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EXTENSION OF SATELLITE LIFETIME VIA PRECISION POINTING OF ORBIT TRANSFER MANEUVERS

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Abstract

We describe an extremely precise, open-loop control of velocity pointing for spin-stabilized rockets and spacecraft. This technique (Velocity Precision-pointing Enhancement System) employs coupling between the spinning spacecraft dynamics and the propulsion system characteristics to virtually eliminate velocity-pointing error. By modifying an engine to have a softer ignition transient, a reduction of nearly two orders of magnitude in velocity-pointing error can be obtained. This reduction of the pointing error can be directly translated into a savings of station-keeping propellant. Since less propellant is needed to correct the error, more is available to keep the spacecraft in orbit. In this paper we assess the mass savings achievable and calculate the potential extensions of satellite lifetimes.

INTRODUCTION

Imperfections in spacecraft construction cause undesired off-axis body fixed torques which perturb the angular momentum vector in inertial space and result in velocity pointing errors during thrusting maneuvers. Spacecraft are spun at high spinrates in order to attempt to minimize the pointing errors associated with high thrust maneuvers.

Current technology uses a thrust history that closely resembles a step function causing the average angular momentum vector to be shifted an angle ρ with respect to the desired direction of the ΔV as shown in Fig. 1a. This angle ρ is known as the velocity pointing error[1,2]. References [1] and [2] show that delaying the rise to

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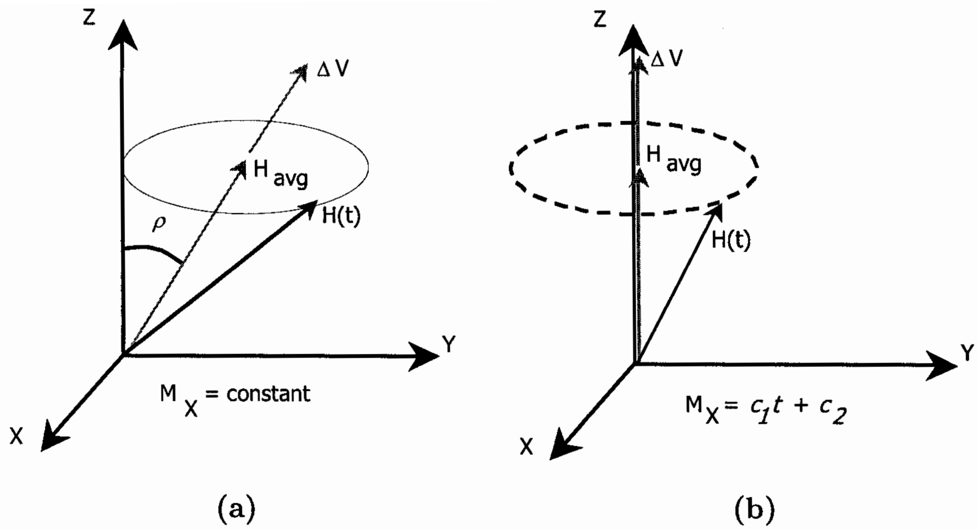

Figure 1 Motion of the angular momentum vector in inertial space for a spacecraft (a) without VPES and (b) with VPES.

maximum thrust – the ignition transient – so that the engine run up takes at least one revolution about the spin axis, results in a substantial reduction of velocity pointing errors (see Fig. 1b). Modification of the thrust profile as mentioned above by softening the ignition transient is called the Velocity Precision-pointing Enhancement System (VPES, pronounced "Vee-Pees"). The reduction of the pointing error arises from the fact that the perturbing body-fixed torque is averaged out over several revolutions of the spacecraft spinrate.

In this paper we discuss the mass savings enjoyed through implementation of VPES. We study two general cases where propellant savings are realized: *i*) Propellant saved during spin-up and despin maneuvers and *ii*) Propellant saved through less costly trajectory correction maneuvers. Since velocity pointing errors are reduced the spacecraft spinrate may be decreased significantly. As a result, propellant saved during spacecraft spin-up and despin maneuvers may be used later for stationkeeping, thereby extending satellite lifetime. If one chooses to maintain the high spinrate the decreased velocity pointing errors will result in a more precise ΔV and savings may be realized when the trajectory correction maneuvers are greatly reduced.

