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THE USE OF ELECTRODYNAMIC TETHERS FOR ORBIT MAINTENANCE AND DEORBIT OF LARGE SPACECRAFT: A TRADE STUDY OF THE NASA GLAST MISSION

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ABSTRACT

The use of electrodynamic tethers (EDTs) to support such mission needs as deorbit, drag make-up, and inclination change are evaluated for a large, 4000-5000 kg class, spacecraft in low Earth orbit for astronomical applications. We specifically evaluate the use of an EDT for NASA's planned GLAST mission. GLAST is a next generation high energy gamma-ray observatory designed for making observations of celestial gainma-ray sources in the energy band extending from 20 MeV to more than 300 GeV and is part of NASA's Office of Space and Science Strategic Plan, with launch anticipated in ,2006. The 4500-kg GLAST observatory is planned to be launched into a 28.5' inclination orbit from Kennedy Space Center by a Delta 2920-10H for a 5-year mission with a goal of 10 years. End of mission deorbit of the GLAST spacecraft using an EDT is assessed against a hydrazine based approach and is found to require approximately a third of the mass. The use of a drag make-up EDT is also evaluated as a means to place the spacecraft at a lower altitude (400 vs. 550 km) to be below more of the negative effects of the South Atlantic Anomoaly (SAA) while assuring the required orbit lifetime. Chemical and electric propulsion alternatives for drag make-up are also evaluated. It is found that the EDT is the lowest mass approach.

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requirements). At the end of mission life the spacecraft shall have an 85% probability of successful controlled re-entry and safe ocean disposal compliant with NSS 1740.14. The baseline mission commensurate with the ten year mission life goal and launch from the Kennedy Space Center has an orbit of approximately 28.5" inclination and 550 km altitude (0.01% *initial* orbit eccentricity). The baseline data rate for the LAT is 300 kbps that will be stored between downlinks to low latitude ground stations in an on-board solid state recorder.

To reduce background from charged particles in the space environment and to reduce observing time lost in the South Atlantic Anomaly (SAA), the desired orbit should be as low in altitude and inclination as is possible. A 400 km orbit would reduce the time lost in the SAA from that experienced at the nominal **550** *km by –40%.*

The GLAST system shall acquire science data in 3 basic observation modes: (1) sky survey mode, (2) pointed observation mode, and (3) repointed observation mode. Specific pointing requirements are:

- The +Z axis of the observatory shall be commanded to point in a direction defined with respect to the orbit fixed frame.
- The rocking angle commands may be piecewise constant, or may vary slowly with time. Rocking angle offsets of the observatory Z-axis shall range from –60" to +6°.
- The observatory shall maintain the +Z-axis to within 2°, 1 sigma, radial, with a goal of 0.5°, of its commanded direction.
- The observatory shall not point the +Z observatory axis within 30 degrees (TBR) of any portion of the Earth, except during a repointing slew or by ground command.
- When the observation target is unocculted, but within 30 degrees of the Earth, the observation target shall be maintained within 30 degrees of the +Z Observatory axis.
- When the observation target is occulted, or unocculted but within 30 degrees of the Earth, the Commanded pointing direction shall be adjusted by the spacecraft to accommodate the Earth avoidance constraint.

It is noted that these pointing requirements are not, in general, a problem for a tether system that is generating low thrust. As will be seen, the dynamic forces will be quite small and necessarily on the order of atmospheric drag forces (for the drag make-up case).

System mass and instrument power budgets as presently understood are surnmarised in Table 1 for reference.

Table 1 – GLAST Mass and Power

Total Mass	**<4460 kg**
S/C Mass	<1100 kg
LAT Mass	<3000 kg
GBM Mass	<70 kg
Re-entry Fuel	290 kg
Instrument Power	
LAT	1650 W
GBM	<65 W

END-OF-MISSION DEORBIT

As noted, the GLAST mission has a requirement for a controlled deorbit of the spacecraft for end-of-mission. It is specified that there should be an 85% probability of successful controlled re-entry and safe ocean disposal compliant with NSS 1740.14. As a first assessment, the basic mass costs to meet this requirement with and without an ED tether are evaluated. It is also noted that to assure adequate probability of success, it would be necessary to assure adequate reliability of critical subsystems at end-of-mission. Specific subsystem reliability requirements are not considered here.

