Propulsion Options for the LISA Mission

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The LISA mission is a constellation of three spacecraft operating at 1 AU from the Sun in a position trailing the Earth. After launch, a propulsion module provides the ΔV necessary to reach this operational orbit, and separates from the spacecraft. A second propulsion system integrated with the spacecraft maintains the operational orbit and reduces non-gravitational disturbances on the instruments. Both chemical and electrical propulsion systems were considered for the propulsion module, and this trade is presented to show the possible benefits of an EP system. Several options for the orbit maintenance and disturbance reduction system are also briefly discussed, along with several important requirements that suggest the use of a FEEP thruster system.

Nomenclature and Acronyms

ACS	=	Attitude Control System
AU	=	astronomical unit
C_3	=	excess launch energy
DISCOS	=	Disturbance Compensation System
DRS	=	Disturbance Reduction System
EP	=	electric propulsion
FEEP	=	field emission electric propulsion
FTR	=	Final Technical Report
FY	=	fiscal year
GEM	=	graphite epoxy motor
GSFC	=	Goddard Space Flight Center
Hz	=	hertz
kg	=	kilograms
kW	=	kilowatts
LISA	=	Laser Interferometer Space Antenna
m_p	=	propellant mass
$m_{s/c}$	=	mass of the spacecraft
O/F	=	oxidizer/fuel ratio
TR	=	Technology Readiness (Level)
TRIP	=	Technology Report and Implementation Plan

I. Introduction

The primary goal of the LISA mission will be to detect gravity waves from galactic and extra-galactic sources in a frequency range from 10^{-4} to 10^{-1} Hz. These detections will enable the following science goals to be met:

- Determination of the role of massive black holes in galaxy evolution,
- Precision testing of the theory of general relativity,
- Determination of the population of ultra-compact binaries within our galaxy, and
- Detection of gravitational waves from the early Universe.

The gravity waves will be detected by measuring the change in distance between proof masses in three satellites as a function of time by means of a laser interferometer. The three satellites are launched from Earth on the same Delta IV launch vehicle, and transfer via onboard propulsion to a final orbit located at 1 AU from the Sun at an angle of 20° behind the Earth. The inclination of the operational orbits of each satellite is offset to one another by

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~0.7° such that the three spacecraft orbit the Sun in formation, roughly as an equilateral triangle with 5 million kilometers between each satellite. Further details on the instruments and operational orbit design are available in the Technology Readiness and Implementation Plan (TRIP).[1]

Two propulsion systems are used by each of the LISA spacecraft. There is a small propulsion system onboard the spacecraft, which is part of the Disturbance Reduction System, or DRS, that is used to measure and compensate for disturbances on the spacecraft, such as solar radiation pressure, when scientific measurements are performed. A separable propulsion module will also be used to transfer the spacecraft to the final operational orbit after launch.

In the preparation of the TRIP report, an initial trade study was conducted to determine whether a chemical or electrical propulsion system should be used for the propulsion module. This trade study compared a chemical propulsion system with the Electric Propulsion (EP) system from the Final Technical Report of April 2000.[2] This trade is presented here along with further work that has been done to consider possible additional options for the EP system. Some discussion of possible alternative systems that could be used for the DRS propulsion subsystem is also included.

II. Baseline Chemical Propulsion System

Prior to the TRIP report, an electric propulsion system was required to fit within the volume constraints of a Delta 7925H fairing. Volume and mass growth, as well as launch vehicle availability constraints on the LISA mission caused a move to be made to a Delta-IV Medium+ launch vehicle. This increase in launch vehicle capability allowed a chemical propulsion system to be considered. A chemical propulsion baseline was established in the TRIP report following a number of design and configuration trades.

Orbital optimization was done by Barden *et al.* to determine the ΔV budgets on the 3 propulsion modules.[1] This work showed that a ΔV of 1.22 km/s was necessary for all three spacecraft to reach their final orbits. One possibility that was encountered as part of the study is the large reduction of ΔV required to reach an operational orbit that is located 20° in front of the Earth instead of behind (1.03 km/s instead of 1.22 km/s).

Based on this 1.22 km/s ΔV requirement, a monopropellant system was rejected because it did not provide sufficient mass margin for launch. The baseline launch vehicle for the chemical propulsion baseline was the Delta IV 4240, which can inject approximately 4000 kg into the desired initial orbit. The baseline dual-mode spacecraft had a total launch mass of 3596 kg. The monopropellant propulsion system would require 36% more propellant, reducing the 11% launch mass margin to 1%. A dual mode system was therefore designed to provide the required amounts of ΔV and attitude control during the orbit transfer using less propellant. The dual mode system uses a bipropellant engine to provide the ΔV and uses the common hydrazine supply for the attitude control system (ACS). The layout of the propulsion modules is governed primarily by the dimensions of the propellant tanks. A schematic of the layout is shown here in Figure 1.