SPINUP SAVINGS

Supporting Background

The Velocity Precision-pointing Enhancement System (VPES) is a modification to existing rocket engine technology, that when implemented, may provide a significant reduction in velocity pointing errors[1,2].

For spinning thrusting spacecraft, the velocity pointing error angle, ρ, is given by

$$\rho = \frac{M_X}{I_z \Omega^2} \tag{1}$$

where M_X is the torque about the inertially fixed X-axis caused by thruster misalignment and offset, Ω is the vehicle spin rate, and I_z is the moment of inertia of the vehicle about the spin axis. For a cylindrical spacecraft the principal moment of inertia, I_z can be described by

$$I_z = \frac{1}{2} m r^2 \tag{2}$$

where m is the vehicle mass and r is the maximum radius of the spacecraft.

Application of VPES to any system introduces the pointing error reduction factor, C_{VPES}, which is a function of the current engine thrust profile, misalignment, and offset as well as the VPES duration and profile. For an engine with a step function profile and minor misalignment and offset[3] $C_{VPES} = 156$ whereas for Thiokol's Star 48B (the payload assist module) $C_{VPES} = 73$. Assuming the current pointing errors are acceptible, VPES permits lower spinrates to obtain the same level of precision. We may then describe the resulting lower spin rate, Ω_{new}, as a function of the original spin rate, Ω_{old}

$$\begin{aligned} C_{VPES} &= \frac{\rho_{new}}{\rho_{old}} = \frac{M_X / I_z \Omega_{new}^2}{M_X / I_z \Omega_{old}^2} = \frac{\Omega_{old}^2}{\Omega_{new}^2} \\ \Rightarrow \Omega_{new} &= \frac{\Omega_{old}}{\sqrt{C_{VPES}}} \end{aligned} \tag{3}$$

Thus after implementing VPES a spacecraft may be spun to a spin rate that is only a fraction (11.8% with the Star 48B) of its old value. Since many spacecraft are currently spun to very high spin rates this can be a potentially large savings in propellant needed to spin up and possibly despin. For example, the Anik spacecraft and rocket combination which uses the Boeing/Hughes HS-376 is spun to 50 rpm before firing the Star 48B (payload assist module) to provide the desired kick to enter the transfer orbit to geostationary orbit. Using VPES the needed spin rate would have only been 5.9 rpm.

We next consider the propellant savings obtained from the spin rate reduction. Starting from the equation for the spacecraft spin rate

$$\Omega = \frac{M_z t}{I_z} \tag{4}$$

where the new variable t is the time for the spin up burn. From the rocket equation[4] we have

$$F = I_{sp}\dot{m}g \tag{5}$$

where F is the force, I_{sp} is the specific impulse, $\dot{m}$ is the mass flow rate, and g is the acceleration due to gravity. Since the torque is given by $M = rF$ we may insert Equation (5) into Equation (4) to get

$$\begin{aligned} \Omega I_z &= M_z t \\ &= rFt \\ &= r(I_{sp}\dot{m}g)t \end{aligned} \tag{6}$$

which we rearrange to obtain the propellant mass used to perform the spin up burn:

$$\begin{aligned} m_{prop} &= \dot{m}t \\ &= \frac{\Omega I_z}{I_{sp}gr} \end{aligned} \tag{7}$$

Now we take the difference between the propellant mass needed to spin up at the different spin rates to obtain the mass saved during spinup (m_{GTOb}),

$$\begin{aligned} m_{GTOb} &= m_{old} - m_{new} \\ &= \frac{\Omega_{old} I_z}{I_{sp}gr} - \frac{\Omega_{new} I_z}{I_{sp}gr} \\ &= \frac{I_z(\Omega_{old} - \Omega_{new})}{I_{sp}gr} \end{aligned} \tag{8}$$