There are two configurations that were studied in detail: (1) Hydrazine propulsion system only and *(2)* a combined EDT plus Hydrazine system. **A** third option of using an EDT only was rejected for the GLAST mission because of the requirement of a controlled descent. It is thought that there is too much uncertainty of performance for the EDT at the lowest altitudes (below approximately 250-200 km) for a controlled deorbit. If this requirement **was** eliminated, or for another mission without this constraint, additional mass savings could be obtained. Further, an EDT only deorbit system potentially could simplify requirements on subsystem lifetime reliability since and EDT could provide a measure of its own attitude control using gravity-gradient forces.

NOMENCLATURE

B	Earth magnetic field (Tesla)
ΔX_L	Size of landing site
C_d	Coefficient of drag
I_t	Tether Current
Fgg	Gravity gradient force of tether
F_t	ED tether thrust
L	Length of tether above center-of-gravity (C.G.) of spacecraft (m)
m	effective mass of endbody and portion of tether above (below) spacecraft C.G. (kg)
S_{SC}	Spacecraft system frontal area (m^2)
ω_O	Spacecraft orbit rotation rate (s^{-1})

BACKGROUND

Electrodynamic tethers can produce thrust for satellites by using a force that is produced when running a current through a wire in a magnetic field. A decelerating force on the tether is produced when a current flows up and in a direction away from earth. This can be a current driven purely by the electromotive force (emf) generated by the motion of the tether with respect to the earth's magnetic field and thus generated at the expense of the satellite's orbit. However, if the motional emf is overcome by another power source (e.g. solar panels, batteries or some other source) the current can be made to change direction producing an accelerating force on the tether system. An important tether configuration for both orbit lowering or boosting that is considered here is a bare-tether which allows electron collection along a portioon of the tether.[1,2,3]

The **NASA** Gamma-ray Large Area Space Telescope (GLAST) is the next high energy gamma-ray space mission, scheduled for launch in 2006 [Gehrels]. GLAST is a high-energy gamma-ray observatory designed for making observations of celestial sources in the energy band extending from 20 MeV to 300 GeV with complementary coverage between 10 keV and 25 MeV for gamma-ray bursts. This mission will identify and study nature's high-energy particle accelerators through observations of active galactic nuclei, pulsars, stellar-mass black holes, supernova remnants, gamma-ray bursts, Solar and stellar flares, the diffuse galactic and extragalactic high-energy radiation, and unidentified high-energy gamma-ray sources.

GLAST MISSION - REQUIREMENTS

The main instrument for the GLAST spacecraft is the Large Area Telescope (LAT).[4] It will provide collecting area, field of view, angular resolution and position resolution that greatly improve on the capabilities of the predecessor instrument EGRET' on the Compton Gamma Ray Observatory. The LAT will cover the energy range 20 MeV - 300 GeV, with capabilities up to 1 TeV. The LAT also achieves sufficient background discrimination against the large fluxes of cosmic-rays, earth albedo gamma rays, and trapped radiation that are encountered in orbit. The secondary GLAST Burst Monitor (GBM) instrument is required to simultaneously observe gamma-ray bursts in the classical low-energy gamma-ray band and provide rapid burst location information. A sketch of the GLAST spacecraft is given in Figure 1.

After instrument checkout and calibration, GLAST will perform a one-year all-sky survey during which the observatory shall scan the 55° half-angle LAT field of view over the full celestial sphere repetitively every 2 orbits. During *sky survey* mode the spacecraft will be oriented to point the LAT instrument in a general zenith direction with some rocking motion around the orbit to improve the uniformity of the sky coverage. There may be occasional interruptions of the survey for pointed observations of particulars transient sources. ***Pointed*** mode has the LAT instrument oriented toward a position of interest to within 30° while it is above the Earth's limb. After the one-year survey, the mission will have a mixture of sky survey and pointed mode observations. In general, GLAST will not point below 30° above the Earth's limb to avoid Earth albedo gamma rays.[6]

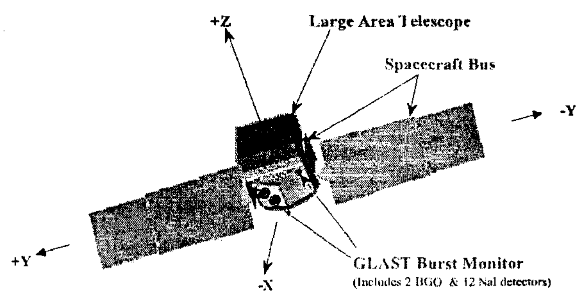

Figure 1 – Example Figure

The operational lifetime of the GLAST system shall be a minimum of 5 years, with a goal of 10 years (not to exceed 25 years per **NASA**

550 km to approximately 150 km. At this point, MR107 system would be fired once for a final deorbit burn. Note that it is also possible to have the EDT stop at some point and allow the spacecraft to coast to 150 km though there is no mass or time advantage in doing this.

