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CubeSat Lunar Mission Using a Miniature Ion Thruster

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The feasibility of CubeSats utilizing the Miniature Xenon Ion (MiXI) thruster for lunar missions is the focus of this investigation. The successful heritage, simplicity, and low cost of CubeSats make them attractive candidates for scientific missions to the Moon. This investigation presents the first-order design process for developing a lunar mission CubeSat. The results from this process are then applied to a 3U CubeSat equipped with a MiXI thruster and specifically designed to reach the lunar surface from a low Earth orbit. The small, 3cm diameter MiXI thruster utilized is capable of producing 0.1–1.553mN of thrust with a specific impulse of over 3000s and is projected to be capable of generating over 7000m/s of ΔV for a CubeSat mission. With all margins and contingencies considered, the total power required for the CubeSat using a 40W ion thruster is approximately 92W during daylight operations. With an assumed solar incidence angle of 45°, this power can be generated from a Spectrolab UTJ solar array of 0.35m^2. Thermal calculations show an average spacecraft surface temperature of approximately 59°C during daylight operations and 0°C during eclipse, which are within the operational and survivable temperature ranges of the spacecraft subsystems. A low-thrust trajectory model is utilized to calculate and plot Earth-Moon trajectories.

Nomenclature

A_A	= albedo lit surface area [m^2]
A_E	= Earth-facing surface area [m^2]
A_E	= spacecraft Earth-facing area [m^2]
A_L	= sunlit surface area [m^2]
A_R	= radiator area [m^2]
A_{SA}	= solar array area [m^2]
A_{SP}	= space-facing surface area [m^2]
α	= absorptivity
$\Delta\alpha$	= inclination change [deg.]
c	= effective exhaust velocity [m/s]
d	= battery depth of discharge
D	= mission-life degradation
D_o	= degradation per year
ε_{sc}	= spacecraft emissivity
F_A	= albedo factor
F_{IR}	= IR factor
g	= gravitational constant [m/s^2]
η_{BOL}	= efficiency, beginning of mission life

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η_{EOL} = efficiency, end of mission life
η_P = battery power usage efficiency
η_p = power efficiency
η_t = thrust efficiency
I_{IR} = IR radiation intensity [W/m^2]
I_S = solar intensity (1367 W/m^2)
I_{sp} = specific impulse [s]
l = mission life [years]
M = spacecraft dry mass [kg]
M_{avg} = spacecraft average mass [kg]
m_b = battery mass [kg]
m_p = propellant mass required [kg]
$\dot{m}_p$ = propellant mass flow rate [kg/s]
m_{SA} = solar array mass [kg]
$P_{b,req}$ = battery power required [W]
P_{BOL} = power generation, beginning of mission life [W]
p_{BOL} = specific power generation, beginning of mission life [W/m^2]
P_{EOL} = power generation, end of mission life [W]
p_{EOL} = specific power generation, end of mission life [W/m^2]
P_{in} = power needed for operation [W]
P_{req} = power required [W]
P_t = thruster input power [W]
P_{tot} = total power [W]
Q_A = heat due to albedo [W]
Q_E = heat radiated to Earth [W]
Q_{in} = heat influx [W]
Q_{int} = internal heat production [W]
Q_{IR} = heat due to IR radiation [W]
Q_{loss} = heat lost from electronic system inefficiencies [W]
Q_{out} = heat outflux [W]
Q_R = heat radiated passively to space *via radiators* [W]
Q_S = heat due to incident sunlight [W]
Q_{SP} = heat radiated passively to space [W]
R = orbit radius [AU]
ρ_A = albedo density
ρ_b = battery power density [W/kg]
ρ_{BOL} = power density, beginning of mission life [W/kg]
ρ_{EOL} = power density, end of mission life [W/kg]
σ = Stefan-Boltzman constant (~5.67 E -8 W/m^{2}K^4)
T = thrust [N]
t = time [s]
t_b = battery usage time [hrs]
t_{burn} = thruster burn time [s]
T_E = Earth temperature (~287 K)
T_{SC} = spacecraft / solar array temperature [K]
T_{SP} = space temperature (~2.7 K)
Θ_S = sunlight incident angle [rad]
V_c = circular orbit velocity [m/s]
ΔV = velocity change [m/s]

I. INTRODUCTION

ALTHOUGH scientific missions to the Moon offer valuable data for solar system researchers, the high costs and elevated risks involved in sending conventional spacecraft to the Moon have significantly restricted the number of such missions. The increasing interest in micro- and nano-satellite technology by the aerospace community, however, has uncovered the potential for autonomous micro-satellite missions to the Moon. This investigation aims to explain the process by which a lunar mission CubeSat can be designed and to demonstrate that such satellites utilizing miniaturized ion thruster technology have the theoretical capability of delivering scientific payloads to the Moon.

CubeSats, small cubic geometry satellites, exhibit many important qualities, such as ease of construction, low cost, and a high degree of mission flexibility. A CubeSat is a satellite whose geometry is based on cubic cells called U's (for "units"), each measuring 10cm along each edge. These cells can then be assembled to create prism geometries and are named based on the number of adjoined cells (i.e. a 2 cell CubeSat is called a 2U). The strength of CubeSat heritage can be seen by the recent proliferation of microsatellite missions. Several recent CubeSat missions include the biological science microsatellites PharmaSat (May 2009)[1] and GeneSat-1 (Dec 2006)[2], the second Canadian advanced nanospace experiment satellite CanX-2 (April 2008)[3], and the geological science QuakeSat (June 2003)[4]. The ELFIN (Electron Loss and Fields Investigation) CubeSat under development at the University of California Los Angeles (UCLA) by the Institute of Geophysics and Planetary Physics (IGPP) is a prime example of a 3U CubeSat, sharing the basic structural geometry that may be incorporated into a lunar mission satellite[5,6].

The primary size limitation for lunar mission satellites utilizing a single MiXI (Miniature Xenon Ion) thruster is based on the thruster-specific performance. Although a larger satellite using a miniature ion thruster may arrive at its destination, the transfer time and propellant mass required may be prohibitively high, especially when compared to a micro-satellite mission. Therefore, candidate satellite configurations being examined for lunar missions using a miniature ion thruster include 3U CubeSats (circa 4kg) and satellites of approximately five times this size (circa 20kg). The smaller, 3U variants are of interest due to several of their key features:

1. Low development and mission cost
2. Ease of construction
3. Frequent launch opportunities
4. Successful heritage

By contrast, the larger satellites benefit from a different set of unique characteristics that may provide for more ambitious mission capabilities:

1. Increased scientific payload
2. Higher system power / higher thrust
3. Potential for high battery capacity, hence continuous thrust during eclipse
4. Improved thermal control and margin

Additionally, alternative miniature thruster technologies may be considered.