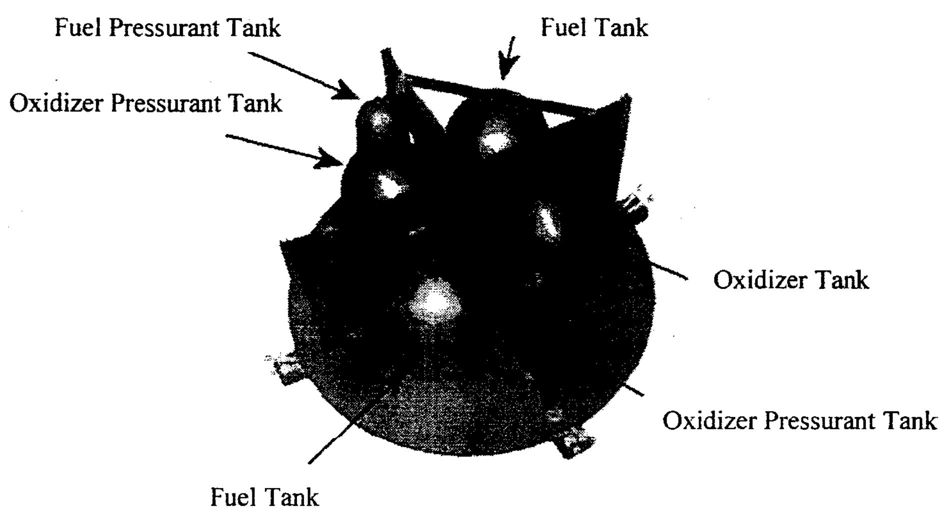


Figure 1. Schematic of the baseline dual mode propulsion system for LISA

The system baselines two redundant 5 lbf (22 N) N_2O_4/hydrazine engines with titanium PMD tanks and a helium pressurization system. Two tanks are used for both the oxidizer and the fuel, and two separate pressurant tanks are used to prevent fuel and oxidizer back leakage into the same tank. The total dry mass of the propulsion system is 95 kg with a 30% margin. Over 50% of this fraction represents the tank mass. The dry mass breakdown of the propulsion system is shown here in Table 1. The wet mass includes 179 kg of oxidizer, 258 kg of hydrazine, and approximately 1 kg of gaseous helium.

Table 1. Mass breakdown of the dry bipropellant propulsion system.

Component	Qty	Description	Mass [kg] each	Mass [kg] total
Hydrazine fuel tank	2	PMD titanium tank	8.1	16.2
Oxidizer fuel tank	2	PMD titanium tank	5.67	11.34
GHe ullage/pressurant tank for fuel	1	Composite Overwrapped Pressure Vessel - 2200 psi	6.35	6.35
GHe ullage/pressurant tank for oxidizer	1	Spherical titanium tank - 4500 psi operating pressure	3.36	3.36
Biprop (N_2H_4-NTO) thrusters	2	22 N (5 lbf) class thruster	2	4
Flow components				
Fill/Drain valve	4		0.82	3.28
Check valve	2		0.57	1.14
Isolation valve	12		0.57	6.84
Tubing	1		8	8
Filter	4		0.25	1
Regulator	2		3	6
Pressure transducer	9		0.1	0.9
Brackets/Harnessing	1		4.1	4.1
		Total Propulsion System Dry Mass:		**72.54**

The propellant mass is designed to provide the baseline chemical ΔV shown in Table 2 for all three spacecraft. A total of three maneuvers were required for all three spacecraft – an Earth departure maneuver followed a five month coast, a mid-course correction burn followed by another 5 month coast, and a final orbit insertion before reaching the operational orbit. The wet mass required to complete these maneuvers, including margin, is within the capability of the Delta IV Medium+ (with 2 strap-on motors and a transfer C_3 of 0.65 km^2/s^2, the launch capacity is approximately 4000 kg). This ΔV budget was to reach a final orbit that was 20 degrees behind the Sun-Earth line. The dry mass of the flight unit considered to calculate these propellant margins is 520 kg.

Table 2. Maneuver and mass budgets.