The EDT system considered here consists of a (1) upwardly deployed endbody that includes the tether deployer, (2) bare, conducting tether between the endbody and the GLAST spacecraft to collect and conduct current down the tether, (3) endbody deployment hardware (e.g. Marmon clamp), and (4) a current control and electron emission subsystem that is integrated in to the GLAST spacecraft.

The endbody's purpose is principally to help generate adequate tension for stabilizing the tether while it experiences the EDT thrusting. The tether tension (gravity gradient) force can be approximated as[7]

$$F_{gg} = 3Lm\omega_o^2 \qquad (1)$$

where it is assumed here that the bottom spacecraft (GLAST) is much larger (massive) than the upper endbody.

The tether is assumed to be of a tape design with a width of approximately 2.5 cm. The conductor is aluminium and is assumed to be 50% porous. The resulting tether mass is 6.75 kg/km and the series electrical resistance is 30 Ω/km. It is assumed that the last 100 m is covered with an insulating material to insulate it from the ionospheric plasma. We do not address important issues associated with proper treatment of the bare tether or insulation coating. But do note that this is an important technical issue.[9] In this study, we consider conducting tether lengths of 2, 3, and *5* km.

At the GLAST spacecraft we would assume some kind of endbody release and deployment system like that used on the ProSEDS mission.' The electron emitter is assumed to be a low-power system that can generate electron emission without the use of consumables as used on. a hollow cathode electron emitter or high-temperature, high-power filament systems. This technology is the subject of on-going research, but a technology such as field-emitter array (FEA) electron emitters are envisioned.[10,11,12] The electron emitter is assumed to utilise 60 V of the available tether emf for electron emission.

The system mass is estimated in Table *2* for the 3 km EDT yielding a mass value of 76.4 kg. The *2* and *5* km tether case is obtained by subtracting 8.78 kg and adding 17.6 kg, respectively for the mass of the tether plus 30% contingency. Using the hyrdazine propulsion system described above to derobit from 150 km, the minimum propulsion system mass that would be required would be 45.0 kg for the ΔX_L= 200 km case, and 23.7 kg. for the ΔX_L= 500 km case. Depending on tether length, the total system mass for the combined system would range from 113 to 146 kg for the ΔX_L= 200 kin case and 92-118 kg for the ΔX_L= 500 km case.

Table 2 – EDT Deorbit System Mass

EDT *System-Deorbit*	**Mass (kg)**
End-body (w deployer):	30
Al Tether (3km):	20.3
Endbody deployer	5
Electron Emitter & HV Control	2.5
Cabting	1
Power Generation and Delivery Equip. (3 W avg)	Negligible
Contingency (30%)	17.6
Total	76.4

Assuming total area for the spacecraft plus tether system of 29 m^2, the time to move from 550 to 150 km at solar minimum is determined to be 140, 101, and 81 days for the 2, 3, and 5 km tether lengths, respectively. By comparison at the 3 km tether length during solar maximum, it would only take 32 days.

Figure 3 shows the altitude versus time profile for the solar min, 5 km tether case. Figure 4 shows the 3 km case at solar maximum. The predicted thrust for the same case as Figure 4 is given in Figure 5. Near the end of the drop in altitude, the peak in thrust is obtained (0.88 N). For simplistic control of tether dynamic stability, the tether tension (Eq. (1)) should be sufficiently larger than this amount. However, it has also been proposed that tether stability can be controlled by phased variation of tether current.[13]

DRAG MAKE-UP

Of the three inclinations initially considered for GLAST, the orbit with 28.5" inclination has the least uniform exposure, owing to the observing time lost to the SAA below the equatorial plane (see Figure 6). The overall average exposure time is also least for this inclination. The average exposure time for a 0" inclination orbit is 17% greater, with essentially all