It should be noted that conventional CubeSats currently require a maximum mass of 4kg and do not allow the use of pressurized vessels beyond 1.2atm. These requirements are based on the limitations of industry-standard CubeSat deployment systems and are aimed at reducing the risk to primary spacecraft sharing the same launch vehicle. Nevertheless, due to the possibility of gaining mission-specific permission to launch a CubeSat that does not fulfill these requirements, this investigation provides valuable insight regarding the potential for lunar mission CubeSats.

There are two primary objectives of this investigation:

1. Describe in detail the first-order mission and satellite design process applied to lunar mission CubeSats.
2. Show the results of this process as it is applied to the 3U lunar mission CubeSat, employing the highly successful MiXI thruster, under development at UCLA.

The first objective will be accomplished by examining key spacecraft subsystems and discussing the necessary considerations and equations associated with each. Although this process is executed specifically for lunar mission CubeSats in this investigation, it can be implemented for a variety of situations and is not restricted to lunar mission CubeSats. The second objective will be accomplished through a discussion of the recent findings from the UCLA-based investigation on lunar mission CubeSats. Details regarding the masses and power budgets, in addition to the performance of each major subsystem, are discussed.

II. METHODOLOGY FOR FIRST-ORDER LUNAR MISSION CUBESAT DESIGN

1. Structure

A 3U CubeSat frame is an extremely favorable design for microsatellite lunar missions. The 3U frame offers sufficient interior volume for all basic hardware necessary for CubeSat operations, along with a complete solar electric propulsion system and a 3-axis control system. It is a proven platform with a solid heritage and readily available, over-the-counter components[7,8]. Additionally, the industry-standard P-POD CubeSat deployment system accepts 3U-frame satellites, allowing for easy integration onto launch vehicles and increased space-flight opportunities[9].

Although a variety of 3U CubeSat designs exist, several consistent guidelines can be found for their structural components. The frame elements are generally machined from aluminum due to its light weight, high strength-to-weight ratio, ease of machining, and low cost[10]. Commercially available 3U space frames have a mass of 200 – 300g depending on the design specifications, materials, etc[7,8]. Additional braces and brackets, ranging in mass from 5 – 50g depending on design, may be required for structural integrity of the frame and to provide mounting points for the CubeSat hardware and payload[5,6,7]. Exterior panels, constructed from either metal or circuit board fiberglass and massing approximately 50g per 10x10cm area of coverage, are fitted to the 3U frame to shield the internal hardware and offer mounting points for either fixed or deployable solar arrays. It is also necessary to add a margin of between 15% and 25% for changes in the mass of these components as they are machined for the prototype[10].

2. Propulsion Subsystem

The propulsion system utilized on a given satellite is highly dependent on the mission's duration and required ΔV. Conventional lunar missions, such as the Apollo program of the 1960's and the Japanese SELENE program, have relied heavily on chemical propulsion to achieve rendezvous with the Moon[11,12]. However, solar-electric propulsion (SEP) systems have demonstrated a highly successful heritage through their implementation on conventional spacecraft for high-ΔV, long-duration missions. The ESA SMART-1 satellite, for example, utilized a PPS-1350 Hall-effect thruster to achieve a lunar orbit while demonstrating a variety of new deep-space technologies[13]. The Hayabusa spacecraft employed four ion thrusters in an attempt to collect an asteroid sample from 433 Eros and return to Earth[14]. The emergence of the CubeSat led to the development of commercially available cold-gas micro-propulsion systems, offering low ΔV maneuvering capabilities to CubeSats[15]. High ΔV chemical propulsion systems are nearly impossible to implement on CubeSats, however, due to strict size constraints and the large propellant volume required.

The long duration and large ΔV requirements of a lunar mission make miniaturized SEP systems a strong candidate for CubeSat operations. Ion thrusters, in particular, are attractive for small spacecraft applications because of their scaling capabilities and high propellant efficiencies[10]. Assuming the use of a 3U CubeSat frame, as described in Section 1, the appropriate ion thruster performance characteristics must be determined. Ion thrusters are capable of providing a range of specific impulses (I_{sp}) from approximately 2000s to over 3500s. By selecting a value for specific impulse, the effective exit velocity (c) for the thruster can be calculated as

$$c = I_{sp} g \tag{1}$$

where g is the acceleration due to gravity at Earth's surface. The thrust (T) produced by the thruster can then be calculated as

$$T = \frac{2\eta_t P_t}{c} \tag{2}$$

where η_t is the thruster power efficiency and P_t is the input power to the thruster. This, in turn, can be used to determine the propellant mass flow rate ($\dot{m}_p$) as

$$\dot{m}_p = \frac{T}{I_{sp}} \tag{3}$$

The total burn duration (t_{burn}) and total ΔV provided by the thruster can then be calculated based on an assumed propellant mass (m_p), determined by the volume available in the CubeSat interior for the propellant tank, and an approximate spacecraft dry mass (M):

$$t_{burn} = \frac{m_p}{\dot{m}_p} \tag{4}$$

$$\Delta V = c \; ln\left(\frac{M+m_p}{M}\right) \tag{5}$$

The ΔV requirement for a low-thrust trajectory can be approximated from a simplified form of Edelbaum's Equation:

$$\Delta V^2 = V_{c1}^2 + V_{c2}^2 - 2V_{c1}V_{c2}\cos\left(\frac{\pi\Delta\alpha}{2}\right) \tag{6a}$$

where V_{c1} and V_{c2} are the initial and final circular orbit velocities of the spacecraft, respectively, and $\Delta\alpha$ is the total inclination change required during the mission to rendezvous with the Moon (in radians). For a lunar mission transfer, these velocities are that of the spacecraft in LEO and that of the Moon, respectively. Eq. 6a can be rearranged to determine the total inclination change possible for a given Earth-centric altitude (which determines the circular orbit velocity, V_c):

$$\Delta\alpha_{max} = \frac{2}{\pi}\cos^{-1}\left(1 - \frac{\Delta V^2}{2V_c^2}\right) \tag{6b}$$

Using Eq. 1 – 5, the spacecraft's achievable ΔV for a given specific impulse, propellant mass, and dry mass can be calculated. This can then be compared to Eq. 6, which determines the approximate total mission ΔV. The results from Eq. 5 (ΔV possible) must be equal to or greater than that of Eq. 6 (ΔV required) for the mission to be possible. With this criteria met, approximate values for the thruster's required specific impulse, thrust, etc. are known and a specific thruster can be selected for the given application.