As shown in Equation (3), $\Omega_{new} = \Omega_{old}/\sqrt{C_{VPES}}$. Thus Equation (8) becomes

$$\begin{aligned} m_{GTOb} &= \frac{(1+\sqrt{C_{VPES}})I_z\Omega_{new}}{I_{sp}gr} \\ &= \frac{(1+\sqrt{C_{VPES}})(m_{sat} + m_{GTO\ booster} + m_{GEO\ booster})r\Omega_{new}}{2I_{sp}g} \end{aligned} \tag{9}$$

where m_{sat} is the satellite mass, $m_{GTO\ booster}$ is the mass of the GTO rocket, and $m_{GEO\ booster}$ is the mass of the rocket used for the geostationary orbit insertion. The sum of these three terms is the total spacecraft mass fully loaded with propellant. Thus we save 88% of the propellant normally used for spin up maneuvers. However, now the engine is pushing a slightly higher mass which in turn imparts a slightly reduced ΔV. The ΔV imparted to the vehicle is described in terms of the spacecraft mass before (m_b) and after (m_a) the maneuver and is given by

$$\Delta V = gI_{sp} \ln\left(\frac{m_b}{m_a}\right) \tag{10}$$

In the implementation of VPES the masses m_b and m_a have changed while the ΔV and I_{sp} remain constant. Since VPES allows the spacecraft to spin at lower spin rates the engine must push a larger mass. Thus rearranging Equation (10) and inserting the appropriate masses for m_b and m_a we arrive at

$$\begin{aligned} \frac{m_b}{m_a} &= \frac{m_{sat} + m_{GTO\ booster} + m_{GEO\ booster}}{m_{sat} + m_{GTO\ booster\ empty} + m_{GEO\ booster}} \\ &= e^{\Delta V/gI_{sp}} \end{aligned} \tag{11}$$

However, after application of VPES the ΔV and I_{sp} remain the same so that the second line of Equation (11) is constant. The introduction of VPES has increased the mass before the maneuver from m_b to $m_b + m_{GTOb}$ where m_{GTOb} is derived in Equation (9). In a similar manner the mass following the maneuver has been increased so that m_a becomes $m_a + m_{GTO}$. Since this extra mass must then be pushed all the way to the geostationary orbit we expect the actual savings from the implementation of VPES, m_{GTO}, to be smaller than the raw spinup savings given by m_{GTOb}. (In effect, some of the saved mass must be used to help propel itself out to the higher orbit.) To show the relationship between m_{GTO} and m_{GTOb} we begin with the observation that

$$\begin{aligned} e^{\Delta V_{GTO}/gI_{sp}} &= \frac{m_b + m_{GTOb}}{m_a + m_{GTO}} \\ &= \frac{m_b(1 + m_{GTOb}/m_b)}{m_a(1 + m_{GTO}/m_a)} \\ &= e^{\Delta V_{GTO}/gI_{sp}} \frac{1 + m_{GTOb}/m_b}{1 + m_{GTO}/m_a} \end{aligned} \tag{12}$$

$$\Rightarrow \quad 1+\frac{m_{GTO}}{m_a} = 1+\frac{m_{GTOb}}{m_b}$$

$$\Rightarrow \quad \frac{m_{GTOb}}{m_{GTO}} = \frac{m_b}{m_a} = e^{\Delta V_{GTO}/gI_{sp}} \tag{13}$$

$$\Rightarrow \quad m_{GTO} = m_{GTOb}\left(e^{-\Delta V_{GTO}/gI_{sp}}\right)$$

A similar procedure is then applied to the orbit insertion burn (GEO) since VPES is then applied to it as well. However, during this burn, the mass has been reduced since the GTO rocket has been spent but the mass also must reflect the fuel saved from the application of VPES to the GTO burn leaving us with

$$m_{GEOb} = \frac{(1+\sqrt{C_{VPES}})(m_{sat}+m_{GEO\ booster}+m_{empty\ GTO\ booster}+m_{GTO})r\Omega_{new}}{2I_{sp}g}$$

$$m_{GEO} = m_{GEOb}\left(e^{-\Delta V_{GEO}/gI_{sp}}\right) \tag{14}$$

This savings in weight may then be translated into either a larger payload mass or more station keeping fuel, extending the life of the vehicle.