Hydrazine Thruster Option

Numerical simulations have been performed to determine the size of a N_2H_4 propulsion system capable of deterministically de-orbiting the GLAST spacecraft at the end of its mission life. The amount of propellant required to de-orbit the GLAST spacecraft can be computed as a function of altitude and the uncertainty in delivered impulse (assumed to be ±1%) and the size of the landing site **(AX,).** For the simulation N_2H_4 is assumed to have a specific gravity of unity and delivered I„ of 220 sec. Assuming a statistically representative PV/W for the tank of 2595 m, the tank mass can be determined as a function of the propellant quantity, blow-down ratio (4:1), and the propellant storage temperature extremes (10-50° C). (The propellant and tank generally comprise 70-90% of system mass of N_2H_4 systems.) Here the de-orbit system has been assumed very simple, and the mass of one General Dynamics MR-107 thruster *(0.9* kg) + 0.5 kg for tubing and other components are added to the propellant and tank masses to approximate the total system mass. The spacecraft drag has been computed as $^1/_2 C_d S_{SC} \rho V^2$ with C_d = 2.2, S_{SC} = 19 m^2, and ρ calculated using an exponential atmospheric model. The GLAST spacecraft mass at the end of mission life has been approximated a constant 4460 kg (for the largest cases, the propellant represents, in fact, –10% of the system mass prior to de-orbit).

Figure 2 can be used to estimate the mass of the propulsion system that would be required to de-orbit the GLAST spacecraft from any altitude between 500 and 150 km. Two different landing site areas (assuming a target landing area with a diameter of 200 km or 500 km) are included. The more precise, or more determininistic the de-orbit, the higher the AV, and thus the higher the mass of the propulsion system. The minimum propulsion system mass that would be required, if the tether system were to deorbit the spacecraft to an altitude of 150 km would be 50 kg for the ΔX_L= 200 km case, and 27 kg. for the ΔX_L= 500 kin case.

The MR-107 provides 220 N of continuous thrust. For this study, it was assumed that the deorbit **AV** would be applied in the last orbit, or last several orbits, as appropriate, always firing for a short period at apogee. For the most demanding deorbit case, (starting at 550 km altitude, and requiring a 200 km window for the landing site) for example, the MR107 would provide three burns of 13.3 minutes in each of the last three orbits. Higher I_{sp}, lower thrust propulsion systems (electro-hydrazine thrusters and arcjets) were also examined, but the low thrust of these devices resulted in much increased system mass for continuous firing scenarios, and years of added mission time for apogee-only firing.

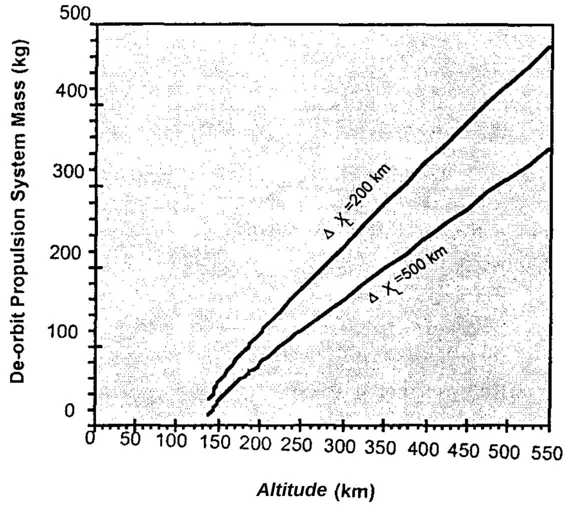

Figure 2 – Hydrazine mass requirements for orbit reduction assuming S_{SC} = 19 m^2.

Some additional, compound scenarios were also examined. Propulsion system mass to reduce orbit from 550 to 350 km and then coast into undeterministic deorbit = 267 kg. Propulsion system mass to reduce orbit from 550 to 350 km, coast to 150 km, and then deterministically de-orbit = 317 kg for ΔX_L = 200 km, 298 kg for ΔX_L = 500 km. (Note that 150 km represents a reasonable minimum altitude. The system can coast somewhat lower, but this won't reduce the required propellant by very much, since most of the it is used for the initial Hohmann transfer in this case.)

This analysis was also completed assuming a spacecraft frontal area of 29 m^2. The results for the compound scenarios were also calculated for the 29 m^2 spacecraft frontal area case. Propulsion system mass to reduce orbit from 550 to 350 km and then coast into undeterministic = 267 kg. Propulsion system mass to reduce orbit from 550 to 350 km, coast to 150km, and then deterministically de-orbit = 312 kg for ΔX_L = 200 km, 291 kg for ΔX_L = 500.

ED Tether + Hydrazine Thruster Option

A hybrid deorbit system was evaluated using the same N_2H_4 MR107 propulsion system' and a ED tether system. The scenario followed is to have the EDT reduce the GLAST spacecraft altitude from

28.5"	14.7%
5"	*5.3%*
0°	0%

[1] For a one-year sky survey.