Maneuvers	ΔV	I_{sp}	$M_{s/c}$	M_p with 30% margin	O/F Ratio	$M_{p\text{-}ox}$	$M_{p\text{-}fuel}$
Earth Departure Maneuver	407	308	957.19	156.97	0.85	72.12	84.85
ACS (10% of the maneuver ΔV)	40.7	223	800.22	19.19	0.00	0.00	19.19
Mid-Course Correction	407	308	781.03	128.08	0.85	58.85	69.23
ACS (10% of the maneuver ΔV)	40.7	223	652.95	15.66	0.00	0.00	15.66
Final Orbit Insertion	407	308	637.29	104.51	0.85	48.02	56.49
ACS (10% of the maneuver ΔV)	40.7	223	532.78	12.78	0.00	0.00	12.78
Totals	1343.1		520.00	437.19		178.98	258.20

The attitude control system (ACS) is run off the same hydrazine supply as the bi-prop engines. The propulsion module will therefore use 8 small hydrazine thrusters. The baseline for the hydrazine attitude control system will be 4.45 N (1 lbf) thrusters (a mass of 0.5 kg each). The total dry mass of the ACS system in each spacecraft is only 18 kg, which includes thrusters, isolation valves, fill and drain valves, tubing, brackets, and harnessing. The hydrazine required for ACS is budgeted under the main propulsion system propellant in Table 2.

III. The Electric Propulsion System

The EP propulsion module baseline described in the FTR was limited to a height of 0.203 m, with no margin.[2] This severely limited the size of the propulsion tanks and components. All of the components were supported around the rim of a thin flat plate, as shown in Figure 2. The mounting mechanism for the ion thrusters was complicated by the need to mount to the circumference of the circle and simultaneously apply a thrust through the center of mass of the vehicle. In order to maintain the location of the center of mass, the Xenon tanks were located far from the ion thrusters. In addition, the tanks were completely exposed, which greatly complicated the thermal control of the propellants inside of the tanks.

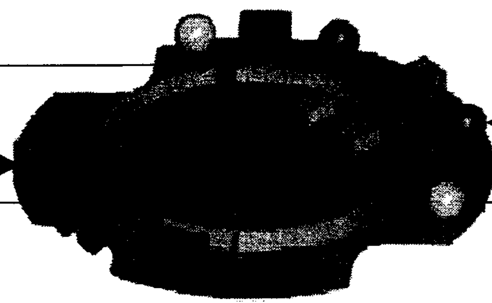


Figure 2. Internal detail of the propulsion module layout in the FTR.

The baseline thrusters in the FTR were the UK-10/T5, the RIT10-Evolution, and the RMT thruster. The first is an electrostatic ion thruster, and the second two are radio-frequency ion thrusters.[2,3,4,5] All three of the thrusters are approximately 10 cm in diameter, and require a neutralizer to cancel the charge of the emitted fuel.

The performance of the first two thrusters is very similar. However, the third thruster, the RMT thruster, does not meet the arbitrary 18 mN thrust requirement given in the FTR. In addition, only European electric thrusters were considered in the FTR. There are in fact several other thrusters that meet this arbitrary thrust requirement and that fit within the available volume. Most of these thrusters do not currently meet the other driving requirement in the FTR for thruster lifetime to be at least 10,100 hours, except one, the XIPS 13 cm thruster. The transfer times with these propulsion systems are on the order of 450 days (15 months).[2]

The baseline for the EP system includes redundant pressure regulators, flow control units, and power units. The current breakdown of the masses of the components of the EP system is shown here in Table 3. The baseline transfer orbit also requires 20 kg of xenon propellant. A schematic of the EP system is shown in Figure 3.

Table 3. Electric Propulsion (EP) system masses.

Xenon Tank Mass	6.4 kg
Xenon Mass	20.0 kg
Ion Thruster Assembly	4.6 kg
Pressure Regulator	0.6 kg
Flow Control Unit	2.0 kg
Ion Thruster Power Unit	9.8 kg
Tubing, valves, harness	2.6 kg
TOTAL	46.0