Often the spacecraft must also be despun after the injection maneuver has been performed. This may be accomplished by either using a thruster couple and performing a similar analysis as outlined above or with a yo-yo device in which expendable masses are attached to the satellite by light cords[5]. Using the thruster couple method we arrive at a mass savings during despin (m_{despin}) of

$$\begin{aligned} m_{despin} &= \frac{(1+\sqrt{C_{VPES}})I_{znew}\Omega_{new}}{I_{sp}gr} \\ &= \frac{(1+\sqrt{C_{VPES}})(m_{sat}+m_{GTO}+m_{GEO})r\Omega_{new}}{2g(I_{sp})_{thruster}} \end{aligned} \tag{15}$$

In the case of a yo-yo device the equation relating the mass of the expendable masses saved by a lower spin rate (m_{yo-yo}) is

$$m_{yo-yo} = \frac{I_z}{l^2\left(\frac{\Omega_{old}+\Omega_{new}}{\Omega_{old}-\Omega_{new}}\right) - r^2} \tag{16}$$

where I_z is the moment of inertia after the burn and l is the length of the cords holding the expendable masses.

There are several methods by which we may interpret the value of the VPES mass savings. One method is a direct application of estimated costs per pound for spacecraft in different orbits. A rough value frequently used is that communication satellites cost \$10,000/lb. Another method to quantify the savings is by normalizing the mass saved in Equations (9)-(15) by the nominal spacecraft mass. In this case, the fractional mass savings ($f_{savings}$) is represented by

$$f_{savings} = \frac{m_{GTO} + m_{GEO} + m_{despin}}{m_{sat} - m_{sk}} \tag{17}$$

where m_{sk} is the mass of station keeping propellant.

Perhaps the best method of reflecting the savings to the industry is to quantify the extension of the spacecraft lifetime. Because the same propellant reserves used to correct the velocity pointing errors are also used in station keeping, VPES permits the satellite to remain aloft for longer. The result is increased revenue by the operating agency. In the mission design case study presented by Humble, Henry and Larson[4] it is stated that "extended on-orbit life is worth \$50M per year." If we know the intended lifetime, T_L (in years), and we know the amount of station keeping propellant on board, m_{sk}, we may determine the extended time VPES provides. We may then calculate the extended lifetime, T_E (in years) by

$$T_E = (m_{GTO} + m_{GEO} + m_{despin})\frac{T_L}{m_{sk}} \tag{18}$$

To translate this into a monetary savings we simply multiply T_E by \$64.5M/yr which is the \$50M/yr reported in Chapter 10 of Ref. [4] corrected for inflation (which is 2.9% per year, so \$1 in 1992 is worth \$1.29 in 2002). Our final representation of the impact of VPES is described in a case study.

Case Study: Telesat Anik (HS-376)

Since the mid-1980's Hughes Space and Communications Division has delivered more than 56 spacecraft using their spacecraft bus HS-376. Now owned by Boeing, the HS-376/BSS-376 is still in high demand and flew four satellites last year (2002) alone, while Boeing launched at least 11 GEO communications satellites in 2002 (four BSS-376's, four BSS-601's, and three BSS-702's). Typically, the mass of satellites supported by the HS-376 range in mass from 1200 to 1450 pounds of which 200 to 350 pounds are used for station keeping. The station keeping is usually maintained by four hydrazine thrusters similar to the MR 106 which has an I_{sp} of 232 seconds.