Calculating the mass of the propulsion subsystem is highly dependent on the specific components selected; however, some first-order approximations can be made to achieve an initial spacecraft design. An iterative process using the aforementioned equations yields a necessary propellant mass for a lunar mission with a given set of launch conditions. It should be noted that a margin *and* contingency on the propellant mass, each of between 10% and 20%, should be added to the total propellant mass to allow for design changes and mission problems during flight[10].

The tankage mass, allocated to the propellant tank and its associated components, usually ranges from 5 – 15% of the required propellant mass[10]. The mass of the necessary feed systems and mounting hardware for the propellant system is approximated as an additional 20 – 30% of the tankage mass[10]. Again, a margin *and* contingency of between 10 – 20% should be added to these mass values[10].

3. Power Subsystem

A CubeSat's power system is highly customizable and should be based on the power requirements of the various subsystems of the spacecraft. As discussed in Section 2, a key component for a lunar mission CubeSat is an SEP system which has relatively high power demands (usually in the kW range) in comparison to miniaturized chemical systems[10]. However, recent developments in thruster miniaturization techniques have led to novel thrusters, such as JPL/UCLA's MiXI thruster, on the scale of centimeters in diameter that can operate below 100W while maintaining the same high specific impulse and precision as larger ion thrusters[16,17,18,19]. Such a thruster is necessary for an autonomous CubeSat to successfully navigate to the lunar surface utilizing self-carried propulsion methods.

The first stage in designing a satellite power system is determining the power requirements for every component of the spacecraft. This includes all propulsion system components, command and data handling (CDH), attitude determination and control systems (ADCS), payload components, etc. For each component, the input power required (P_{req}) for that component can be determined by

$$P_{req} = \frac{P_{in}}{\eta_p} \tag{7}$$

where P_{in} is the power needed to operate the device and η_p is the component's power efficiency. The sum of these required input powers yields the total power required (P_{tot}) by the spacecraft:

$$P_{tot} = \sum P_{req} \tag{8}$$

This total power must then be fed to each component by a power distribution unit (PDU), which in turn is fed by a power processing unit (PPU) that connects to the high voltage electronics assembly (HVEA). Eq. 7 can be used to determine the power required at the PDU (where P_{in} is the total power needed for all of the spacecraft systems), then at the PPU (where P_{in} is the power required at the PDU), and finally the HVEA (where P_{in} is the power required at the HVEA) based on their individual efficiencies. A wiring/cable efficiency (usually around 95%) must be added to the pre-PPU required power (again using Eq. 7). It is also necessary to add an additional margin *and* contingency of between 10 – 20% onto this power budget. This final value is the power required from the solar arrays.

After the power budget has been completed, appropriate solar arrays must be implemented to supply the required spacecraft power. In order to ensure that the solar arrays will provide sufficient power throughout the lifetime of the CubeSat's designed mission, the condition of the arrays must be considered at the end of life (EOL) in addition to the beginning of life (BOL). This yields a degradation coefficient (D) defined as

$$D = (1 - D_o)^l \tag{9}$$

where D_o is the approximate degradation per year (%) and l is the expected mission duration in years. The performance of photovoltaic solar cells ranges from 12 – 25% with several new commercially available designs that are 30% efficient[10,20]. The performance of these cells at EOL can be calculated as

$$\eta_{EOL} = \eta_{BOL}(1 - D) \tag{10}$$

where η_{EOL} and η_{BOL} are the end of life and beginning of life solar cell efficiencies, respectively. The specific power (p) and power density (ρ) of the cells can be calculated in a similar fashion:

$$p_{EOL} = p_{BOL}(1 - D) \tag{11}$$

$$\rho_{EOL} = \rho_{BOL}(1 - D) \tag{12}$$

From these values, the approximate mass and solar cell array area can be determined:

$$m_{SA} = \frac{P_{tot}}{\rho_{EOL}} \tag{13}$$

$$A_{SA} = \frac{P_{tot}}{I_{s,min}\eta_{EOL}\cos(\theta_s)} \tag{14}$$

where θ_s is the angle of incidence to the sun and the minimum solar intensity ($I_{s,min}$) is defined by

$$I_{s,min} = I_E \frac{I(R_{SE})^2}{(R_{max})^2} \tag{15}$$

where I_E is the solar intensity at the Earth (approx. 1367 W/m^2), R_{SE} is the radius from the Sun to the Earth in AU, and R_{max} is the maximum spacecraft trajectory radius from the sun in AU. The maximum radius must be considered to ensure mission success because it represents the minimum power generation situation. Eq. 15 can be altered slightly to calculate the maximum solar intensity ($I_{s,max}$):

$$I_{s,max} = I_E \frac{I(R_{SE})^2}{(R_{min})^2} \tag{16}$$

An additional mass consideration may be necessary if the solar array uses a motorized deployment system, extra support braces, etc. The power delivered by the solar arrays (P_{SA}) can now be determined at EOL and BOL as

$$P_{SA,EOL} = A_{SA}\eta_{EOL}I_{s,min} \tag{17}$$

$$P_{SA,BOL} = A_{SA}\eta_{BOL}I_{s,min} \tag{18}$$

If a battery is required, its mass (m_b) can be approximated by

$$m_b = \frac{P_{b,req}t_b}{\rho_b\eta_b d} \tag{19}$$

where $P_{b,req}$ is the power required from the battery, t_b is the time duration for the battery to supply power in hours, ρ_b is the power density of the battery, η_b is the battery power usage efficiency, and d is the depth of discharge. The final power system mass considerations are cabling and harnessing, which consists of approximately 10 – 25% of the total mass of the power system, and a margin *and* contingency of between 10 – 20% for the power system's mass budget[10].