Chemical and EP Drag Make-Up

The mass of three propulsion systems required to provide the drag make up for the GLAST spacecraft at 400 altitudes (and 550 km for reference) has been calculated. For each altitude, the integrated drag of the spacecraft over the nominal 5 year mission life was calculated using the average atmospheric densities listed in Table 4 (which are the median values of the approximate MSIS solar max and min, since the mission is approximately one half of a solar cycle). The total impulse required to compensate for this total mission drag was then calculated. This total impulse was then used to size the propulsion systems listed below. The actual frequency and duration of thruster firing is a function of the thrust level of each thruster. The N_2H_4 REA was assumed to be a General Dynamics MR-103 Rocket Engine Assembly, which provides 1 N of continuous thrust at a specific impulse of 210 seconds'. The N_2H_4 Arcjet was assumed to be a General Dynamics MR-510 Arcjet, which provides 220 mN of continuous thrust at a specific impulse of 600 seconds[15]. The Ion thruster was based on the NSTAR ion thruster that flew on the DS-1 spacecraft, which provides 100 mN of continuous thrust at a specific impulse of 3,000 seconds,[16,17,18] The spacecraft frontal area that was used for all calculations is 29.0 m^2 to be conservative, and the coefficient of drag that was assumed was 2.2. In all cases, a mass was added for the power generation and delivery system. This mass was calculated assuming 35 W/kg. This calculation did not include the additional frontal area of solar arrays that could be necessitated by the electric propulsion systems, but the assumed area is conservative. The resulting mass break-down for each of the propulsion systems is listed in Table 5.

As is seen in Table 4, the required masses ranging from 1021 down to 164 kg. The higher ISP DS-1 ion thruster requires approximately 2500 W power to operate once PCU efficiencies are considered (92-94%). It may be possible that the average power would be less given the approximate 100 mN thrust is a factor of just under two larger than possible drag forces at solar max. While it is possible that the ion engine could be optimised, it is noted that the 5 year mission duration is considered the minimum with 10 years desired.

EDT Drag Make-up

The EDT system considered here is similar to the deorbit case above except that the endbody is assumed to be 25 kg and the tether length is fixed at 2 km. For the drag make-up case here much of the tether is insulated leaving only 100 m of the bottom portion of the tether bare for current collection (This is not necessarily optimized, but is representative). **A** much larger mass must be allowed for the power generation and delivery equipment as summarize in Table 6 giving a value of 83.3 kg which includes a 30% contingency factor. Other components **are** assumed to be approximately the same as the deorbit case. It useful to point out that assuming no consumables it is possible to consider an EDT that can operate for a full 10 year lifetime. By making the tether short and robust issues of tether lifetime can be mitigated.

For the proposed EDT drag make-up configuration the performance at solar maximum (average atmospheric density $= 7.6 \times 10^{-12}$ kg/m^3) conditions is shown in Figure 7 where both altitude and thrust ($F_t \sim I_t L \times B$) are indicated for one day of operation. In this example, it is assumed that thrust is applied at all times and then peaked around apogee for 20 minutes. The base power is 135 W, peak power is 600 W (accounting for a PCU 92% efficiency), and the average power is 221 W. The importance of variable thrusting is illustrated in Figure 8 where a constant power of 150 W is seen to maintain apogee, but perigee is steadily increasing. This is due to the variation in atmospheric density **as** well as the variability in the EDT thrusting due to ionospheric variations. EMT and tether current for this same situation is included in Figures **9** and 10, respectively.

Solar minimum (average atmospheric density $= 9.7 \times 10^{-13}$ kg/m^3) conditions were also studied in even greater detail and a survey of possible thrusting scenarios were evaluated. Specifically, it was considered that peak thrust was applied every orbit, every second orbit, or, every third orbit. And, thrust was applied for **5**, 10, and 20 minutes. The results of this survey are shown in Table 7 below. The drop in atmospheric density clearly resulted in a substantially smaller power requirements (base power is 10 W, average power is –16 W). The peak power ranged from 32 to 300 W. Figures 11 and 12 show examples of two thrusting scenarios.