Table 1
TELESAT CANADA'S ANIK-C RELEVEANT PARAMETERS

Parameter	Value
Satellite mass, m_{sat} =	568 kg (1250 lbs)
Station keeping propellant mass, m_{sk} =	100 kg (220 lbs)
Radius, r =	1.08 m (3.54 ft)
Spinrate, Ω =	5.24 rad/sec (50 rpm)
Lifetime, T_L =	10 years
Star 48 mass, m_{S48} =	2141 kg (4710 lbs)
Star 48 empty mass, m_{S48e} =	232 kg (510 lbs)
Star 48 I_{sp} =	292 sec
Star 30 mass, m_{S30} =	492 kg (1082 lbs)
Star 30 empty mass, m_{S30e} =	28 kg (62 lbs)
Star 30 I_{sp} =	293 sec
Hydrazine MR 106 Thruster I_{sp} =	232 sec

The HS-376 is launched from a Delta, the Space Shuttle, or Ariane and is spin-stabilized at 50 rpm during the transfer orbit. Transfer orbit injection is accomplished by the PAM (Star 48) and the final apogee circularization burn is provided by the Star 30. As a result, the VPES savings would apply to both the Star 48 and Star 30 which are manufactured by Thiokol and have error reduction factors of $C_{VPES} = 73$.

Telesat Canada's Anik-C spacecraft gives us a good opportunity for our case study and is typical of satellites in this category. Table 1 gives the relevant parameters. The equations of the preceding section permit us to calculate the mass saved during the spinup and de-spin maneuvers, as well as the corresponding $f_{savings}$ and T_E. Table 2 provides a summary of the relevant calculations.

From the table we see that we are able to increase the lifetimes by over 4% and in-

Table 2
VPES SPINUP SAVINGS (IN 2002 DOLLARS) FOR TELESAT CANADA'S ANIK-C FINAL RESULTS

Savings	Mass [kg]	$f_{savings}$ [%]	T_E [yr]	$ per Launch [$M]
GTO	1.41	0.3	0.14	9.09
GTO Despin	1.42	0.3	0.14	9.09
GEO	0.66	0.14	0.07	4.26
GEO Despin	0.66	0.14	0.07	4.26
GTO+GEO+Despin	4.15	0.9	0.42	26.77

Table 3
SPACECRAFT BREAKEVEN REPAIR COSTS FOR COMMUNICATION SATELLITES LAUNCHED FROM 1993 TO 1996 TRANSFERRED INTO 2002 DOLLARS[6]

	Commercial	Civil	Defense
No. of Spacecraft	116	30	68
Low	\$52M	\$77M	\$81M
Average	\$121M	\$200M	\$366M
High	\$328M	\$968M	\$947M

crease the usable mass by 0.9%. George Levin's review of a study by INTEC[6] provides a perspective of the quantity and value of satellites launched by the communications aerospace industry. At a conference entitled "Prospects for Commercialization of SELV-Based In-Space Operations" held on October 18-19, 1993, George Levin of NASA reviewed a study conducted by INTEC. The study concludes that typical net revenues for communications satellites are in excess of \$100M per year over a 10-year life which is \$126M per year in 2002 dollars. Table 3 provides some of the results of the INTEC report. From the table we note that some spacecraft can cost as much as \$968M; while the average communications satellite cost is \$210M. This study included satellites sent to both low Earth orbit as well as geostationary Earth orbit. Humble, Henry, and Larson[4] indicate that GEO spacecraft have a figure of merit of \$600M which is consistent with the INTEC study[6].

Finally the equations of this section may be easily applied to any spacecraft given the original spinrate, mass, radius, propellant quantity, and lifetime.

TRAJECTORY CORRECTION SAVINGS

In the previous section we assumed that the existing errors were acceptable. We then used VPES to lower spacecraft spinrate to save station keeping propellant. In this section we investigate the situation in which the previously acceptable pointing errors are no longer acceptable and must be eliminated by a trajectory correction maneuver (TCM). We will show that the TCM savings enjoyed by the VPES system can be quite significant. We focus on the GEO circularization burn as an example.