4. Thermal Subsystem

Thermal considerations for a lunar mission CubeSat are perhaps the most demanding of all the subsystems. The spacecraft's proximity to the sun (approximately 1AU for the duration of any lunar mission), Earth and Moon albedo, and very low spacecraft surface area lead to relatively high daytime operation temperatures for both the spacecraft body and solar arrays. It is therefore imperative that thermal control of the spacecraft is maintained to ensure that all subsystems are kept at their operational (or if necessary, survival temperatures) for the duration of the mission. Internal components of the spacecraft can often be operated between approximately -10°C and 40°C, whereas the solar arrays are usable between -150°C and 100°C[10]. The survival temperature depends on the specific system, but usually extends an additional +/-10°C from the operational range[10].

In order to determine the net flux of heat to or from the spacecraft, a thermal balance must be performed. To maintain an equilibrium condition in the spacecraft and avoid uncontrolled heating or cooling, we must have

$$\sum Q_{in} - \sum Q_{out} = 0 \tag{20}$$

where Q_{in} and Q_{out} is the total heat into and out of the spacecraft, respectively. There are four primary sources of heat flux to the spacecraft: solar energy (Q_S), Earth and Moon albedo ($Q_{A,E}$ and $Q_{A,M}$), infrared energy (Q_{IR}), and internal heat production (Q_{int}). The values for these heats can be determined as follows:

$$Q_{S,cold} = \alpha I_{s,min} A_L \cos(\theta_s) \tag{21}$$

$$Q_{S,hot} = \alpha I_{s,max} A_L \cos(\theta_s) \tag{22}$$

where $Q_{s,cold}$ is the influx of heat at the spacecraft's maximum radius from the sun, $Q_{s,hot}$ is the influx of heat at the spacecraft's minimum radius from the sun, α is the absorptivity and A_L is the sunlit surface area (I_s and θ_s are as defined previously).

$$Q_A = \alpha I_s A_A F_A \rho_A \tag{23}$$

where A_A is the albedo-illuminated surface area, F_A is the albedo intensity factor (fraction of reflected solar intensity), and ρ_A is the albedo of the celestial body being orbited. Eq. 23 can be applied to both the Earth and the Moon in the same manner.

$$Q_{IR} = I_{IR} \varepsilon_{SC} A_E F_{IR} \tag{24}$$

where I_{IR} is the infrared energy intensity, ε_{SC} is the emissivity of the spacecraft, A_E is the Earth-facing surface area, and F_{IR} is the infrared intensity factor.

$$Q_{int} = \sum Q_{loss} = \sum \frac{P_{req}}{1-\eta_p} \tag{25}$$

where Q_{loss} is the heat dissipation by each powered component of the spacecraft due to inherent power usage inefficiencies as shown.

The three primary sources of heat loss from the spacecraft are passive radiation to space (Q_{SP}), heat radiated to and absorbed by the Earth (Q_E), and heat radiated by the use of radiators or special radiating surfaces (Q_R). These are defined as follows:

$$Q_{SP} = \sigma \varepsilon_{SC} A_{SP} (T_{SC}^4 - T_{SP}^4) \tag{26}$$

$$Q_E = \sigma \varepsilon_{SC} A_E (T_{SC}^4 - T_E^4) \tag{27}$$

$$Q_R = \sigma \varepsilon_{SC} A_R (T_{SC}^4 - T_{SP}^4) \tag{28}$$

where σ is the Stefan-Boltzman constant, A_{SP} is the space-facing surface area, A_E is the Earth-facing surface area, A_R is the radiator surface area, T_{SC} is the spacecraft (or solar array) surface temperature, T_{SP} is the temperature of free space (approx. 2.7°K), and T_E is the Earth's surface temperature.

The surface temperature of the spacecraft body or solar array during daylight operations can be determined from inputting Eq. 21 – 28 into Eq. 20 and solving for T_{SC} (note that the terms relating to radiators must be neglected when considering the solar array temperature). For operations during eclipse, heat terms relating to the solar energy and albedo must be neglected as these do not affect the spacecraft.

5. Additional Considerations (Trajectory, ADCS, Comm., & CDH)

A major component of a first-order mission design is an analysis of the spacecraft's proposed flight trajectory. The approximate mission duration can be determined from a simple impulse calculation utilizing the results for the mission's ΔV requirement found from Eq. 6a:

$$M_{avg} * \Delta V = T * t \tag{29}$$

where M_{avg} is the spacecraft average mass during the transfer, T is the spacecraft thrust, and t is the time of to produce the required impulse in seconds. This can be rearranged to produce the approximate time of the transfer:

$$t = \frac{M_{avg} * \Delta V}{T} \tag{30}$$

Depending on the results from this approximate transfer duration, it may be necessary to take into consideration the effects of radiation exposure due to time spent in the Earth's Van Allan belts. This may necessitate radiation hardening of the spacecraft. Plotting the orbit trajectory can be accomplished with either professional software or in-house algorithms. AGI's Satellite Tool Kit (STK) is a powerful tool that permits highly detailed orbit mapping and provides numerous customization options for a given mission plan[21]. UCLA-developed computational software for plotting orbit trajectories is also possible for rough approximations.

Although not the focus of this investigation, a complete first-order spacecraft design requires consideration of the ADCS and CDH subsystems. Because this investigation focuses on a lunar mission spacecraft, it is understood that high orbit average power is required to operate the aforementioned SEP system. To maintain this power during daylight operations, a 3-axis stability system is necessary to ensure that the solar arrays remain accurately pointed towards the sun for the maximum possible portion of the orbit day. Such systems are beginning to become commercially available and may be applicable to 3U CubeSats; however, many designs are currently too large for use on microsatellites and many are not yet flight qualified[22]. The power requirements and mass of any electronics package supporting the 3-axis control system must also be considered. Finally, a comprehensive autonomous navigation software package is necessary for the spacecraft to operate successfully.

The CDH subsystem is a CubeSat's link to Earth. It is comprised of the satellite's onboard antennas, radios, and data circuit boards, all of which vary in size and complexity depending on the given mission. A key difficulty with lunar missions is transmitting and receiving information over the large distances between the Earth and the Moon (approx. 384,000km). A 3U CubeSat's limited internal dimensions require a space-efficient, high-powered receiver and transmitter in conjunction with any appropriate antennas to deliver the performance necessary for a lunar transfer[23,24]. Again, such hardware and supporting software is currently entering the commercial market[23,24].

