t	0.37	0.46	0.52	0.15	----	0.76	0.33
PPU Efficiency, η_{ppu}	0.93	0.93	0.88	0.85	----	1.0	0.90
Specific Impulse, I_{sp} (sec)	1200	1510	2585	1000	220	300	425
Max. Thruster Life (hrs)	2000	3100	12000	1200	40	500	1200
α_p	0.12	0.12	0.12	0.0	0.09	0.09	0.09
α_0 (N=1) (kg)	13.4	16.2	22.2	5.4	3.8	4.5	6.6
α_0 (w/redund. thruster) (kg)	19.2	23.0	39.2	12.8	5.5	6.7	11.0

Table 2: Nominal performance characteristics of the thruster technologies examined in this study.

The desired output of this study is a plot (for each thruster) of the resolution ratio (equation 23) as a function of satellite design life. With the inputs from table 2, we now have all of the parameters needed for this study, except for f_l and τ^*. f_l was varied from $1-f_e$ to 0, which is equivalent to varying the satellite design life with τ^* fixed (see eq. 11). τ^* was varied from the maximum thruster life, tabulated in Table 2, to zero. Within this range, a maximum in the resolution ratio is expected because when τ^* is very small, the thruster has a small total impulse capability, and thus the satellite is placed in a relatively high altitude where the thruster operates infrequently enough to satisfy satellite design life requirements. At the other extreme, a thruster may have a total impulse capability which is much greater than needed for the mission, which allows for very low altitudes, but also corresponds to excessive propellant mass useage. The above optimization was performed for the SPT-50, SPT-70, XIPS, and 500W arcjet.

The chemical thruster represents a special case (f_l=0 and $n=\infty$), where equation 16 becomes a function of the total impulse of the thruster, $T\tau^*$. In this case, it is straightforward to show that an optimum $T\tau^*$ exists to obtain a maximum resolution ratio. At low altitudes, corresponding to high total impulse, and the propellant mass is extremely high (taking capability away from the optical payload); as the total impulse decreases, the propellant mass decreases, but the altitude grows (reducing resolution).

The PPT and resistojet (MR-501 EHT) are unique in that they can be operated for very short time periods without affecting performance. For these technologies, the battery requirements are small (n assumed to be infinity); and like the chemical thruster, the resolution ratio becomes a function of the total impulse only.

Figure 2 shows (R_0/R) versus τ_L for all thruster technologies assuming β_{pl}/α_{pl}=0.75, and N=1 (no redundant thruster). When maximizing ground resolution, the benefits of using EP compared to a hydrazine thruster appear small compared to the overall increase in ground resolution obtained by reducing the altitude of the standard remote sensing satellite. The ground resolution is relatively insensitive to thruster performance primarily due to the exponential variation of the density with altitude. For example, replacing a monoprop with a XIPS on a 5-year satellite, increases ground resolution by only 11%, in part from reducing the altitude from 331 km to 319 km. Table 3 shows a mass breakout for all of the technologies for a 5 year design life.

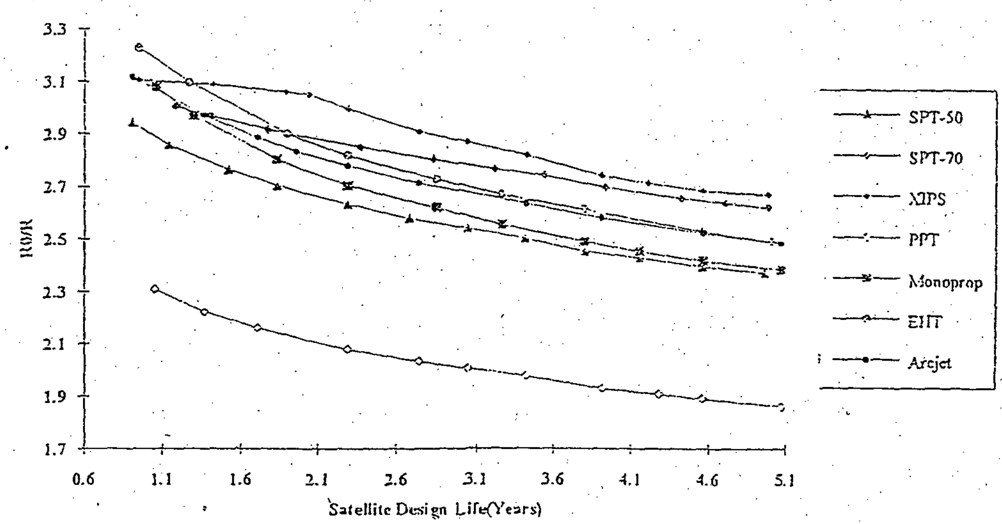

Figure 2: Resolution ratio for various thruster technologies versus satellite design life

	SPT-50	SPT-70	XIPS (w/bat)	XIPS (w/o bat)	Arcjet	Mono-prop	PPT	EHT
Spacecraft Mass(kg)	305.2	317.0	319.6	315.0	316.4	312.5	281.1	316.7
Payload Mass(kg)	255.1	217.2	203.1	241.5	188.8	195.9	268.5	188.5
Solar Array Mass(kg)	5.4	6.2	9.8	17.0	4.2	3.7	5.2	3.7
Battery Mass(kg)	18.3	43.6	50.2	0.0	27.7	0.0	0.0	~0.0
Prop. Sys. Dry Mass(kg)	14.8	19.8	25.9	25.9	14.0	12.8	5.4	14.4
Propellant Mass (kg)	11.6	30.1	30.7	30.7	81.9	100.0	2.0	110.1
Array Power for EP(W)	26.4	86.7	237.5	498.8	26.4	0.0	8.4	7.0
Array Power for P/L(W)	191.3	162.9	152.3	181.1	141.6	147.0	201.3	141.4
Altitude(km)	363	311	299	319	313	331	469	312
f_t	0.046	0.071	0.27	0.275	0.027	0.0009	0.027	0.0114
R/R_0	0.423	0.382	0.376	0.379	0.403	0.420	0.537	0.402