We note again that for tether dynamic stability it is desired to have the tether tension (Eq. 1) larger of the difference due to SAA passage at the higher inclination (Table 3). Even the 5° inclination orbit has 10% greater average exposure time than the present baseline GLAST orbit. For all of the inclinations considered, the most uniform exposure is obtained with a 35° rocking angle.[14]

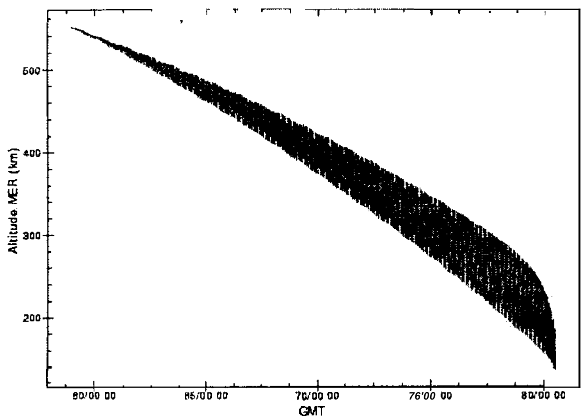

Figure 3 – 5 km long tether system drop from 550 to 150 km altitude during solar minimum conditions.

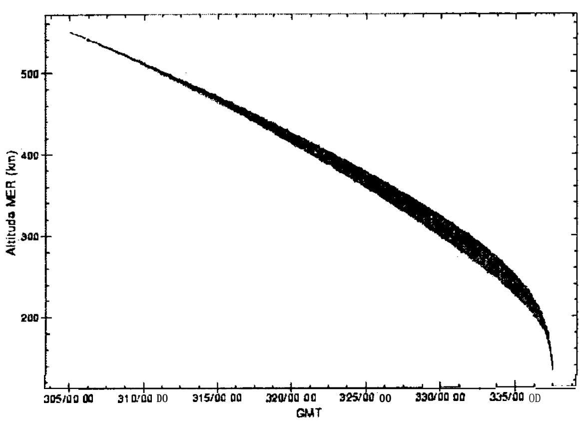

Figure **4** – 3km tether system drop from 550 to 150 km altitude during solar maximum conditions.

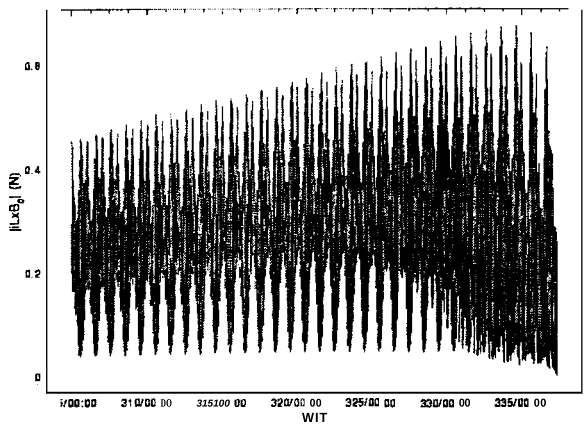

Figure **5** – Thrust for case in Figure **4.**

Digel[14] has also analyzed the impact of lower orbits at 28.5" inclination. To reduce observing time lost in the **SAA,** the desired orbit should be as low in altitude and inclination as is practical. For example, a 400 km orbit would reduce the time lost in the SAA from that experienced at the nominal *550* km by –40%. Thus, a hypothetical situation of using drag make-up propulsion to assure adequate mission lifetime has been studied using a 400 km, 28.5° inclination orbit.

South Atlantic Anomaly

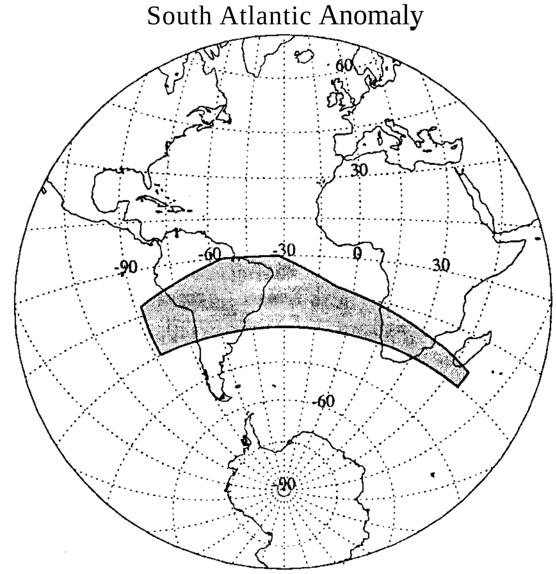

Figure ***6*** - The extent of the South Atlantic Anomaly. No exposure is accumulated when GLAST is within the shaded region. [14]