III. RESULTS FOR LUNAR MISSION CUBESAT INVESTIGATION

For ease of reference, the satellite under investigation at UCLA will herein be referred to as the LuMi (Lunar Mission) CubeSat. A payload mass of 0.75kg with a payload power requirement of 15W was assumed for all ensuing mass and power budgets. These values are loosely based on the UCLA-based ELFIN CubeSat payload requirements of approximately 13% of the total spacecraft dry mass and 20% of the total spacecraft power required[5,6].

1. Structure

After considering a variety of CubeSat frame designs, the LuMi spacecraft will utilize a 3U chassis. Although it has yet to be determined whether the frame will be purchased or machined in-house, many of LuMi's structural elements are loosely based on the ELFIN CubeSat under development at UCLA[5]. Although ELFIN is 3U spinner-type spacecraft which incorporates a scientific instrument on an extendable boom, the basic design elements and overall mass of the chassis is analogous to LuMi and are suitable for initial sizing and mass calculations[5,6].

The majority of LuMi's structure, including the frame, brackets, and braces will be machined from 6061-T6 aluminum to ensure a highly rigid, lightweight spacecraft. The 3U frame designed for LuMi has an estimated mass of 200g, requiring 20 brackets and 8 braces, each massing approximately 5g. The exterior panels will be constructed from 47mm or 63mm thick circuit board fiberglass or machined from thin-wall 6061 T6 aluminum. The mass per 10x10cm section of either of these panel materials is approximately 50g; a 3U CubeSat such as LuMi requires four 10x30cm sections and two 10x10 sections for complete external coverage. Each of the two deployable solar arrays' mounting hardware was approximated at 100g. A dry mass margin of 20% and a dry mass contingency of 15% were applied, yielding an approximate total structural mass of 1.99kg.

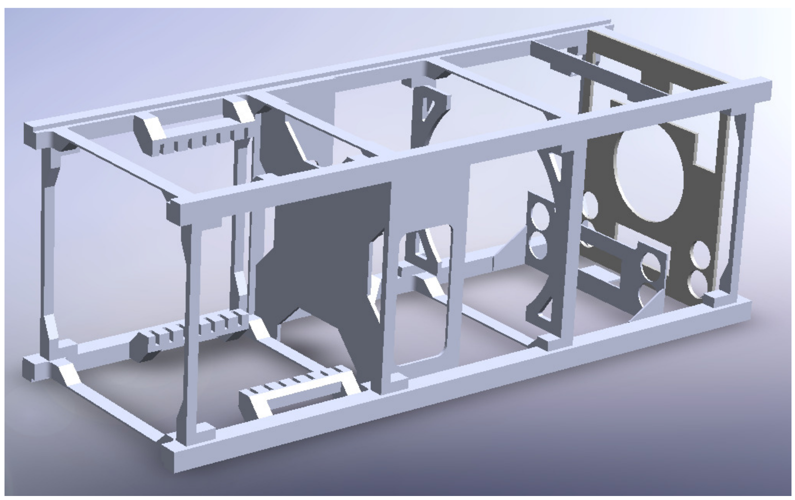

Figure 1. LuMi structural model. *LuMi Spacecraft frame with internal components, exterior panels, and solar arrays hidden. All brackets and braces for internal components are shown.*

2. Propulsion Subsystem

The high ΔV and relatively long mission life requirements of a lunar mission CubeSat led to the implementation of an SEP system on the LuMi spacecraft. The MiXI thruster, developed by Richard Wirz, *et al*, is well suited for use on a CubeSat mission due to its small 3cm diameter and favorable performance characteristics[16,17,18]. The MiXI thruster is capable of producing 0.1–1.553mN of thrust at a specific impulse of over 3000s and achieving a ΔV of over 7000m/s for the current LuMi configuration[16,17,18]. Although the thruster can be calibrated to operate anywhere between 20 – 60W, a 40W model was selected for use on the LuMi spacecraft. For the sake of this study, a specific impulse of 3000s and a thrust of approximately 1.36mN is assumed. The thruster was assumed to deliver a conservative power efficiency of 25% and a conservative thrust efficiency of 50% (these are below the proven efficiencies, but were used for conservative estimates) with a mass of approximately 250g. The tankage and feed system associated with the MiXI thruster were approximated at 180g and 100g, respectively.

It should be noted that current efforts to improve the already successful MiXI design are underway, and a new version of the thruster should be completed by the end of 2011. This design will offer the same performance with increased reliability and a simpler manufacturing process, eventually leading to a production model.

To estimate the necessary ΔV for an Earth-Moon spiral transfer, a launch inclination of 60° and an initial orbit of 600km were selected. It was assumed that the LuMi spacecraft begins its transfer with no excess ΔV granted by the launch vehicle or deployment mechanism, thereby generating all of the mission required ΔV from its ion thruster. Given these initial conditions, the total mission ΔV is approximately 6950m/s. The 6.2kg LuMi spacecraft requires approximately 2.172kg of propellant, including a 10% margin, 15% contingency, and a 5% propellant fraction remaining after the transfer for near-Moon maneuvers, to reach the lunar surface. To house this amount of supercritical xenon fuel, a propellant tank consisting of a cylindrical middle section with semispherical ends will be employed. The semispherical ends require a radius of 4.5cm and the central cylinder must be approximately 6.7cm in length. The tank's inner diameter was assumed to be 95% of the outer diameter, which results in a conservatively robust design for a tank of this size. These tank dimensions will allow the entire propellant system, including the thruster and propellant feed equipment, to fit within approximately 1.8U of the 3U LuMi spacecraft, leaving the remainder of the interior CubeSat volume for the necessary navigation systems, SEP equipment, and payload (see Figure 2).