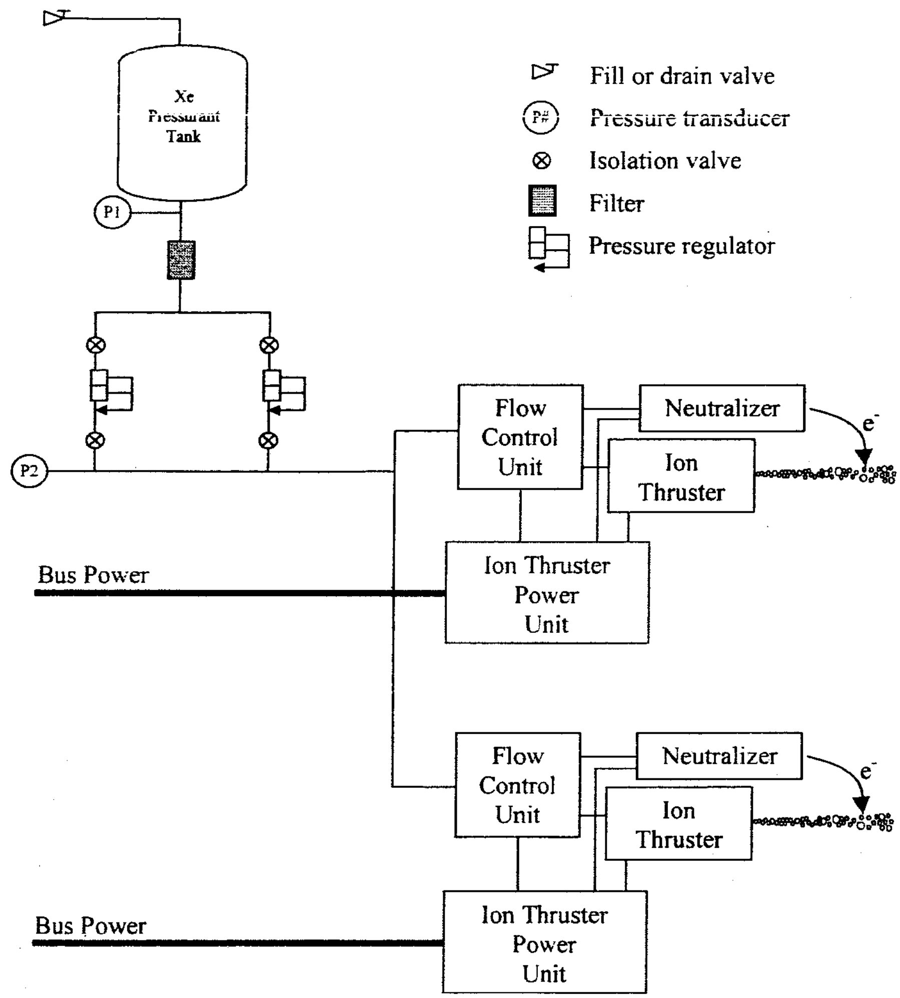


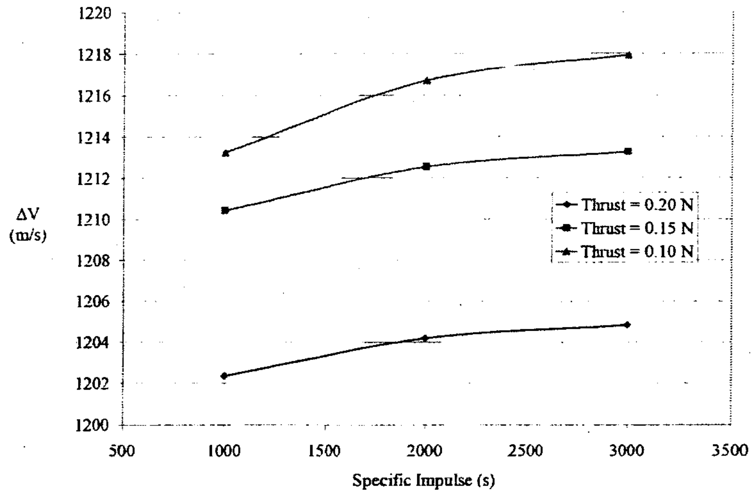
Figure 3. Schematic of the Electric Propulsion system.

The baseline hydrazine attitude control system in the FTR was based primarily on the LEO ONE reaction control system. It uses a minimal approach to provide three-axis control with four small hydrazine thrusters. However, this assumes that there is no additional requirement not to produce parasitic thrusts along the velocity vector. Four thrusters can be used to provide attitude control torques, but this produces undesirable thrust along the spacecraft velocity vector. The propulsion module was therefore designed to use 8 small 1 N hydrazine ACS thrusters.