Table 3: Mass breakout for propulsion systems studied (5 year satellite design life)

Ultimately, the satellite designer will have to specify a minimum ground resolution, altitude, and design life. Rather than striving for maximum resolution, other benefits can be realized by using EP instead of chemical thrusters. Figure 2 shows that at a fixed resolution, the satellite design life can be significantly increased by using EP. For instance, for $(R_0/R) = 2.8$ the hydrazine monoprop can provide a satellite life of about 2 years, while the use of the XIPS (+2 years), SPT-70 (+1.5 years), and resistojet (+0.5 years) can extend the design life considerably. As typical satellite design lifes increase to 5 years and beyond, EP's advantage over chemical propulsion is even greater.

	Mono-prop	EHT	SPT-70	XIPS (w/o bat)
Spacecraft Mass(kg)	312.5	312.5	312.5	312.5
Payload Mass(kg)	195.9	195.9	195.9	195.9
Solar Array Mass(kg)	3.7	3.8	4.9	16.1
Battery Mass(kg)	0.0	~0.0	37.7	0.0
Prop. Sys. Dry Mass(kg)	12,8	11.2	18.2	25.0
Propellant Mass (kg)	100.0	74.3	17.0	23.4
Array Power for EP(W)	0.0	4.7	48.9	498.8
Array Power for P/L(W)	147.0	147.0	147.0	147.0
Altitude(km)	331	331	331	331
f_t	0.0009	0.0077	0.04	0.21
R/R_0	0.42	0.42	0.42	0.42
Margin(kg)	0.0	27.3	38.8	52.1
Margin w/red. thruster(kg)	----	25.1	32.0	35.1

Table 4: Mass margin using electric propulsion versus chemical propulsion systems

Again, if we fix the resolution requirement, we can also calculate the increase in payload mass associated with the use of EP instead of chemical thrusters. Shown in Table 4, is the mass breakout for a

satellite (5 year design life) which has the same optical payload, altitude, and thus resolution of the chemical option. This table shows that the use of a XIPS results in a 27% increase in payload mass. Such an increase in mass can be used to increase payload capabilities other than resolution, and/or for additional propellant for margin to protect against anomalous solar events, and/or to provide for a redundant thruster. The EHT is particularly attractive, providing a 14% increase in payload mass with only an additional 5 W of power, and with negligible battery requirements.

New Concepts

Remote sensing is just one mission that can benefit from electric propulsion. In assessing the performance merits of EP, several integration concepts could potentially increase the viability against chemical systems. Although not analyzed for gains/losses versus the standard model on remote sensing here, these are presented as performance enhancers that could be used for this mission and others.

Direct Drive: Proposed by Hamley of the NASA-Lewis Research Center[29], the direct drive option represents a mode of operation where the thruster is operated directly off the solar array. Such a configuration allows for more efficient power useage, and significant reductions in PPU, harness, and thermal system mass. Although this option is only viable for the V-I characteristics of the SPT, it does warrant further examination for small satellite missions.

New Propellants: The use of EP on small satellites reopens the use of new propellants to provide unique mission benefits. For the arcjet, the use of ammonia not only provides increased performance, but much better ground handling characteristics. Such benefits were paramount in the decision to use an ammonia arcject on the AMSAT-P3-D satellite[30]. Other propellants, such as C60 for the ion engine, and possibly methane for the arcjet require more investigation.

Integrated Propulsion Structure: This option would integrate the propulsion tankage and tubing to the structure of the spacecraft. Precedent has been set in small vehicle designs for KKVs where the high pressure toroidal tanks were used as circular stabilizers to the structural stiffness of the entire vehicle. For EP, tubing can be machined integral to the sidewalls of the structure, avoiding external appendages and lowering volumetric requirements. Again this approach would lower the overall mass booked directly against the EP subsystem. This option also applies directly to chemical and cold-gas propulsion systems.

Dual Mode Systems: An option that combines high thrust/low Isp, coupled with low thrust/high Isp systems would offer extended operating regimes for some missions. The PPT is one example of a propulsion system which can upgrade at power/thrust levels which are orders of magnitude apart. For higher thrust requirements another approach would be to couple into a single unit two separate systems. Clever geometry's might unite an ion engine with an arcjet. Creative layout of cathode/anode arrangement may offer coupled high/low Isp/thrust systems. Current dual use mentality uses an electric propulsion/liquid approach. Replacing this combination with an all electric system, that has a capability close to that of the liquid system, as well as the longer life provided by the EP device could prove attractive for smaller satellites.

Conclusion

Although advanced research in higher efficiency EP devices is ongoing and should be pursued, additional design and development efforts focussing on an integrated perspective on EP to spacecraft may help carry the industry into the next century. Using integrated ideas, dual use modes, and direct drive systems enhances the capability of electric propulsion to handle larger constellations of small satellites in a very cost efficient and mass efficient manner.

The concept of striving to develop a technology for a target market is far from new. The intent of this paper is to stimulate a new type of thinking for electric propulsion technologists, and to focus

efforts on a specific arena that EP can have a direct impact. The remote sensing mission was examined in this paper, but is certainly not the most compelling market opportunity. The explosion in proposed commercial small satellite constellations presents an opportunity to get in on the ground floor in the design trade process. A concerted effort now by electric propulsion technologies to address the needs and requirements of small satellites can afford a performance enhancement for both fields.

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