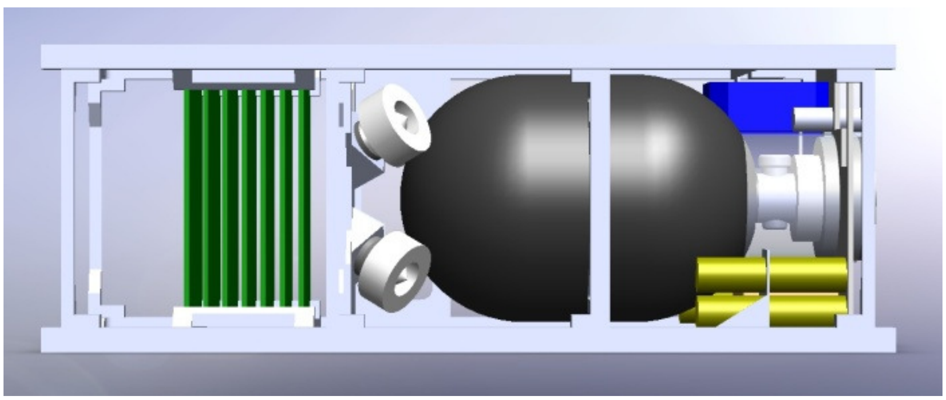

Figure 2. LuMi propulsion subsystem. *MiXI thruster, propellant tank, feed system, and thruster heat shield (faded around thruster) are shown with other internal components. Solar arrays, external panels, and PPU/PDU box foreground are hidden.*

It should be noted that the large propellant volumes required for an Earth-Moon transfer (compared to the total interior volume of a 3U CubeSat) is a critical design feature that will be difficult to implement. However, if the LuMi CubeSat were to reduce its required mission ΔV to 4500m/s and be optimized to achieve maximum inclination change at a low-Earth orbit (LEO), the required propellant would be reduced to approximately 1.35kg (including all margins and contingencies). This would reduce the necessary propellant system volume to approximately 1.4U, offering over 25% more interior volume than a Moon-bound spacecraft. Such a CubeSat design would be capable of an impressive total inclination change of approximately 32° at a 600km LEO (Eq. 6b), thereby establishing itself as a novel approach to Earth-observing satellites.

3. Power Subsystem

Power generation requirements for the LuMi spacecraft depend primarily on the input power for the MiXI thruster with additional considerations for the communications, command and data handling, and payload subsystems. LuMi is designed to employ deployable, gimbaled solar arrays to collect the necessary power for daylight operations. A combination of the spacecraft's 3-axis control system and the gimbaled solar arrays will allow for accurate sun-pointing. The 3-axis system employed consists of 4 miniature momentum wheels with an approximated combined mass of 0.2kg and a power of 5W. Highly efficient methods for despinning the momentum wheel system have been developed utilizing a gimbaled electric thruster[25]. Future iterations of LuMi may therefore incorporate a gimbaled propulsion system.

The LuMi spacecraft will utilize Spectrolab UTJ (ultra triple junction) solar cells which are capable of providing a specific power of over 380W/m^2, a power density of over 450W/kg, and an efficiency of approximately 28% at BOL[20]. With an assumed average solar incidence angle of 45° (a very conservative estimate given gimbaled solar arrays), these values lead to a necessary solar array area of approximately 0.35m^2 with a mass of approximately 0.25kg.

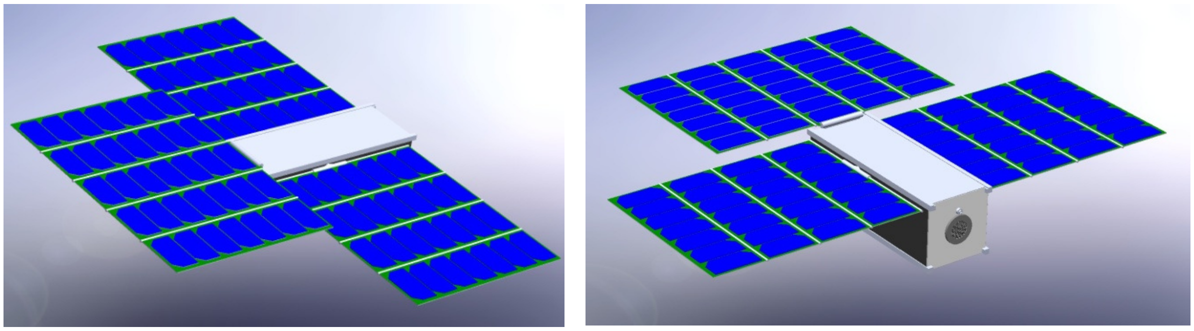

Figure 3. LuMi deployed configuration. *LuMi spacecraft with deployed solar arrays.*

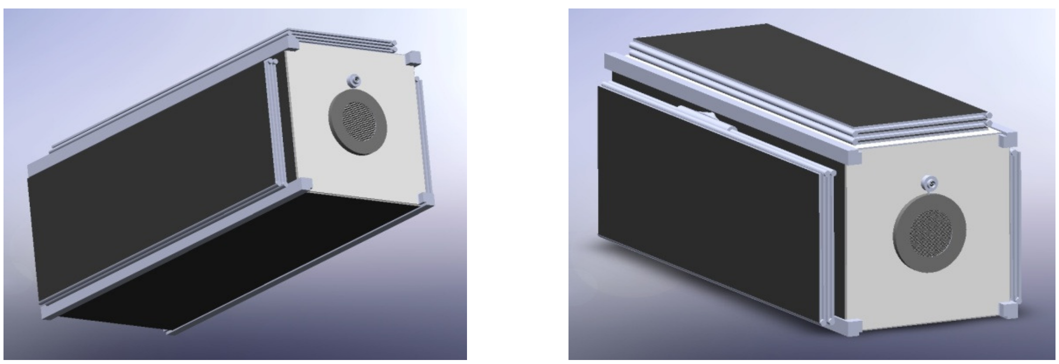

Figure 4. LuMi stowed configuration. *LuMi spacecraft with stowed solar arrays.*

LuMi's PPU has an approximate mass of 0.2kg with a 92% efficiency, while the PDU has a mass of approximately 0.1kg and has an efficiency of 85%. The HVEA's mass was approximated as 0.2kg with an efficiency of 0.97%. These three electronic components are represented as the blue boxes in Figure 5. A set of battery cells (the yellow cylinders in Figure 5) are included in the design to keep the MiXI thruster's cathodes warm during eclipse, which generates greater reliability in the thruster's startup. The batteries are also designed to provide one hour of emergency power for flight-critical subsystems during eclipse. The required battery mass was approximately 0.19kg. A battery charger (the orange box behind batteries in Figure 5) was also included at a mass of 0.05kg. A power margin of 20% and a power contingency of 15% were applied to all power requirements for the spacecraft. The power budget for LuMi utilizing a nominally 40W MiXI thruster is 92W after all subsystem component efficiencies, margins, and contingencies are considered. Although this is a very high power budget for a 3U CubeSat, new and commercially available designs are capable of providing nearly 70W orbit average power, suggesting that 92W is certainly possible in the near future[7].