First we introduce the relative motion equations which relate orbits that are close to each other. The derivation for such equations of motion may be found in any text on orbit mechanics (see Prussing and Conway[7] Chapter 8 or Kaplan[5] Chapter 3). The result is to derive the Hill-Clohessy-Wiltshire equations which describe the motion relative to the target circular orbit and are given by

$$
\begin{aligned}
\ddot{x} - 2n\dot{y} - 3n^2x &= 0 \\
\ddot{y} + 2n\dot{x} &= 0 \\
\ddot{z} + n^2z &= 0
\end{aligned} \tag{19}
$$

where the variables x, y, and z are defined as deviations in the radial, tangential, and axial directions respectively when compared to the target orbit (see Figure 2). The parameter n is defined as

$$
n = \sqrt{\frac{GM}{R^3}} = \sqrt{\frac{\mu}{R^3}} \tag{20}
$$

where G is the Newtonian universal constant of gravitation, M is the mass of the Earth and R is the radius of the target orbit. We have also introduced the gravitational parameter, $\mu = GM$, for the body of interest.

Solving these equations gives

$$
\begin{aligned}
x(t) &= \frac{\dot{x}_0}{n}\sin nt - \left(\frac{2\dot{y}_0}{n} + 3x_0\right)\cos nt + \left(\frac{2\dot{y}_0}{n} + 4x_0\right) \\
y(t) &= \frac{2\dot{x}_0}{n}\cos nt - \left(\frac{4\dot{y}_0}{n} + 6x_0\right)\sin nt + \left(y_0 - \frac{2\dot{x}_0}{n}\right) - (3\dot{y}_0 + 6nx_0)t \\
z(t) &= z_0\cos nt + \frac{\dot{z}_0}{n}\sin nt
\end{aligned} \tag{21}
$$

where values denoted by the subscript 0 are from the initial conditions. From these equations we see that a velocity error along the y axis, $\dot{y}_0$, which is equivalent to a ΔV magnitude error results in a secular error due to the time term, t, in the last expression

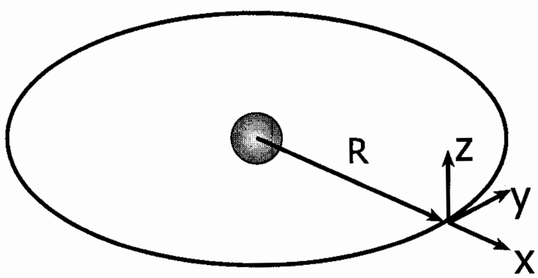

Figure 2 Definitions of variables used in the Hill-Clohessy-Whiltshire Equations.

of the $y(t)$ solution. This means that a magnitude error has a disastrous effect on the orbit, because the position error grows (without bound in these linearized equations). This is easily understood as the effect on period that a magnitude error will have during the circularization burn: the orbit period will not be exactly geosynchronous and the spacecraft will quickly drift away from its intended orbit. On the other hand, velocity pointing errors contribute to the terms $\dot{x}_0$ and $\dot{z}_0$, which have no secular effects. Thus the $\dot{x}_0$ error causes oscillatory motion in radial distance, which is equivalent to an eccentricity different from zero and the $\dot{z}_0$ causes an oscillation in the out-of-plane direction, which corresponds to an inclination error. Small errors in eccentricity and inclination may be tolerable in some GEO satellites, as long as the orbit period is correct. The satellite will appear to oscillate in the sky, but will remain, on average in the correct orbit. However, in the application we are considering now, we assume that errors in eccentricity and inclination are not tolerable, and therefore must be removed by a TCM.

For a more quantitative analysis of the savings which result by a reduction of the velocity pointing error may be derived starting with the ΔV required for the maneuver. In this case we use the relationship describing the ΔV for a Hohmann transfer[5] in terms of the radii of the two orbits r_1 and r_2,

$$\Delta V = \sqrt{2\mu}\left[\sqrt{\left(\frac{1}{r_1} - \frac{1}{r_1 + r_2}\right)} - \sqrt{\frac{1}{2r_1}}\right]. \tag{22}$$

If the current pointing error of the system is ρ then the error in the ΔV is given by

$$\Delta V_{error} = (\sin\rho)\Delta V. \tag{23}$$

With the introduction of VPES the ΔV_{error} is reduced by the same factor C_{VPES} discussed in the preceeding section so that

$$\Delta V_{VPES} = \frac{\Delta V_{error}}{C_{VPES}} = \frac{(\sin\rho)\Delta V}{C_{VPES}}. \tag{24}$$

