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Propulsion Requirements and Options for the New Millennium Interferometer (DS-3) Mission

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Abstract

The New Millennium Interferometer (DS-3) Mission is described along with a new candidate observation plan which takes advantage of an observation mode in which data is collected as the spacecraft are in relative motion. This observation mode reduces both the time and total impulse requirements to complete a representative set of observations at three different levels of coverage quality. In addition to the aperture plane filling requirements, attitude control requirements which include retargeting slews and deadband control are also described. Three propulsion systems; Cold Gas, Pulsed Plasma Thrusters (PPT), and Field Emission Electric Propulsion (FEEP) are considered in a performance trade with respect to their capability to perform the attitude control and aperture plane filling roles. All three systems were found to meet the basic requirements described although each introduces specific challenges with respect to spacecraft integration and/or interactions. Specific issues associated with use of the PPT as well as a technology development plan in support of the possible use of PPTs on DS-3 are also described.

Introduction

The third deep space mission (DS-3) for the New Millennium Program is a separated spacecraft interferometer, scheduled for launch in 2002, designed to validate technologies associated with precision formation flying and spaceborne interferometry. These technologies are essential to ambitious 21st century missions as part of NASA's Origins Program requiring constellations of spacecraft flying in formation.

In the current DS-3 mission scenario, the spacecraft will be inserted into a heliocentric orbit for a nominal mission duration of nine months. Figure 1 shows the DS-3 configuration which consists of two collector spacecraft which collect and reflect a 12 cm. diameter starlight beam to a third combiner spacecraft. Starlight traveling through the two optical paths are compressed and combined to produce an interference pattern or series of fringes. The two primary detectors are a CCD fringe spectrometer and avalanche photodiode[1,2] operating in the optical bands (550 - 900 nm). Information related to the amplitude of these fringes is collected at a discrete number of points within a synthetic aperture (u-v Fourier Transform) plane corresponding to a two dimensional plane in physical space (see Figures 1 and 2). By taking the inverse transform of this information it is possible to reconstruct an optical image. Interferometers are characterized by much higher angular resolution within a narrower field of view than obtainable with a single aperture, monolithic telescope. With a separated spacecraft interferometer, baselines on the order of a kilometer or larger can be achieved resulting in a previously unattainable capability to observe the fine scale structure of stellar objects.

Independently controlled, separated spacecraft flying in precise formation result in demanding requirements on the propulsion control system. For DS-3, the optical pathlength will be actively controlled through a series of mechanisms with increasingly finer level of resolution. At the coarsest level is the spacecraft propulsion system which must control the position of the spacecraft with 1 cm resolution over a range of hundreds of meters. An optical delay line located on the combiner spacecraft uses a voice coil actuator to bring the control down to 10 micron resolution over a 1 cm. range and a piezoelectric transducer controls movements down to 1 nm resolution over a 10 micron range. Optical pathlengths are determined with a laser metrology subsystem. A critical component of the formation flying technology demonstration is the Autonomous Formation Flying (AFF) sensor on each of the spacecraft. The AFF is an RF based sensor which enables relative separations and angles between the spacecraft to be determined. With this system the spacecraft can be positioned within the 1 arcminute, 1 cm acquisition range of the laser metrology[1]. The current mission design incorporates some fundamental changes over the previously described[3] mission concept. In particular, the requirement that the separated spacecraft come to rest relative to each other during data collection has been relaxed permitting consideration of the so called "observe-on-the-fly" mode in addition to the "stop-and-stare" mode previously investigated. This fundamental change has led to consideration of novel aperture filling strategies with different requirements levied on the propulsion system over those considered in the previous trade study. In addition, the desired quiescent time period without thruster firings for deadband control has been increased. These changes have altered the trade space significantly enough to warrant a new evaluation of the options. This paper summarizes the results of this trade.

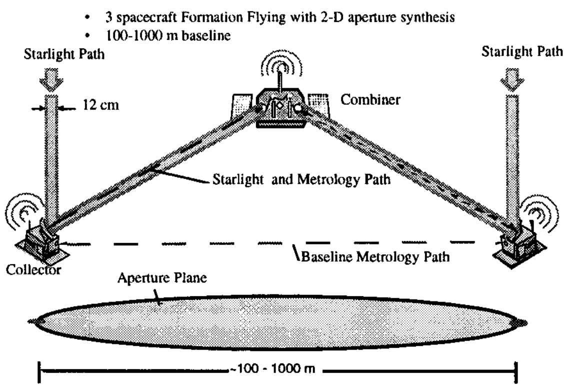


Figure 1. The New Millennium Separated Spacecraft Interferometer (Figure courtesy of G. Blackwood, JPL)

The next section will describe specific mission requirements which drive the propulsion technology. This will be followed by a summary of the trade study performed as part of the pre-phase A activity. The last section will describe specific issues associated with use of the PPT as well as a technology development plan in support of the possible use of PPTs on DS-3.

Mission Requirements

Mission Overview

The DS-3 mission is nominally expected to last nine months during which experiments will be performed to demonstrate a number of different technologies related to precision formation flying and spaceborne interferometery. The different mission phases and approximate duration are listed in Table 1. Initially, the three spacecraft are physically connected to each other forming a rigid, fixed baseline interferometer in a configuration referred to as a cluster. After injection into a SIRTF like Earth trailing orbit ($C_3 = 0.5 km^2/s^2$) aboard a Delta II launch vehicle the three spacecraft will begin operations in the cluster mode for four months. During this time spacecraft and interferometer checkouts will be performed, and the instrument will be used for Earth and celestial target imaging. Some subset of the thrusters on all three spacecraft will be required to provided attitude control of the cluster during this period.

After this phase of the mission is complete, a separation mechanism will release the three spacecraft with some drift velocity. The propulsion systems on each spacecraft will bring them to rest relative to each other once they have drifted to a separation distance deemed to be safe for thruster firing, probably on the order of 50 - 100m. Experiments focusing of formation flying will then be performed for approximately one month. These experiments will test the capabilities of different formation sensors and actuators as well as fault recovery maneuvers.