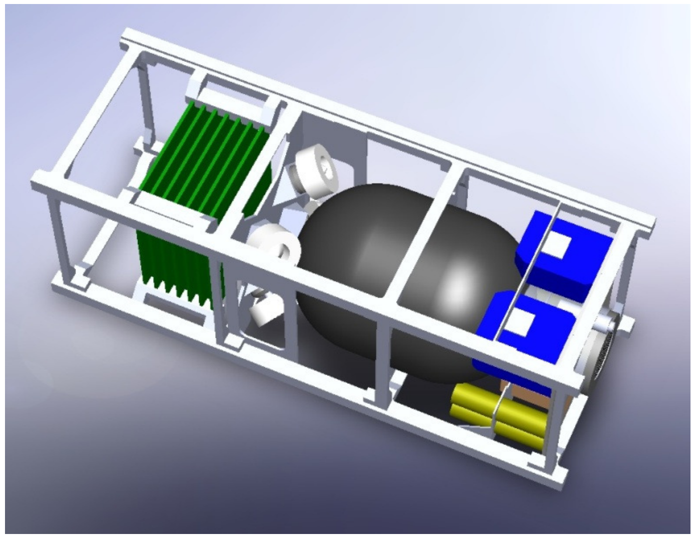

Figure 5. LuMi power subsystem. PPU (blue), PDU (blue), HVEA (blue), battery cells (yellow), and charger (orange) shown with other internal components. Solar arrays and external panels are hidden.

4. Thermal Subsystem

As previously discussed, the thermal considerations for an SEP 3U CubeSat are critical to ensure survivability of the spacecraft. LuMi's small size makes it highly susceptible to large temperature fluctuations during the course of a single orbit. The current model of LuMi assumes no external radiators. The emissivity and absorptivity of the 30x10cm spacecraft body surface facing the sun and the 10x10cm spacecraft body surfaces were set to 0.6 and 0.4, respectively. The emissivity and absorptivity of the three 30x10cm spacecraft body surfaces facing away from the sun were both set to 0.8 and 0.8, respectively. An average body surface temperature of approximately 332K (59°C) during daylight operations and 273K (0°C) during eclipse is expected. Both of these values drop approximately 40K during emergency operations (only necessary subsystems remain powered and operational when a hardware or software error occurs, limiting internal heat generation). Because these temperatures are at the upper limits of the spacecraft subsystem's operable conditions, significant future efforts must be made to dissipate the heat absorbed and generated by LuMi. Considerations to lower these temperatures include the possibility of passive radiator surfaces on the spacecraft body, amongst other options. Additionally, volumetric thermal models to check the temperature of each internal component utilizing ComSol are in progress.

With a larger range of operational and survivable temperatures, the solar arrays are less of a thermal concern. The solar panels are assumed to have a front emissivity and absorptivity of 0.15 and 0.8, respectively, and a back emissivity and absorptivity of 0.85 and 0.85, respectively. The arrays reach a temperature of approximately 346K (73°C) during daylight operations and drop to approximately 258K (-15°C) during eclipse. These values are well within the operational and survivable temperatures of typical solar arrays[10].

6. Additional Considerations (Trajectory, ADCS, Comm., & CDH)

As previously suggested, it is assumed that the LuMi spacecraft will be inserted into a 600km LEO at an inclination of 60°. LuMi will then use its MiXI thruster to begin a spiral transfer to the Moon, which for the purposes of this study was assumed to have an average altitude of approximately 384,000km and an average inclination of approximately 23°. Utilizing Eq. 30 with the spacecraft average mass (approximately 7.29kg), the total transfer time for this trajectory is approximately 431 days. This value agrees with Eq. 4 if we assume constant thrusting during the transfer. This is a relatively long transfer duration and, because of the time spent in the Earth's Van Allan belts, radiation hardening would need to be considered for the LuMi spacecraft. Although not considered during this investigation, beginning the Earth-Moon transfer in a higher energy orbit (GTO, GEO, etc) or providing excess ΔV from the launch vehicle would significantly reduce this transfer time.

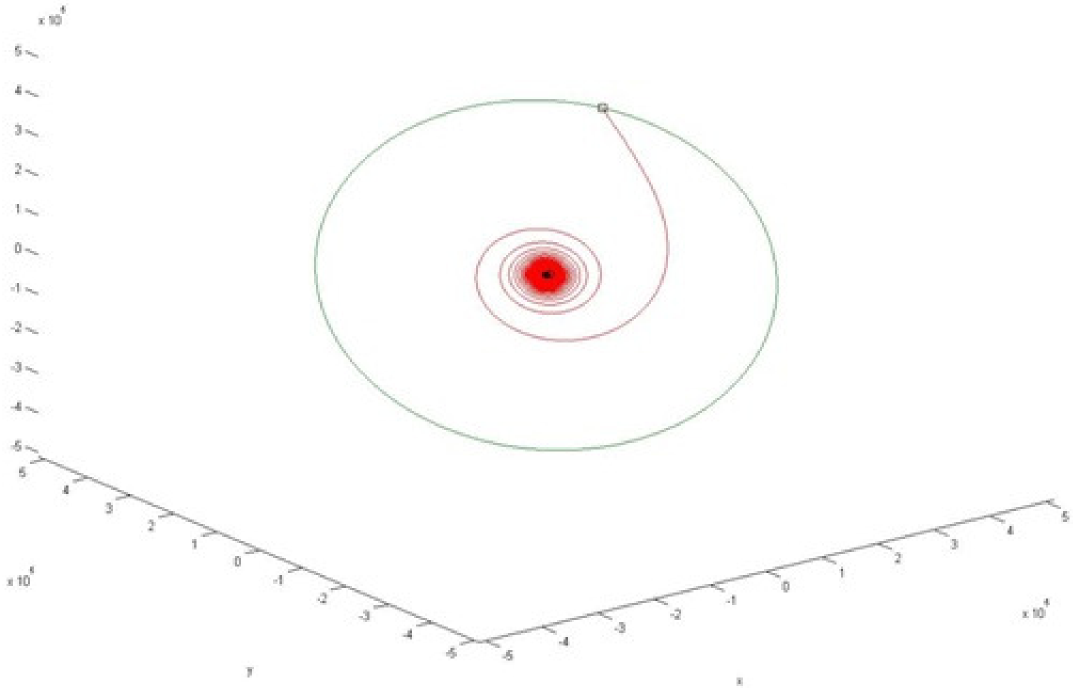

Figure 6. Spiral trajectory flight path. *Hypothetical spiral trajectory of LuMi (red) beginning at a 600km, 60° inclination LEO to the Moon's average orbit path with an altitude of approximately 384,000km and an inclination of approximately 23° (green).*

A spiral trajectory MatLab algorithm developed by Zahi Tarzi at UCLA was utilized to plot the approximate trajectory of the CubeSat from the Earth to the Moon[26]. This algorithm, based on the findings of T.N. Edelbaum, accepts inputs relating to initial orbit orientation and spacecraft properties and generates graphical representations of the spiral transfer[27]. Figure 5 displays the Earth-Moon trajectory of the 8.2kg LuMi spacecraft, utilizing the aforementioned MiXI thruster parameters.