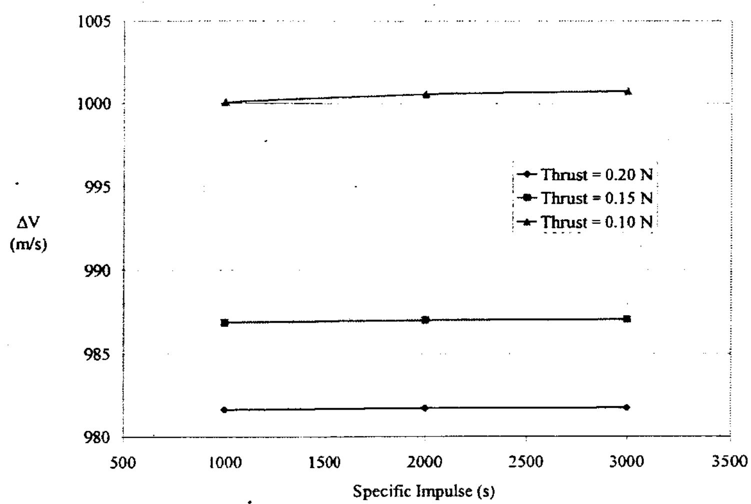
Because a low thrust trajectory is employed, the thrust levels required by the attitude control system are generally small. The baseline for the hydrazine attitude control system will be 1 N thrusters (a mass of 0.5 kg each) with a blowdown pressurization system. The hydrazine mass required for ACS is 4 kg. Each propulsion module contains two small hydrazine tanks, both of which are 9.4 in (24 cm) in diameter, and a small nitrogen pressurant tank. In addition to the tanks, there are approximately 2 kg of components (valves, flow regulators, pressure transducers, and tubing).

IV. Post-TRIP Trade Study

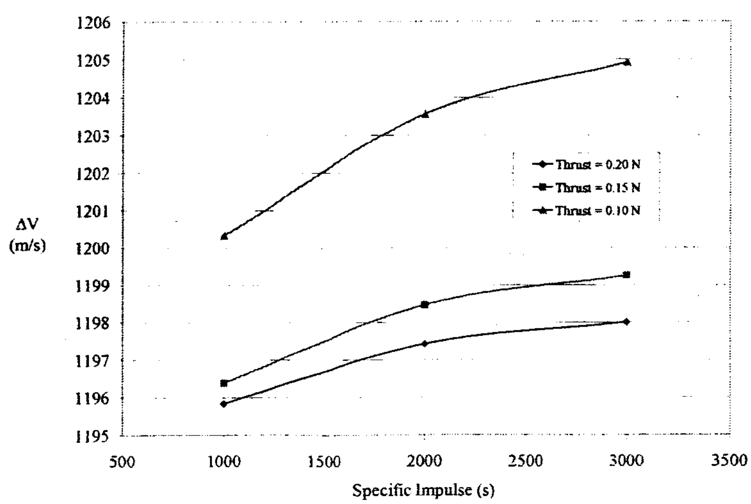
A new study was conducted after the TRIP report to examine the effect of specific impulse and thrust level on the requirements for the electric propulsion system. Trajectory simulations were conducted in SWINGBY™ at three different thrust levels and three different specific impulses for all three spacecraft. The maneuvers were partially-optimized as a three-burn system, with coast periods between the burns. Final conditions for the maneuvers were obtained from Hughes *et al.*[6] The total ΔVs required for all three spacecraft are shown in Figure 4.



(a) ΔV for the first spacecraft



(b) ΔV for the second spacecraft



(c) ΔV for the third spacecraft

Figure 4. ΔV requirements for all three of the LISA spacecraft

A database of over 100 currently available electric propulsion thrusters was queried to determine whether they could perform the required maneuvers. Two main criteria were considered: the demonstrated total impulse (or propellant throughput, or flight time), and the amount of power required to obtain the given performance parameters. The total impulse requirements for the analysis cases are shown in Figure 5. The total power is assumed to be limited to the amount available from a body-mounted array on the top of the spacecraft and a small deployable array. The body-mounted array is limited to the available area on the top of the spacecraft surface, an area of approximately 5 m^2. The current baseline for the power subsystem does not use up all of the area on the top of the spacecraft surface, and uses a gallium arsenide array with an area of 3.45 m^2. The deployable array has a total area of 3m^2. This is an addition of nearly 23 kg. For multi-junction gallium arsenide arrays with an efficiency of 22%, this would provide approximately 2.40 kW of power.

These two selection criteria, the power availability and the total impulse, reduce the number of available thrusters to seven thrusters, 3 ion engines and 4 Hall thrusters. These thrusters are listed here in Table 4 along with the ranges of performance values and system masses.

Table 4. Thrusters selected for the EP transfer

Device Name	**Impulse**		**Thrust**		**Power to PPU**		**Thruster Mass**	**PPU Mass**
-	Low (s)	High (s)	Low (N)	High (N)	Low (kW)	High (kW)	kg	kg
SPT-140 Hall	1700	1775	0.135	0.3	2	5	6.5	12
PPS-1350 Hall / Smart 1	1060	1600	0.03	0.092	0.462	1.5	5.3	9.5
BPT-2000 Hall	1765	1765	0.123	0.123	1.2	2.7	5.2	12.5
BPT-4000 Hall	1950	1950	0.2	0.2	2	6	7.5	12.5
NSTAR	1961	3035	0.0208	0.0753	2.3	2.5	8.33	13.3
NEXT	2300	4125	0.0496	0.238	1.11	10	28.8	12
XIPS 25	2800	3800	0.0635	0.165	1.4	4.5	6	11