Mission Phase	Duration (months)
Cluster Mode Operation	
Cluster Spacecraft Checkout	1
Cluster Interferometer Checkout	2
Cluster Observations	0.5
Formation Flying Demonstration	
Formation Flying Checkout	0.5
Formation Flying Experiments	1
Spaceborne Interferometry Demonstration	
Celestial Observations	4
TOTAL	9

Table 1. Mission Timeline

The final celestial observation phase is expected to last approximately four months. It is during this phase that the aperture plane filling maneuvers described in the next section will be carried out.

Baseline Changes and Aperture Filling

Figure 2 is a diagram showing the three spacecraft in a triangular formation. The aperture plane is a region of physical space in which a number of measurements of fringe amplitude are collected for later reconstruction into an image. In the coordinate system shown in Figure 2, the aperture plane is represented by a dotted circle with a diameter equal to the baseline for that particular data point. Various strategies for "filling" the aperture plane have been proposed and no plan has yet emerged as being optimal with respect to all the variables being considered; i.e., time, propellant usage, contamination, data point density etc. While no observation plan has yet been officially adopted by the project team, one candidate is presented here which is an adaptation of an earlier approach[2,3] but takes advantage of the ability to collect data while in relative motion to minimize time and propellant. In the "observe-on-the-fly" mode the two collectors accelerate away from each other in the X-Y plane increasing their baseline. Once their relative velocity has reached a value of 5.0 cm/sec the thrusters are turned off and the two vehicles coast away from each other for a predetermined period of time and then an opposite pair of thrusters is fired to decelerate and bring the spacecraft to rest. Except in the lowest thrust cases, the coasting period is longer than the acceleration and deceleration periods and provide an opportunity for collecting data with no translational thruster firings. The only thruster firings during the coast period are those required to maintain the deadband as described later.

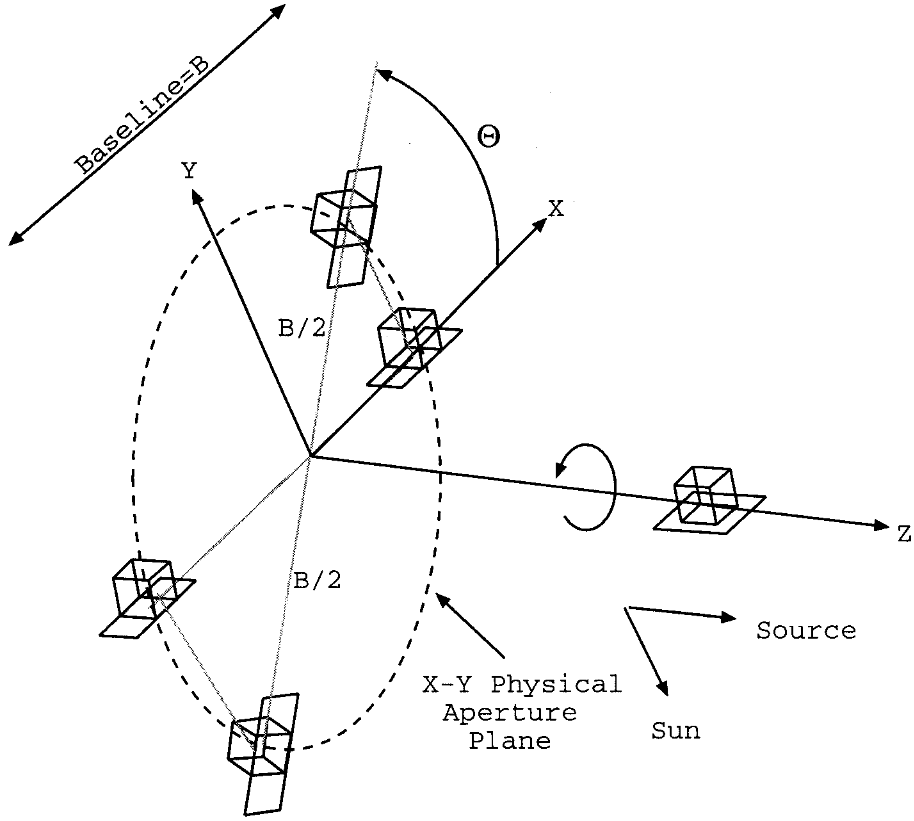


Figure 2. Diagram showing positions of collector spacecraft before and after movement along a straight path in the X-Y aperture plane. Collector spacecraft translation along non-radial path requires rotation to remain in proper orientation (see Figure 5). Dotted line represents circular synthetic aperture with diameter equal to baseline "B". Rotation of combiner not shown for clarity.

Filling the aperture plane involves movements of the collectors which is a combination of purely radial and non-radial moves. Figure 2 shows the two collectors in a position before and then after a non-radial move to another position with the same baseline but different angle with respect to the X axis (this is the Θ coordinate in Figure 2). The design of the collection mirrors (siderostats) and their gimbal drive mechanisms imposes limitations on the angles between the spacecraft. The observation plan therefore calls for the formation to be in an equilateral triangle configuration which is maintained during all formation re-sizing and pointing. In the plan described here, the two collectors always move in the X-Y plane which implies the combiner spacecraft must move in the negative Z direction (towards the X-Y plane) when the baseline is reduced and positive Z direction (away from the X-Y plane) when the baseline is increased in order maintain the equilateral configuration. Because no purely tangential moves (along a circular arc) are considered in this plan, a non-radial movement as shown in Figure 2 would require the combiner to move first in the -Z direction, and then back in the +Z direction to its original location while rotating in order to maintain the equilateral configuration.

As the two collector spacecraft move, their paths map a pattern on the (X-Y) physical aperture plane. This mapping is a convenient representation in order to describe the observation strategy. There are varying degrees of coverage of the aperture plane which yield useful information. If fringe data is collected at a number of points along a line (along the X-axis for example) then one can reconstruct a one dimensional image of the source which may be adequate if the source is circularly symmetric[4]. More complete coverage of the aperture plane will provide more detailed information at the expense of additional observation time and perhaps fewer total number of objects which can be observed. The observation plan described here consists of a mix of objects observed with what will are referred to as one dimensional (1-D), limited two dimensional (L-2D), and full two dimensional (F-2D) imaging following the conventions used previously[2,3].