The resulting ΔV savings from the use of VPES is then described by

$$\begin{aligned} \Delta V_{saved} &= \Delta V_{error} - \Delta V_{VPES} \\ &= \left(1 - \frac{1}{C_{VPES}}\right)\Delta V_{error} \\ &= \left(1 - \frac{1}{C_{VPES}}\right)(\sin\rho)\Delta V. \end{aligned} \tag{25}$$

From Equation (10) we transform this ΔV_{saved} into a mass fraction of the spacecraft saved:

$$\begin{aligned}
\frac{m_{sat} + m_{saved}}{m_{sat}} &= \exp\left(\frac{\Delta V_{saved}}{g I_{sp}}\right) \\
&= \exp\left[\frac{\left(1 - \frac{1}{C_{VPES}}\right)(\sin\rho)\Delta V}{g I_{sp}}\right] \qquad (26) \\
\Rightarrow \qquad m_{saved} &= m_{sat}\left\{\exp\left[\frac{\left(1 - \frac{1}{C_{VPES}}\right)(\sin\rho)\Delta V}{g I_{sp}}\right] - 1\right\}
\end{aligned}$$

Using the example of Table 1, but assuming a parking orbit at 300 km ($r_1 = 6678$ km) and the geosynchronous orbit at $r_2 = 42,160$ km we arrive at the savings provided in Table 4.

We note that the TCM Savings in Table 4 are significantly greater than the Spinup savings of Table 2. We also understand that these savings are less likely to be applicable to the communications satellite industry since the existing precision is probably acceptable. However, the savings presented in this section represent those satellite systems which require high precision. In this case we assume the spacecraft is spun at the originally designed spin rate which produces pointing errors that are greatly reduced when VPES is implemented. The most likely application of this increased precision would be in a Department of Defense or military satellite.

IMPLEMENTATION CONSIDERATIONS

While control of an engine thrust profile is clearly nontrivial, it is a well understood process. For solid rocket motors manipulation of the propellant grain geometry and choice of fuel and oxidizers are just a few methods one may change an engine thrust history. For throttleable liquid rocket engines, the profile may be changed by simply modifying the propellant flow rates. However, most liquid engines do not possess

Table 4
VPES TCM SAVINGS (IN 2002 DOLLARS) FOR TELESAT CANADA'S ANIK-C

Savings	Mass [kg]	$f_{savings}$ [%]	T_E [yr]	$ per Launch [$M]
GTO	29.25	6.3	2.93	189
GEO	17.52	3.7	1.75	113
GTO+GEO	46.77	10.0	4.68	302

deeply throttling capability and so one may generate an average thrust profile through many judiciously timed pulses of the engine.

In addition to these fundamental methods there are several techniques which may be applied to existing engines. By pre-pressurizing the engine combustion chamber with a gas lighter in molecular mass than the propellant byproducts we can soften the ignition transient. It is also possible to install one-way relief "flapper" valves which allow combustion gases to escape without being expanded through the nozzle. Probably most simple would be the installation of a nozzle throat ring, which comprises a material that ablates away during the engine start. Clearly, there are many ways the ignition transient phase of an engine thrust history may be modified using current technology to achieve the benefits of VPES.

CONCLUSIONS

The Velocity Precision-pointing System can greatly reduce velocity pointing errors, in some cases by nearly two orders of magnitude. We have provided two generic methods in which these reduced errors translate into saved payload. In the first method, the spacecraft is spun to a lower spinrate so that a payload increase arises from savings of spinup propellant. In the second method the payload increase comes from precise orbit placement since much less propellant needs to be spent on initial trajectory correction maneuvers. In both cases we provided a simple cost analysis that interprets the mass savings in terms of extension of spacecraft lifetime and shows that the savings can be quite significant. Finally, we note that current technologies permit application of VPES to all rocket engines and may even be retro-fit in some instances.

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