Device Name	**TR Level**	**Total Impulse**	**Demonstrated/Design Flight Time**	**Demonstrated Fuel Throughput**
-	#	N.s	hrs	kg
SPT-140 Hall	7	6,000,000	7,200	497
PPS-1350 Hall / Smart 1	9	2,000,000	-	-
BPT-2000 Hall	8	2,600,000	6,000	153.3
BPT-4000 Hall	7	5,800,000	6,000	304.56
NSTAR	9	-	15,000	140
NEXT	5	-	18,000	300
XIPS 25	9	-	4,350	-

The optimal performance values for each thruster were used to calculate the actual propellant mass required and thruster on-times by correlations with results shown in Figure 5. The selected thruster performance parameters are shown in Table 5. These performance characteristics were used with the tables of data obtained from the flight dynamics simulations to determine the ΔV, and hence the amount of propellant required for each thruster and spacecraft combination. The ΔV results are shown in Table 5 for the first spacecraft. All three of the spacecraft must be identical in configuration, so the propellant capacity was determined from the largest ΔV results for the first spacecraft. The subsystem mass includes two EP thrusters for redundancy, as well as the other flow control and power production hardware.

Table 5. Performance values for the selected thrusters

Device Name	Impulse	Thrust	Power	ΔV	Propellant Mass	Dry System Mass
-	s	N	kW	m/s	kg	kg
SPT-140 Hall	1710	0.157	2.4	1212.6	34.9	51.0
PPS-1350 Hall / Smart 1	1600	0.092	1.5	1203.0	36.9	46.1
BPT-2000 Hall	1765	0.123	2.4	1208.0	33.7	48.9
BPT-4000 Hall	1950	0.2	2.4	1212.5	30.7	53.5
NSTAR	2498	0.0961	2.4	1204.2	24.0	56.0
NEXT	2980	0.102	2.4	1205.2	20.2	95.6
XIPS 25	3000	0.088	2.4	1203.0	20.0	49.0

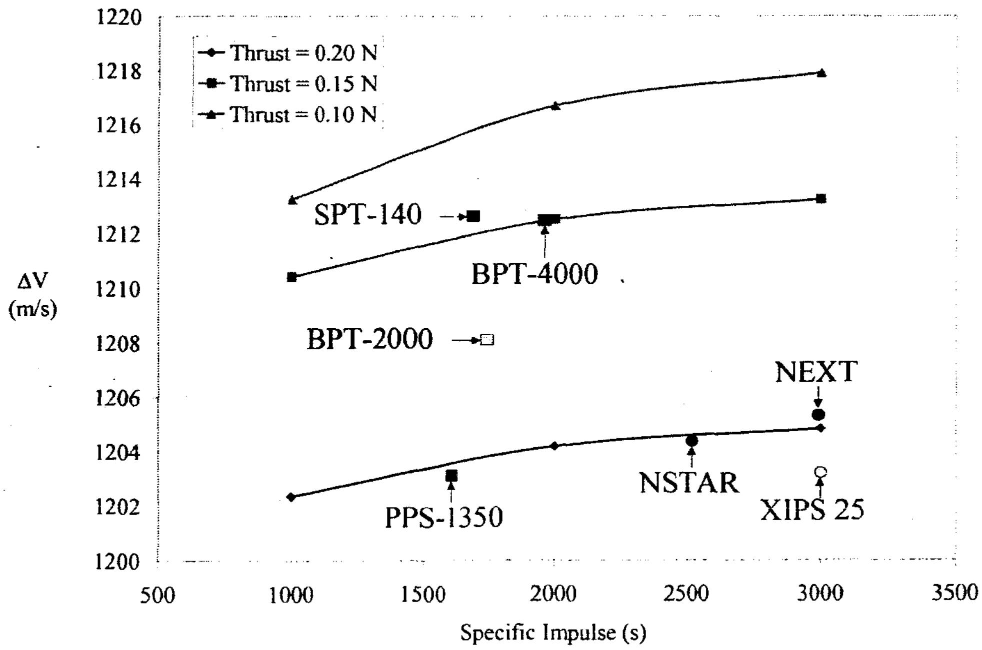


Figure 5. Maximum ΔV requirements used to size the propellant mass for all of the thruster systems.