Figure 3 shows the mapping of the two collector spacecraft paths onto the physical aperture plane for all three levels of coverage. In Figure 3a), the (1D) coverage level is just a straight path from 50 - 500m (in radius) along the X axis. In Figure 3b), the (L2D) coverage level is a 45 degree triangle formed by three paths. Finally, in Figure 3c), the (F2D) coverage consists of 12 paths which span the quadrant (Q1) in the X-Y plane. As one collector traverses these paths the second moves along the mirror image in quadrant Q2. The constellation will be oriented with one side facing the sun at some angle (see solar arrays and angle Θ in Figure 2). A maximum propulsion system

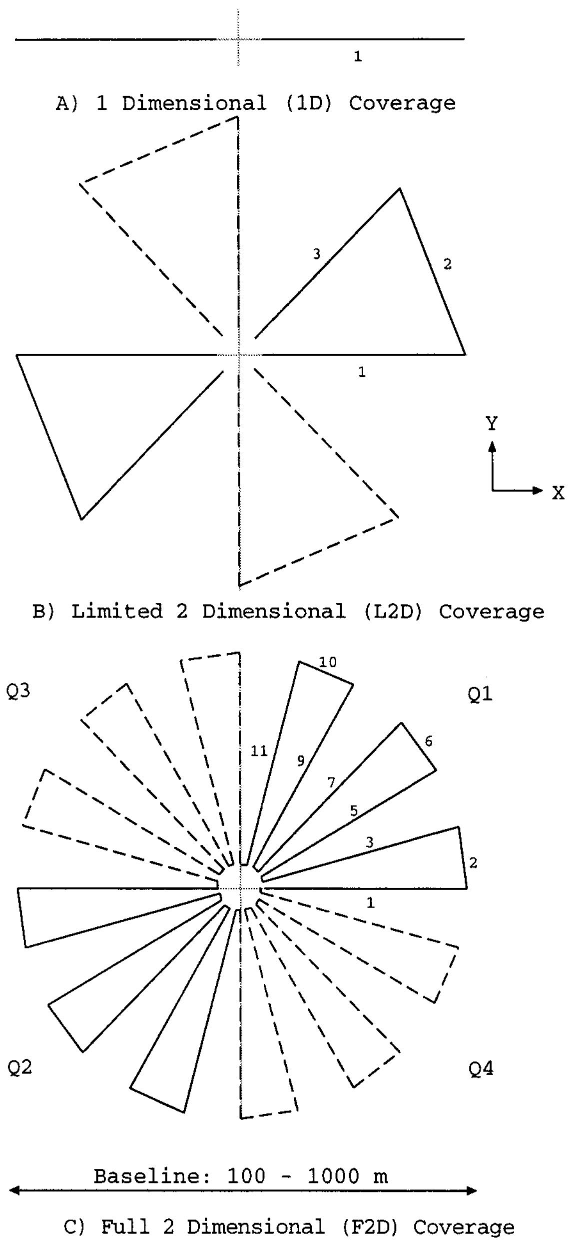


Figure 3. Collector spacecraft paths in the X-Y physical aperture plane for one dimensional (1D), limited two dimensional (L2D), and full two dimensional (F2D) levels of coverage.

power allocation of 150W was adopted as a target for the purpose of this trade. The power requirement limits the amount of plane coverage which can be achieved for a given sun-source angle. For this reason, the second half of the image plane, represented by the dotted paths in quadrants Q3 and Q4 is mapped three months later when the sun-source angle has rotated by 90 degrees. Because the celestial observation phase of the mission is only four months long, any L2D and F2D observations would have to be performed during the first and last month of this phase to enable complete coverage. The candidate observation plan used for the analyses presented in this paper consists of sixty objects imaged at the 1-D level, thirty five at the L-2D level, and five at the F-2D level. This distribution is somewhat arbitrary at this point and would eventually have to account for the fact that L2D and F2D observations would all have to occur in the first and last month of this phase as noted earlier due to sun angle limitations.

The impulse required to complete this observation plan was calculated assuming a spacecraft mass of 250 kg and a thruster couple operating at 1.05mN. The maximum collector velocity along any path was limited to 2.5 cm/sec. These results are presented in Figure 4. The impulse includes a ten percent margin as well as an additional allocation for four hundred 90 degree slews (the impulse for an additional one hundred slews is included in the budget for the formation flying experiment phase of the mission). The maneuver time, also including a ten percent margin is plotted as well.

Several features of the curves in Figure 4 can be attributed to the fact that the relative collector spacecraft velocity has been limited to 5 cm/sec (2.5 cm/sec for each collector). First the difference in total maneuver time over the range of thrust levels considered is less than a month. The reason for this is that much of the time is spent coasting, so as thrust is increased, the burn time needed to get to the maximum velocity is decreased but this only results in a modest decrease in the total time. In the limit of infinite thrust, the time curve would asymptote to a value one would have if all the paths were traversed at 2.5 cm/sec. The impulse curve has a distinct knee at approximately 0.6 mN. At low thrust levels, the acceleration is low enough that there is no coast period for any of the paths. At some value of the acceleration, the maximum translational velocity is reached and an intermediate coasting period is the result. Once this point is reached, the impulse for that path is no longer a function of the thrust level even though the burn time still is. This changes the rate at which impulse is accumulated as a function of thrust and the result is a change in the slope of the impulse curve as seen in Figure 4.

Attitude Control

As described in the introduction, the spacecraft propulsion represents the coarse adjustment in the overall control of the optical delay line. To prevent saturation of the delay line as well as the siderostat gimbals[1] the propulsion system must be capable of maintaining spacecraft angular position with one arcminute and translational position within one centimeter. To first order, translational thruster firings and the subsequent displacement of the spacecraft represent a more significant disturbance to the optical path length than a purely rotation firing. For this reason the desired quiescent time between thruster firings to stay within the deadband is