Table 3 – Sky **Survey** Exposures for Different Inclination Orbits. [14]

Inclination	Fraction of Time in SAA[1]

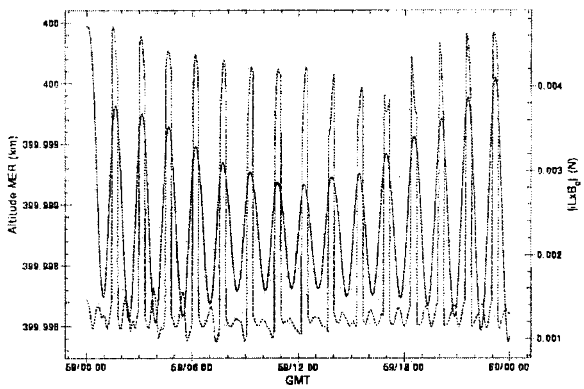

Figure 11 - Thrust and altitude variation over one day for drag make-up EDT at solar minimum . Peak thrust once per orbit for 20 minutes.

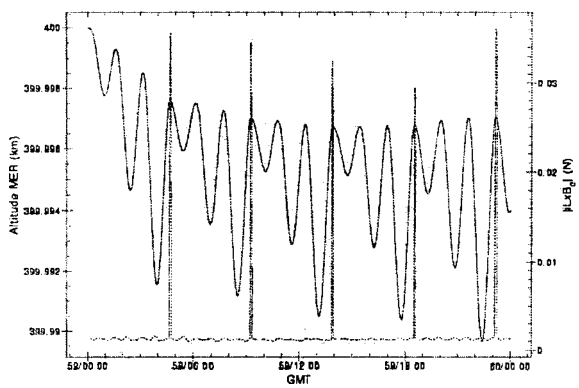

Figure 12 - Thrust and altitude variation over one day for drag make-up EDT at solar minimum. Peak thrust every 3rd orbit for 5 minutes.

CONCLUSION

The use of even a hybrid hydrazine plus EDT propulsion system for end-of-mission of a GLAST class space science mission yields approximately a factor of 3 improvement in mass requirements. This system satisfies requirements for a deterministic deorbit.

An EDT drag make-up system to allow GL:AST to operate at 400 km and 28.5° inclination and thus improve both observation time and survey uniformity would require approximately 83 kg (assuming a 30% contingency). It is useful to note that this lower altitude would require less propellant for deorbit even if an EDT deorbit system was not used.

These results serve to demonstrate that ED tethers provide new capabilities for NASA's space science missions that could both improve performance and lower cost.

Acknowledgements

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than the tether force. In this situation, the tether tension is estimated to be approxiinately 0.2 N which is compared with 0.055 N thrust indicated in Figure 7. The tape tether itself will likely introduce additional features into the tether dynamics that will need to be studied.

Table 6 – EDT Drag Make-up System Mass

EDT *System-Deorbit*	**Mass (kg)**
End-body (w deployer):	25
Al Tether (2km):	13.5
Endbody deployer	5
Electron Emitter & HV Control	2.5
Cabling	1
Power Generation and Delivery Equip. (3 W avg)	17.1
Contingency (30%)	19.2
Total	83.3

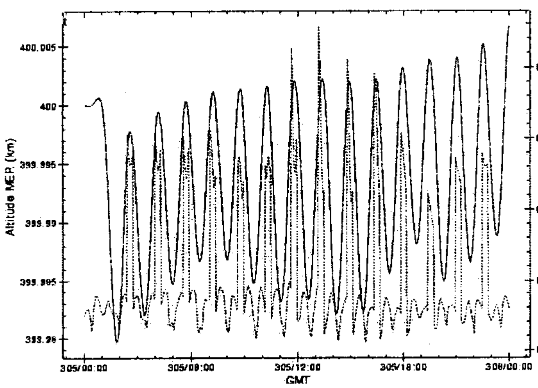

Figure 7 – Thrust and altitude variation over one day for drag make-up EDT at solar maximum