Navigation to the Moon and accurate pointing of the solar arrays will be accomplished by a 3-axis momentum wheel system in conjunction with gimbaled solar array panels. According to unofficial data sheets released by several commercial spacecraft companies, such a system would require less than a 10x10x2.5cm (or 0.25U) section of a CubeSat. Navigation to the Moon can be accomplished partially by utilizing the NASA Goddard-developed GPS-Enhanced Onboard Navigation System (GEONS)[23]. This system is only useful to approximately 50 Earth radii, however, necessitating supplemental celestial navigation for the remainder of the transfer[24].

The LuMi CDH subsystem has been based on the ELFIN spacecraft described earlier for initial mass and power considerations[5,6]. The system employs a series of CDH and EPS (energy power system) circuit boards in addition to two radios and two antennas. This system has a total mass of approximately 0.57kg and a daylight operations power budget of approximately 28W after a 20% margin and 15% contingency for both mass and power are considered. Although this system is known to be sufficient for near-Earth missions, certain components will necessitate modifications or substitutions to ensure that the system will be able to receive and transmit information across the vast distance from the Moon to the Earth.

The complete ADCS, Communications, and CDH systems, including the micro-momentum wheel system and electronics software circuit boards (green) are shown mounted inside the spacecraft in Figures 2 and 5.

5. Summary of Results

The tables below summarize the key performance and design values for the LuMi spacecraft using the relationships discussed in the previous sections. The values below *do not* include any margins or contingencies unless specified in the notes.

Table1. Summary of subsystems. *Subsystem performance and design values for structures, propulsion, power, thermal, and avionics*

Structures Subsystem	Quantity	Mass (ea.)	Material	Notes
Frame	1	200g	6061-T6 Al	
Brackets	20	5g	6061-T6 Al	
Braces	8	5g	6061-T6 Al	
30x10cm Exterior Panel	4	150g	Circuit Board Fiberglass or 6061-T6 Al	Thickness of either 47mm or 63mm
10x10cm Exterior Panel	2	50g	Circuit Board Fiberglass or 6061-T6 Al	Thickness of either 47mm or 63mm
Solar Array Mounting System	2	100g	6061-T6 Al	

Propulsion Subsystem	Mass	Power	Notes
MiXI Thruster	0.25kg	40W	1.36mN thrust; 3000s specific impulse; 50% thrust efficiency; 25% power efficiency
Tankage System	0.18kg	n/a	
Feed System	0.1kg	n/a	
Propellant Req. for Lunar Mission	2.172kg	n/a	Delivers 6950m/s ΔV for 6.2kg spacecraft; includes 10% margin and 15% contingency
Propellant Req. for 4500m/s ΔV	1.35kg	n/a	32° inclination change @ 600km orbit; 25% increase in spacecraft interior usable volume

Power Subsystem	Mass	Power	Notes
Solar Arrays	0.25kg	n/a	Spectrolab UTJ cells (380W/m^2, 450W/kg)
3-Axis ACS	0.2kg	5W	4 momentum wheel system
Battery	0.19kg	n/a	1hr emergency power
PPU	0.2kg	n/a	92% efficiency
PDU	0.1kg	n/a	85% efficiency
HVEA	0.2kg	n/a	97% efficiency
Wiring/Cables	0.2kg	n/a	95% efficiency

Thermal Subsystem	Temperature (Avg)		Notes
Solar Arrays	347.9K	74.9°C	Daylight operations
	258K	-15°C	Eclipse operations
Spacecraft Body	331.5K	58.5°C	Daylight operations
	273.4K	0.4°C	Eclipse operations
	284.2K	11.2°C	Daylight emergency ops
	246K	-27°C	Eclipse emergency ops

Avionics Subsystem	Mass	Power	Notes
ADCS	0.085kg	5W	
CDH	0.05kg	5W	
EPS	0.16kg	5W	
Antennas	0.15kg	2W	single redundancy (2 total)
Radios	0.078kg	1W	single redundancy (2 total)

IV. CONCLUSIONS

The above analysis shows that the proposed LuMi spacecraft is a feasible lunar mission CubeSat. The spacecraft adheres to the overall size requirements of a standard 3U CubeSat while containing all of the required hardware for a lunar mission. The MiXI thruster's small size allows for its use on a CubeSat while its impressive performance provides the necessary propulsion for a lunar mission trajectory. Power and thermal models suggest that the spacecraft is capable of completing a mission to the Moon's surface. These results are further confirmed by the trajectory models developed at UCLA. Additionally, successful navigation to the Moon can be accomplished through the use of new GPS-based NASA software packages and celestial navigation techniques.

V. FUTURE EFFORTS

Future work on this investigation will focus on generating a complete model of the LuMi spacecraft designed specifically for missions to the Moon. Various spacecraft sizes will be considered, from the current 3U CubeSat up to a larger satellite of approximately 20kg. Specific equipment for each of the spacecraft subsystems will be considered and implemented on the final model. The scientific payload capacity will be maximized, offering the greatest flexibility for mission planning. Detailed thermal modeling of the spacecraft's internal components during daylight and eclipse operations will be completed using thermal modeling software.

Trajectory models will be used to show a variety of flight paths for the spacecraft based on initial altitude, inclination, desired mission time, and initial excess ΔV. Radiation effects on the spacecraft will be considered during its traversal of the Van Allan belts. As mentioned in Section III.3, the despinning of the 3-axis momentum wheel system can be accomplished by use of a gimbaled thruster and judicious applications of thrust. The use of such a system on LuMi will be considered, and new mass estimates for the gimbaled system, thruster mounts, and flexible power and propellant lines will be developed.

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