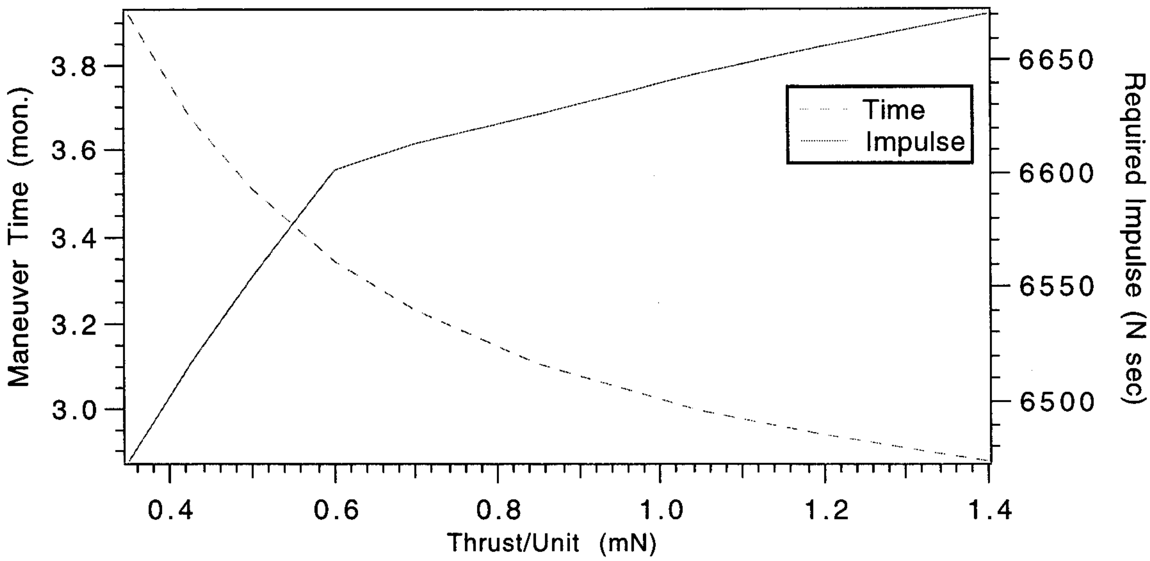


Figure 4. Time (in 30 day months) required to complete candidate observation plan and required impulse as a function of thrust level.

different for different phases of the mission. For cluster mode, minimum quiescent times of 2500 sec (42 min) are desired although rotational firings could occur more frequently. During the formation flying experiments phase of the mission as well as during celestial observations, a period of 3600 sec (60 min) is desired. During periods when the laser metrology system is going through its acquisition and initialization sequence, a period of 10 sec

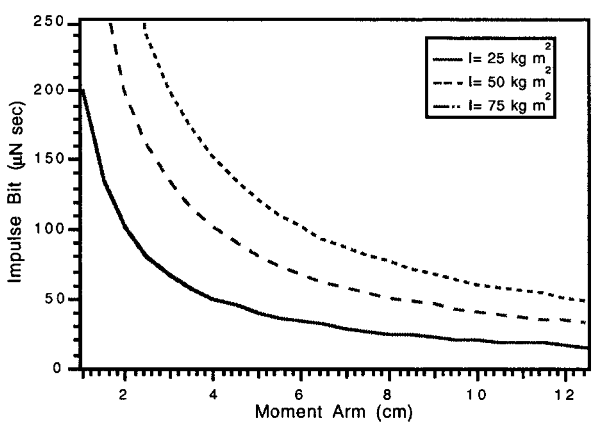


Figure 5. Maximum allowable impulse bit to insure 3600 sec quiescent time (in rotation) as a function of moment arm and inertia.

without any (translational or rotational) firings of any thruster in the entire formation is needed. For the purposes of this trade, a goal of 3600 sec between translational or rotational thruster firings was selected to be conservative. Figure 5 shows the maximum allowable impulse bit consistent with a 3600 sec quiescent time in rotation as a function of moment arm and inertia about the rotational axis which is of course strongly configuration dependent.

In addition to fine pointing control, the attitude control function must also provide for rotation maneuvers or slews both for retargeting and fault recovery. A desired

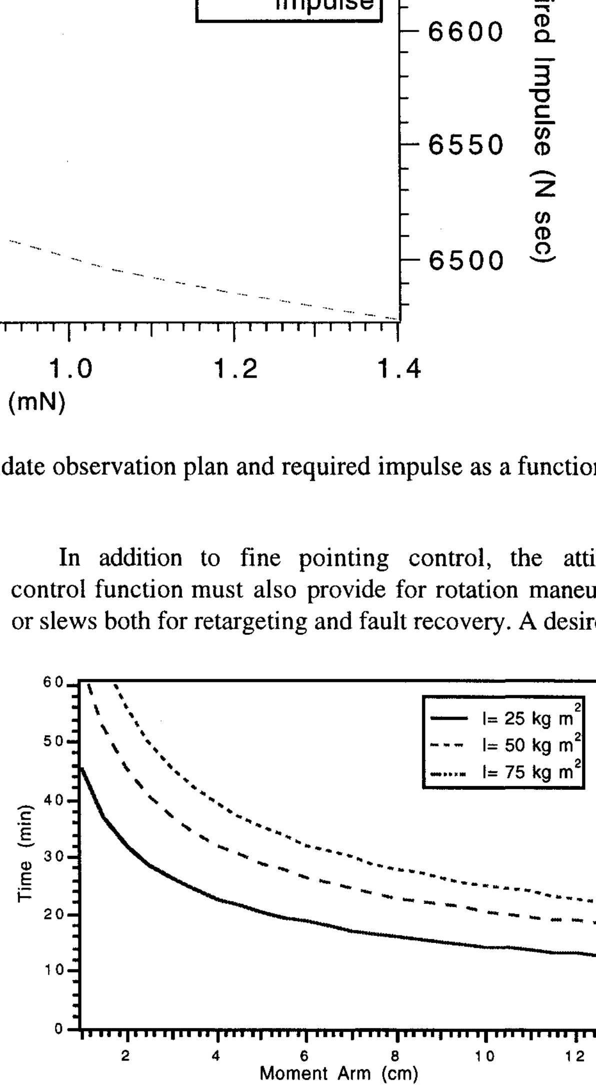
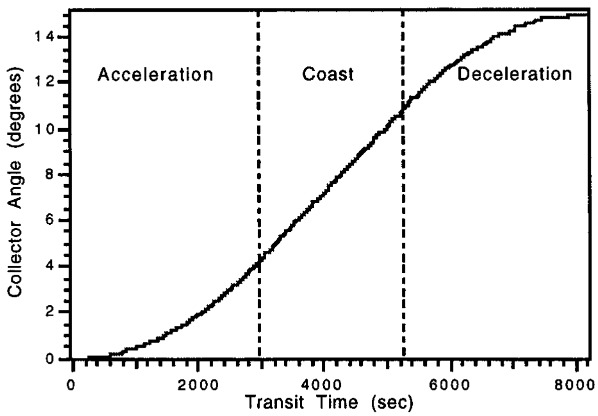