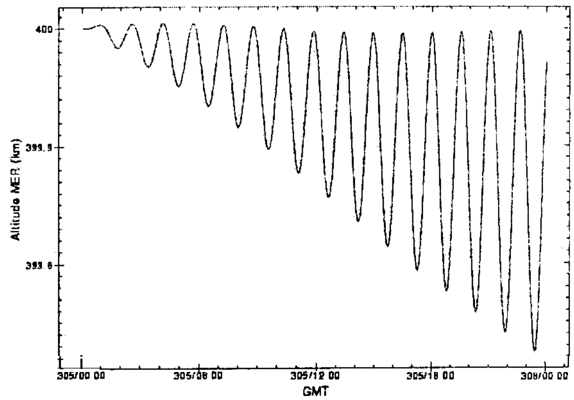

Figure 8 – Thrust and altitude variation over one day for drag make-up EDT at solar maximum

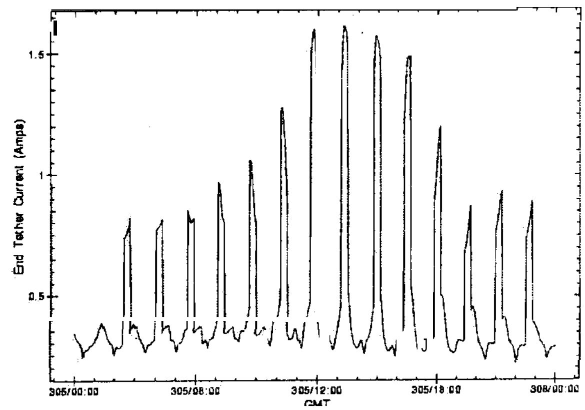

Figure 9 – Tether current variation that goes with Figure **7.**

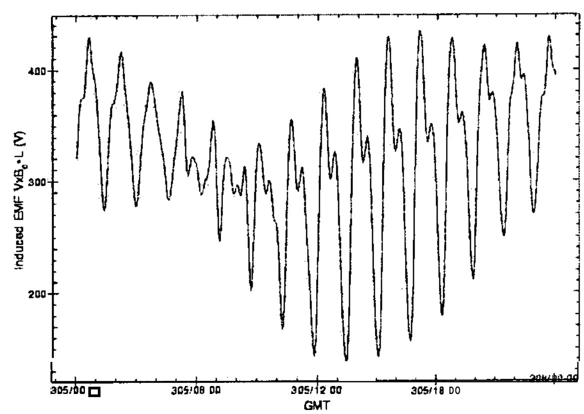

Figure 10 – EMF variation that goes with Figure 7.

	Isp (sec)	Altitude (km)	Required Drag Make-up Propellant (kg)	Estimated Total Mass (kg)	Comments:
N_2H_4 REA	220	**400**	857	**1021**	One 1 N thruster. 40% ullage Tank PV/W = 3,810 m.
		550	140	171	
N_2H_4 Arcjet	590	**400**	320	**448**	One 220 mN arcjet w/ PPU 40% ullage. Tank PV/W = 3,810 m
		550	52	131	
Ion	3130	**400**	60	**164**	One 100 mN ion thruster w/ PPU Tank scaled from actual NSTAR propellant system.
		550	10	112	

Table 5 – Propulsion system mass break for drag make-up case

		400 km	550 km
N_2H_4 REA	Propellant:	857.4	139.5
	Propellant Tank:	154.5	25.1
General *Dynamics MR-103*	Pressurant:	4.3	0.7
	Power Generation and Delivery Equip.	0.3	0.3
	Feed System	5.0	5.0
	1 N REA (9 W heater power)	0.7	0.7
N_2H_4 Arcjet	Propellant:	319.7	52.0
	Propellant Tank:	57.6	9.4
General *Dynamics MR-510*	Pressurant:	1.6	0.3
	Power Generation and Delivery Equip.	57.4	57.4
	Feed System	5.0	5.0
	220 mN Arcjet	1.5	1.5
	Arcjet PCU (2008 W)	5.0	5.0
Ion Thruster	Propellant:	60.3	9.8
	Propellant Tank:	2.2	0.4
Based on the *NSTAR ION*	Power Generation and Delivery Equip.	72.1	72.1
thruster used on DS-1	Feed System	5.0	5.0
	100 mN Ion Thruster	8.3	8.3
	Ion Thruster PCU (2522 W)	13.3	13.3
	Ion Thruster DCIU	2.5	2.5

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	Every Orbit			Every Two Orbits			Every Three Orbits		
	5 Min	10 Min	20 Min	5 Min	10 Min	20 Min	5 Min	10 Min	20 Min
Average Power (W)	15.2	15.6	16.9	15.2	15.2	15.8	15.3	15.2	15.5
Peak Power (W)	100	54	32	200	106	55	300	150	80
Minimum Power (W)	10	10	10	10	10	10	10	10	10