V. Comparison

Although these propellant masses are relatively close to the masses required for the smaller-thrust systems described in the FTR, the transit times are significantly shorter. All three of the spacecraft took approximately 10 months to achieve their final operational orbits with the seven propulsion systems selected by this trade (thrust from 0.092 – 0.157N), compared to over 15 months for the systems in the FTR (thrust = 0.018 N). The transfer times for the dual-mode propulsion system were approximately 10-11 months. This transfer time is primarily determined by the durations of the burn phases of the trajectories, which are considerably larger for the 18 mN EP system.

The main benefit in using an EP propulsion system for the propulsion module is the drastic decrease in launch vehicle mass. The dual-mode baseline has a launch mass of 3596 kg, which includes a 30% reserve. The plan is for this baseline design to launch on the Delta IV 4240. With an electric propulsion system, the launch mass is reduced to 2559 kg for the heaviest possible EP system (the NEXT engine), including a 30% mass reserve. All of the launch masses are shown in Table 6. The launch mass margin is also shown in Table 6 for a launch on a Delta IV Medium 4040.

Table 6. Total launch masses, including fuel and propulsion systems for three EP-based spacecraft.

Device Name	Propellant Mass	Dry System Mass	Dry Mass Margin	Total Launch Mass	Launch Mass Margin
-	kg	kg	%	kg	%
SPT-140 Hall	34.9	51.0	30%	2429.1	9.7
PPS-1350 Hall / Smart 1	36.9	46.1	30%	2416.1	10.2
BPT-2000 Hall	33.7	48.9	30%	2417.4	10.1
BPT-4000 Hall	30.7	53.5	30%	2426.4	9.8
NSTAR	24.0	56.0	30%	2415.8	10.2
NEXT	20.2	95.6	30%	2559.0	4.8
XIPS 25	20.0	49.0	30%	2376.8	11.6

This reduction in launch mass allows a smaller launch vehicle to be used (the Delta IV 4040), with tremendous cost savings. The Delta IV 4042 is essentially the same launch vehicle as the Delta IV 4040 with two GEM solid motors strapped on.

In addition, the cost of the dual-mode propulsion system for one spacecraft has been estimated at $14.6M FY2004. The cost of the dual mode system was obtained from the GSFC Multi-variable Instrument Cost Model.[2] The cost of the EP system is a strong function of the development status of the thrusters and flight heritage. Using a cost model developed previously[7], the costs of the seven EP systems were calculated as shown in Table 7. These costs reflect flight programmatics and personnel (assuming an in-house NASA program), as well as hardware for one spacecraft. The cost estimates do not budget for any development of the thrusters, nor do they account for any changes to the thrusters or flight systems beyond what has flown in past programs. The calculations are heavily weighted on the technology readiness level and the mass of the system, and the calculations for the programmatics assume a six-year design, development, and build schedule. This six-year schedule corresponds to the schedule presented in the TRIP report, but may not represent the current schedule.

Table 7. Estimated programmatic and hardware costs for the seven EP propulsion systems

Device Name	Cost
-	$
SPT-140 Hall	13.7
PPS-1350 Hall / Smart 1	10.2
BPT-2000 Hall	12.1
BPT-4000 Hall	13.8
NSTAR	11.2
NEXT	17.1
XIPS 25	10.4

The system costs shown here for the EP system are all comparable to the costs estimated for the dual-mode propulsion system. The level of uncertainty for all of these numbers is quite high and the cost of the dual mode system cannot be compared directly with the costs of the EP systems because two different cost models were used with different weighting factors. However, all of the cost estimates are significantly less than the change in the cost of the launch vehicle.

VI. The DRS

The micronewton thrusters in the DRS are used in a low-bandwidth feedback loop controller to center the spacecraft around one of the free-floating proof masses. The main sources of environmental disturbances that cause the spacecraft to move are the solar wind, radiation pressure and gravity gradients. Although the current baseline Field Emission Electric Propulsion (FEEP) thrusters appear to be the most attractive solution, other technologies must be considered. Some of the requirements that force the baseline towards FEEPs are presented here. The primary requirements on the microthrusters in the DRS are listed here in Table 8.

Table 8. Primary requirements on the DRS thrusters

Requirement	Values
Thrust Range	100 μN - 0.1 μN
Nominal operating thrust level	20 μN
Thrust resolution	0.1 μN
Thrust noise	<0.1 μN/Hz^0.5
Operate continuously	5 yr
Total impulse	> 3000 Ns
Exhaust contamination on the spacecraft	None
Magnetic field	No transient fields
Change in self-gravity field at the proof mass	Minimal change allowed during the mission
Neutralization	No residual charging of the spacecraft
Latency	<1 ms
Specific impulse	>1000 s
Calibration requirements	Minimal calibration required after 1 year
Minimum impulse bit	<1 μN
Thermal requirements	0-50 C

The requirement to minimize the change in the self-gravity field (the gravity field created by the mass of the spacecraft surrounding the proof mass) at the proof mass allows us to derive a secondary requirement for the specific impulse of the thrusters. There are a total of 6 thrusters placed optimally around the spacecraft to counteract the disturbance forces. Each FEEP thruster will require ~50 grams of propellant, assuming an average specific impulse of 6,000 seconds throughout the 5-year mission and a total impulse of 3000 Ns per thruster. For a colloidal thruster with a mission average specific impulse of 1000 seconds, the propellant requirement is only 0.3 kg per thruster.