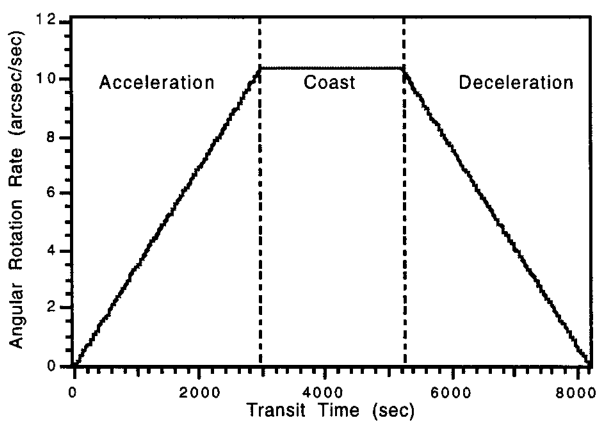
Figure 6. Time required to complete 90° slew (with zero initial and final angular velocity) as a function of moment degree rotation with a thruster couple operating at a thrust level of 1.05 mN as a function of moment arm and inertia.

time of no longer than 30 minutes to complete a 90 degree with zero initial and final angular velocity was selected as a goal. Figure 6 shows the time required to perform a 90 degree rotation with a thruster couple operating at a thrust level of 1.05 mN as a function of moment arm and inertia.

Another challenging attitude control requirement emerges from the need to maintain laser metrology lock, that is stay within the 1 arcminute angular deadband, during non-radial movements in the aperture plane. Referring again to Figure 2, which shows the two collector spacecraft before and after such a non-radial



A) Collector Angle vs. Time



B) Collector Angular Rotation Rate vs. Time

Figure 7. a) Collector spacecraft angle (Θ in Figure 2) and b) angular rotation rate required to remain pointed through origin of X-Y aperture plane (and maintain laser metrology lock) during transit of path 2 in Figure 3c. Translational acceleration and deceleration profile assumes thruster couple operating at 1.05 mN.

movement, it is clear that in order to remain pointed towards each other within the +/- 30 arcsec constraint requires them to rotate in prescribed manner determined by the velocity of the spacecraft along the path. Figure 7a is a plot of the collector spacecraft angle as a function of time for the case of path 2 (or equivalently paths 6, 10 etc.) in Figure 3c corresponding to the F2D plane coverage case. The dotted lines in Figure 7a indicate the time at which the transition occurs from translational acceleration to coasting, and from coasting to translational deceleration. Figure 7b is a plot of the corresponding rate of change of the spacecraft angle as a function of time for the same path. Although not easily discerned from the plot, the angular rotation rate during the coast period is not constant either, although its variation is much smaller than during the acceleration-deceleration periods. It is evident from Figure 7b that if this angular rate is not carefully controlled, it will not take long before the limits of the deadband are exceeded and the metrology lock is lost. This suggests that rotational deadband control thruster firings will have to be more frequent during the non-radial translations than during purely radial translations. A possible consequence of this could be the inability to collect data during these non-radial translations in the observe-on-the-fly mode. Some number of data points may have to be collected in the stop-and-stare mode especially during the longer non-radial paths in the L2D coverage case (i.e. path 2 in Figure 3b).

Propulsion Trade

Assumptions and Options Considered

Three options, listed in Table 2 were considered in the present trade study. These were cold gas nitrogen thrusters, Teflon® pulsed plasma thrusters (PPT) and Field Emission Electric Propulsion (FEEP). For each technology, performance at thrust levels of 0.35 and 1.05 mN were considered with performance assumptions listed in the first four rows of the table. The cold gas system is based on the Cold Gas Microthruster currently under development at MOOG Inc[5]. The low thrust and minimum impulse bit is achieved through a combination of low pressure operation (1 and 5 psia for 0.35 and 1.05 mN respectively), fast valve, and less than 200 micron throat diameter. The PPT is a Primex Aerospace Co. design with significant heritage from EO-1 which is described more fully in a later section of this paper as well as Reference 6. The thrust levels assumed correspond to operation at pulse frequencies of 0.5 and 1.5 Hz with approximately 40 J, 700 μN sec pulses. The minimum impulse bit corresponds to pulses energies of approximately 5 J. The FEEPs which are a cesium liquid metal ion thruster under development by Centrospazio are described fully in References 7 - 10. The two thrust levels considered assumed a 7 cm slit operating at emission currents of 3.2 and 8.8 mA for the low and high thrust levels respectively. The thrust specific power for these emitters was assumed to be 60.0 W/mN[11]. The neutralizer assumed a low work function thermionic cathode with a filament heater and emission current specific power of 0.2 W/mA[11,12]. FEEPs are currently under consideration for use on a drag free technology demonstration mission as well as the Laser Interferometer Space Antenna (LISA), a 21st century gravity wave detection mission. While combinations of different thruster technologies remain under consideration, reaction wheels were ground ruled out because of concern with vibration at frequencies which cannot be adequately filtered out by the delay line.

The second section of Table 2 summarizes the performance of the different technologies with respect to the aperture plane filling requirements for the candidate observation plan described earlier. The first row lists the time (in 30 day months) required to complete the movements. The second row in this section lists total impulse required. For a given spacecraft mass the movement time and impulse are only a function of the thrust level. Three assumptions make the required time and impulse calculations conservative. First, the aperture filling calculations assumed the higher mass of the combiner (250 kg) instead of the lower collector mass (150 - 175 kg). Secondly, the spacecraft mass was assumed constant. Thirdly, a ten percent margin was added to the impulse and time needed for the aperture plane coverage. In addition to the impulse required to complete the observation plan, the totals listed include an allocation for the cluster and formation flying mission phases (720 and 1300 Nsec respectively) as well as an additional allocation for retargeting slews (1890 Nsec equivalent to five hundred 90 degree turns).

The next three rows summarize the propellant mass required, as well as the dry and wet mass for the entire propulsion subsystem. The last row in this section is an estimate of the propulsion subsystem volume required. No specific thruster-spacecraft configuration is being proposed in this study. However, it was necessary to make some assumptions in order to estimate dry mass and volume. In order to do this, the only requirement assumed was that the configuration ensure decoupled rotations and translations about all three axes. The number of thrusters per cluster assumed designs which have been proposed by the sources for these technologies and are described below.

For the cold gas system, the dry mass and volume are dominated by the nitrogen tanks. For this analysis, the collector spacecraft are assumed to have two cylindrical, carbon composite wound tanks with 10 mil aluminum liners approximately 90 cm in length and 18 cm in diameter. The cold gas system is assumed to consist of four triad thruster clusters mounted on four corners of the spacecraft bus to achieve decoupled translations and rotations about all three axes.

In order to estimate dry mass and volume, the PPT system was assumed to consist of six units, one on the center of each face of the spacecraft bus with two fuel bar electrode sets and one capacitor each. The two fuel bars on each unit are spaced approximately six inches apart and could be canted at some angle if needed. This configuration was chosen because it provides the required functionality in terms of decoupled rotations and translations with a relatively simple design. The actual configuration would be very dependent on the configuration of the instrument and shields on a particular bus design and Reference 6 explores a number of possibilities. For example, it is unlikely that a PPT could be mounted in the center of the face which houses the interferometer in the case of the combiner or the siderostat in the case of the collectors.

The FEEP dry mass and volume assumes four quad thruster clusters as described in Reference 10. Each cluster of four emitters has a corresponding neutralizer and power processor. Four such clusters mounted on four corners of the spacecraft provide decoupled rotations and translations with some redundancy.

The last section of Table 2 summarizes the performance of the different technologies with respect to the attitude control requirements. The first row in this section lists the time required to complete a 90 degree slew for retargeting of the spacecraft assuming the thrust level listed with a moment arm of 7.5 cm and an inertia about the rotation axis of 50 kg m^2. The last two rows in this section list the quiescent deadband time in rotation and translation for the assumed minimum impulse bit, the same moment arm and inertia as assumed for the slews, and a spacecraft mass of 150 kg (the lower collector mass estimate is more conservative here). The deadbands listed, 1 arcminute in rotation and 1 cm in translation are peak to peak, with control achieved in a bang-bang mode to avoid any thruster firings during data collection if possible.

Results

As already mentioned, for a given spacecraft mass, the time required to complete the candidate observation plan is only a function of the thrust level. One characteristic of the observe-on-the-fly mode is that by limiting the relative spacecraft velocity to some value, as thrust is increased one simply shortens the time required in either the acceleration or deceleration portion of the transit. For all but the short, non-radial paths of the plane filling movements (Figure 3, F2D paths 4, 8, 12 etc.) the time spent coasting is a significant fraction of the total movement time. The result of this is that total movement time is a much weaker function of thrust level than was the case in the stop-and-stare mode with no coasting periods assumed in the previous trade study[3]. As seen in Table 2, the movement time varies by less than a month over the range of thrust levels considered but are within the four months allocated to this part of the mission.

As expected, the cold gas system has the largest propellant requirement. However, unlike the result in the previous trade study where only the minimum time (no coasting) stop-and-stare observation mode was considered, the propellant mass requirements are not inconsistent with the overall mass of the spacecraft. In fact the wet mass is the lowest of all three options considered. Some of the assumptions in the impulse calculations such as constant spacecraft mass for example will may also be excessively conservative for the cold gas option which has the largest propellant mass fraction. Also as expected, the cold gas system has the largest volume requirement because of the nitrogen storage tanks. These are already assumed to be pressurized to 4500 psia and further volume reduction is unlikely. Because the volume of each bus, assumed to be on the order of a meter on a side in order to accommodate the three spacecraft cluster on a Delta II, is already constrained this could be a significant issue.

The PPT mass is dominated by the dry mass of these devices which is in turn dominated by the capacitor which is the single most massive component in the thruster. The FEEP system, despite its high specific impulse had approximately the same dry mass as the PPT system. Like the PPT, its dry mass is dominated by a single component, in this case the power processor which has a mass of over 5 kg (and there is one for each cluster). The

	Cold Gas		PPT		FEEP	
Thrust (single unit) (mN)	0.35[a]	1.05[a]	0.35[b]	1.05[b]	0.35[c]	1.05[c]
Isp (seconds)	70	70	1150	1150	8600	8600
Minimum Impulse Bit (N sec)	2.0E-6	5.0E-6	5.0E-5	5.0E-5	1.0E-8	1.0E-08
Average Power (for couple) (W)	< 20W	< 20W	40	120	40	130
APERTURE FILLING[d]						
Time (months)	3.9	3.0	3.9	3.0	3.9	3.0
Impulse (N sec)	8920	9120	8920	9120	8920	9120
Propellant Mass (kg)	13.0	13.3	0.79	0.81	0.106	0.108
Propulsion System Dry Mass (kg)	7.0[e]	7.2[e]	30.0[f]	30.0[f]	29.2[g]	29.2[g]
Propulsion System Wet Mass (kg)	20.0	20.5	30.8	30.8	29.3	29.3
Propulsion System Volume (cm^3)	4.1E+4	4.2E+4	1.3E+4	1.3E+4	4900	4900
ATTITUDE CONTROL[h]						
Time for 90 deg Slew (minutes)	41	24	41	24	41	24
Time between Firings:						
1 arcmin Rotation (minutes)	1600	650	65	65	3.3E+5	3.2E+5
1 cm Translation (minutes)	1.3E+4	5000	500	500	2.5 E+6	2.5E+6

a Cold Gas Microthruster operating pressure regulated down to 1 psia and 5 psia for thrust levels of 0.35 and 1.05 mN respectively. Minimum Ibits correspond to valve on-times of 5 ms.
b. PPT operating at 0.5 and 1.5 Hz for thrust levels of 0.35 and 1.05 mN respectively with 40 J, discharge, 700 μN-sec pulses. Minimum Ibit corresponds to approximately 5 J discharge.
c. FEEP with 7cm slit operating at 3.2 and 8.8 mA emission current for thrust levels of 0.35 and 1.05 mN respectively. Power assumes emitter specific power of 60 W/mN and use of low work function thermionic cathode neutralizer with 0.20 W/mA.
d. For conservatism, aperture filling calculations assumed higher mass of combiner (250 kg) instead of lower collector mass (150 - 175 kg). Impulse listed includes allocation for cluster and formation flying mission phases (720 and 1300 Nsec respectively) as well as additional allocation for retargeting slews (1890 Nsec).
e. Assumes use of dual cylindrical composite wound tanks with 10 mil aluminum liner. (Approximate size each: 0.90m x 0.18m).
f. Assumes 6 units each with two fuel bar-electrode sets.
g. Assumes 4 quad clusters, each with 1 neutralizer and power processor.
h. Slew and deadband performance calculations assume Mass = 150 kg (for translation) and moment arm = 7.5 cm, inertia = 50 kg m^2 (for rotation)

Table 2. Propulsion System Candidate Performance Summary

European Space Agency is supporting development of lighter weight power processors for the FEEPs and it is likely this mass will be significantly reduced in the near future.