The propellant mass per thruster scales linearly with the specific impulse. In order to minimize the change in the self-gravity field, the specific impulse must be maximized. For example, a cold helium gas system could theoretically produce approximately 180 seconds of specific impulse, which would require approximately 1.25 kg of gas (assuming a regulated system), which is 25 times more propellant than the FEEP system – a significant change in the self-gravity field. It has been shown previously for the DISCOS system on the TRIAD 1 satellite that the self-gravity can be the single greatest source of disturbance on a drag-free spacecraft.[8]

In addition, there are only limited resources available for the microthruster. The power currently budgeted for the DRS propulsion subsystem (including margin) is 20 W. Some interest has been expressed in using this power to heat a "cold gas" system instead of the FEEP system, but this small amount of power does not significantly increase the specific impulse (approximately 190 seconds for a mass flow rate of 0.05 mg/s). In addition, the thrust noise induced by an actuation valve in a pulsed thrust modulation system is considerable, significantly above the thrust noise level requirement for LISA. The cold gas microthruster on each ST5 spacecraft has a thrust noise of approximately 2%. This noise level is well above the maximum thrust noise requirement for LISA.

VII. Conclusion

This detailed study on the trade between a chemical propulsion and an electric propulsion system for the three LISA spacecraft showed a potential launch mass reduction of between 1000 kg and 1200 kg compared to the baseline chemical propulsion design. This mass reduction specifically allows a smaller launch vehicle to be used. The most launch mass margin is achieved with the XIPS 25 thruster. The estimated cost of this system is also the lowest, but this must be verified with the vendor, and the programmatic requirements should be modeled with greater fidelity. The XIPS 25 is as well qualified as the baseline dual mode propulsion system design, and no objections based on risk can be levied on this thruster.

In addition to this recommended change in the baseline for the propulsion module, the FEEP thruster is recommended as the baseline for the DRS thruster. Substantial risks remain for the FEEP technology, specifically that the lifetime has not yet been demonstrated, but it is one of the two possible technologies that can fulfill all of the requirements on the DRS microthruster system. The other possible technology, the colloidal thruster, has the same technical risks, and does not have the same performance as the FEEP thrusters. Further excellent discussion of the

DRS microthruster requirements and possible technologies, as well as the plans to mitigate the risks for the principal candidate technologies are given in Ziemer *et al.*[9]

Acknowledgments

Optimization of the chemical propulsion trajectory was performed by Brian Barden of JPL.

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(Last Update: January 7, 2002)

1. **Section I:** Check whether the document is **modified or original**; enter the **Title, Author(s), RTOP# (civil servants) or Contract# (contractors); check the type(s) of publication**; enter the **document details**. Contractors list your contracting company as the **Performing Organization**. Detailed instructions: List the **Author(s) in the order listed in the document**, in the format **Lastname, Firstname (employer)**. The Ames employee submitting the 1676 form should **give your full phone number instead of your employer**. Give the **document details** (where published/presented, as space permits) in the box next to the black triangle/arrow. If Web publication is a possibility, also check the "URL" box, and give the URL or website in the document details box.

2. **Section II: Check "Unclassified"**. Note: If your document is *not* unclassified, check the appropriate box; your document must have limited distribution and be given a thorough export control review.

3. **Section III: Check "Publicly Available"**, and **complete the ARC1676A** form. Note: If your document does *not* qualify to be publicly available, do not check this box; your document must have limited distribution and be given a thorough export control review.

4. **Print the NF1676 form.**

5. **Section VI: Sign and date** the form. Check the **"Does NOT"** box. Note: If you did not check both Unclassified and Publicly Available on Page 1, check the "DOES" box, meaning that document distribution is limited and/or export controlled.

6. **Attach a copy of your document** to the form. [To conserve paper, please use duplex printing. Talk slides should be printed 2 or 4 to a page whenever possible.]

7. **Route the "1676 package" to get all indicated approvals.** Wait for final approval before releasing your document.

Please report form changes to me (Bryan Biegel, bbiegel@mail.arc.nasa.gov), so that I can update the online form at:
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