Reviewing the results in Table 2 for the attitude control functions, it is evident that for the assumed inertia and moment arm, a thrust level of 0.35 μN is inadequate to accomplish the 90 degree rotation in less than 30 minutes. The PPT and FEEP thrusters have the capability to operate at both thrust levels with a single design, so the low thrust, lower power operating point could be used for aperture filling and the higher thrust, higher power could be reserved for turning the spacecraft in a minimal time, perhaps with supplemental power from batteries if required. While the cold gas system does not have this flexibility, its power requirements are minimal and so it could be sized to operate at the higher thrust level all the time with negligible propellant mass penalty.

For deadband control, all systems have adequate margins in translation and rotation. In rotation, the cold gas and FEEPs have at least one order of magnitude margin in terms of quiescent time. The PPTs meet the 60 minute goal with the assumed inertia and moment arm but any changes in these parameters would require a close evaluation of the impact on this performance.

Conclusions of the Propulsion Trade

Given the assumptions made and described in the previous section, all three technologies could potentially fulfill all the mission requirements. Each presents a unique challenge to the spacecraft designer and all three technologies would require some degree of development work prior to use on DS-3. For the cold gas system, demonstrating the reliability of the specific micro-valve and pressure regulator chosen would be high priorities. Integration of the propellant tanks on (or in) the bus would also be an issue. For the FEEPs, demonstration of thruster cluster operation with a single neutralizer of the type described would be required. One advantage of a nine month mission time is that a full life test would be feasible. The power requirements of the FEEPs are slightly higher at the 1.05mN thrust level but still within the 150 W target. The PPT would have significant heritage from the EO-1 design (see next section) and the assumed performance is already within its demonstrated envelope. The single largest challenge for the PPT option is its incorporation into a overall spacecraft bus configuration. This is particularly difficult given the desire to have a single PPT thruster design and development activity and constraints due to deployable solar arrays, glint shields, and keep-out zones near the interferometer components. While somewhat more flexible because of their distributed structure and masses, the cold gas and FEEP options would also present significant challenges with respect to integrating into an overall bus configuration.

A major goal of the third New Millennium Program Deep Space Mission is to demonstrate technologies needed for precision formation flying on future missions. While the cold gas system can meet the requirements assumed in this work, it is likely that future spacecraft constellations would have mission times in excess of nine months with as yet undefined impulse requirements. For a cold gas system, the issue of gas leakage over time and low specific impulse is likely to be a greater factor for these missions than for DS-3. With respect to the FEEPs, one concern for the instrument is the use of liquid metal propellant in such close proximity to the siderostat surface on the collector spacecraft and the afocal telescopes on the combiner. Unlike the LISA mission where spacecraft separations are on the order of 10^5 km, the spacecraft separation could be as close as 100 m while thrusting in the optical line-of-sight of the other collector. Because of its electrical conductivity and chemical reactivity, cesium vapor deposition is considered even less desirable than Teflon® which is made up of carbon and fluorine.

The fact that all three options considered meet the requirements assumed in this study indicate there are multiple options, each with its own development and integration issues. A Request For Proposal (FRP) will be released in mid-July as the initial step in selecting a prime contractor for the three spacecraft buses. In responding to solicitation, respondents have the latitude to propose any propulsion system which they believe can satisfy the mission requirements set forth in the RFP and has the greatest potential to reduce risk and insure mission success. No final decision will be made by the Project on the propulsion subsystem selection until after the selection of a prime contractor for the spacecraft.

Given DS-3's importance as a formation flying technology demonstration mission, a PPT development activity begun as part of the NASA EO-1 Program has included planning and initial work to support a possible use of PPTs on DS-3. The next section describes areas identified for more focused study as part of this development plan.

DS-3 Pulsed Plasma Thruster

Spacecraft Interactions

As already mentioned, a significant challenge in using the PPTs on DS-3 will be identifying an acceptable thruster configuration on the spacecraft that simultaneously meets the disparate requirements of providing independent rotations and translations, sufficiently small moment arm to maintain adequate deadband quiescent times, and avoiding keep-out zones around the instrument all while maintaining significant EO-1 heritage and using a single thruster design for all locations.

Arrangement of components on the PPT itself to meet these demands present special challenges in themselves and are the subject of separate paper[6]. Beyond the purely configuration related issues of locating the PPTs on the bus is a broader area of electromagnetic and mechanical interactions which will have to be fully understood as part of any non-recurring development effort. These are briefly described in the following paragraphs.

As with any sensitive optical detector, the presence of stray light is a serious concern. Measures such as glint shields can be taken to reduce the probability that reflected sunlight off one of the other two spacecraft in the formation will saturate the CCD detector during observations of faint sources. However the unavoidable and intense flashes from PPT firings present a similar risk. Ideally, such concerns could be mitigated through purely operational means. The requirement of a 3600 second quiescent time during observation should accomplish this for most target sources. Even while observing on the fly, most of the time is spent coasting during which observations could be made. Because PPT emission energy spectra are concentrated in specific bands, it may also be possible to edit out selective parts of the collected spectrum should a firing occur during observation. Of course it may also be possible to take a sufficient number of data points so that any that include a thruster firing can simply be discarded. The question of whether the intensity of a flash could actually damage the detector has not been addressed yet and would depend on the power distribution of the emission among other factors.

The importance of mechanical disturbances from a PPT pulse again depend on the spectral energy content of the disturbance. In particular, 10 nm amplitude vibrations at frequencies above 50 - 100Hz may be problematic as they degrade the fringe visibility or intensity. Quantifying this risk is again a priority for the development effort and will require close collaboration between the PPT and spacecraft design teams to insure a suitable mechanical interface is developed.

The effect of the plume from the PPT on optical surfaces is a recurring concern and will eventually need to be addressed by first quantifying the extent to which particulates and condensed vapor are likely to deposit on sensitive surfaces from both the direct and back-flow fields. This deposition will be a function of the plane coverage strategy as this directly drives both the amount of material expelled (impulse) and the extent to which the firing is in the line of sight of another spacecraft. Secondly, once the extent of deposition has been quantified, its effect on the visibility budget will have to be characterized to see if it is in fact a problem. While it is unlikely these issues can be resolved in a completely definitive way before the mission. The necessary level of confidence should be obtainable through a combination of carefully performed designed tests in terrestrial vacuum chambers in conjunction with numerical modelling. This work has already begun as part of the EO-1 PPT development effort and has been led by studies at the NASA Lewis Research Center (LeRC)[13] and Worcester Polytechnic Institute (WPI)[14,15,16]. WPI has been leading the effort to model the levels of deposition for different U-V plane coverage strategies[17].

A related but distinct issue has to do with the interaction of the ionized plumes with different electromagnetic sources on the spacecraft which is in addition to the usual EMI/EMC issues. In particular, four separate electromagnetic signals could potentially be attenuated, reflected or otherwise corrupted through interaction with the plume. These are the inter-spacecraft communication system (S Band), the AFF transponder-receiver (30 GHz), spacecraft to earth downlink (X or Ka Band) and the laser metrology beam (1.3 micron). Significant absorption of any of these signals could pose problems and multi-path effects of the AFF signal due to reflection from the plasma could be particularly problematic if not well understood.

Development and Test Plan

The PPT technology was one of the new technologies submitted for consideration for DS-3 as part of the New Millennium Program Modular and Multifunctional Systems Integrated Product Development Team (MAMS-IPDT). A ground rule of the PPT configuration trades underway at Primex Aerospace Company (PAC) for the DS-3 mission has been to maximize the heritage from the EO-1 PPT design. Several critical elements of the design are assumed to be identical - including the energy storage capacitor and the discharge initiation sparkplug. These are critical components to the life of the PPT system. In addition, the improvements to the main charge converter topology that enabled the high power density of the EO-1 charging electronics will be employed on DS-3. By taking this approach, non-recurring engineering costs are reduced and the schedule risk of new component development is minimized.

The EO-1 configuration is a single thrust axis with two opposing thrusters. The electronics and the energy storage capacitor are housed in a single chassis, which also provides the major structural backbone of the PPT system. The capacitor charging supply and the discharge initiation (DI) circuit are contained on a single circuit card assembly (CCA). The DI circuit selects which of the two thrusters are fired by firing the appropriate sparkplug. If additional thrusters are added, i.e. to provide three axis capability, an additional DI circuit must be added.

The major change for the DS-3 mission is the configuration of the thrusters. The two opposing thrusters on the EO-1 design limit the amount of Teflon® propellant that can be stored for each of the two thrusters. Also, it appears from the trade studies now underway[6] that it is desirable to use a single PPT module (electronics, capacitor, etc.) to provide thrust in more than one axis. The main design driver is to achieve the desired translation and rotation of the spacecraft with as few PPT modules as possible. The design philosophy being employed in the trade study therefore is to make maximum use of existing elements of the EO-1 design and explore methods of meeting the design goal of minimizing PPT modules. Table 3 summarizes the overall PPT performance goals for the DS-3 mission.

	Performance Target
Thrust[a]	1.4 mN
Ibit Range	50 -700 μN-s
Min. Ibit	≤ 50 μN-s
Isp	> 1000 sec
Power[a]	up to 80 W
No of Shots @ > 40 J	$13 \cdot 10^6$
No of Shots @ < 10 J	$1.0 \cdot 10^5$
Dry Mass[b]	5 kg

a. Maximum sustained per unit (not couple)
b. Mass based on a configuration with two fuel bars

Table 3. DS-3 PPT Performance Targets

The DS-3 pre-project status will be reviewed for a likely new start at the beginning of FY99. Selection of a prime contractor for the three spacecraft buses should be formally announced toward the end of calendar 1998. The current project schedule calls for a Preliminary Design Review in 9/99, a Critical Design Review in 10/00, beginning of Assembly Test and Launch Operations (ATLO) is scheduled for 7/01 and launch in 12/02..

Planning activities between JPL, LeRC and PAC have been underway for the past year to help set in place a technology development activity to support the DS-3 Project should the PPT be selected as the propulsion system for the flight. Because the specific direction any development activity would take is strongly configuration dependent, this planning will eventually include the prime contractor which would design the spacecraft bus. Regardless of the final configuration however, certain issues which have been described in this paper would be (and in some cases already have been for EO-1) included in the non-recurring technology development. These include experimental and numerical characterization of the plume with respect to potential contamination, work which has been performed jointly with NASA LeRC and WPI. Further tests at NASA LeRC would evaluate performance, life, EMI and optical characterization of the discharge. In addition, PAC, as a member of the MAMS-IPDT is performing it own internally funded configuration trades for the PPT in support of the DS-3 Project.

Conclusion

An observation strategy for the New Millennium Interferometer (DS-3) Mission was described which takes advantage of the relaxed constraint on relative spacecraft motion during observations. The propulsion requirements for this new observation strategy as well as for precision attitude control were described. The need to control the spacecraft angular rotation rate during movement along non-radial paths may limit the ability to satisfy the quiescent time requirements during these moves. This may require some number of stop-and-stare movements if data is to be collected along these paths.

Three propulsion system options were considered in the trade; cold gas, PPTs and FEEPs. As a result of the lower total impulse requirements for this observation plan, and to the level of detail considered in the analysis, all three options were viable candidates for the aperture filling role. In terms of attitude control all three candidates satisfied the deadband control requirements. With the higher of the two thrust levels considered all three options could perform the required retargeting maneuver within the allowed time and power constraints. Finally, several PPT-spacecraft interaction issues were briefly described along with an outline of the technology development plan already underway.

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