FemtoSats for Exploring Permanently Shadowed Regions on the Moon

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Abstract— Rapid advancement in space exploration has been possible thanks to the accelerated miniaturization of electronics components on a spacecraft that reduces satellites’ mass, volume, and cost. Yet, access to space remains a distant dream as there is growing complexity in what is required of satellites and increasing space traffic. Interplanetary exploration is even harder and has limited possibilities for low-cost missions. All of these factors make even CubeSats, the entry-level standard, too expensive for most, and therefore, a better way needs to be found. In this paper, we propose using FemtoSats, a low-mass, low-cost, disposable solution that exploits the latest advances in electronics and is relatively easy to integrate. FemtoSats are sub-100-gram spacecraft. The FemtoSat concept is based on launching a swarm where the main tasks are divided between the members of the swarm. This means that if one fails, the swarm can take its place and therefore substitute it without risking the whole mission. In this paper, we explore the utility of FemtoSats to explore and map a Lunar PSR. This concept was recognized as a finalist for the NASA BIG Competition in 2020. This is an example of a high-risk, high-reward mission where losing one FemtoSat does not mean the mission is in danger as it happens with regular satellite missions. The work developed here consists of making a generic conceptual design for FemtoSats and apply it to the exploration of the PSR’s. The work results in the development of prototype models and simulations. We are exploring an innovative approach by implementing LASER power beaming to power and keep-alive the FemtoSats in a cold-dark region such as the PSRs. Within the PSR, we want to prove that batteries can be charged through a laser beam. Currently, the FemtoSat design for the PSR is being tested, and the first results look promising. Advances made on this project will take FemtoSat technology one major step closer towards flight readiness and operations and eventually towards applications on ambitious next-generation science exploration missions.

INTRODUCTION

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We see electronics, sensors and actuators getting more compact, reliable, and have increased energy conversion efficiency at current technological trends. This has been driven by rapid advancement and commercial appetite for smartphones, personal computing devices, and smart devices. Advances in these terrestrial areas have now crossed over to space applications. Thanks to Commercial Off The Shelf (COTS) high-reliability, it is possible to use these components in space [1]. These COTS components could survive long-enough on a short high-risk, high-reward mission in extreme environments with sufficient redundancy.

Thanks to miniaturization and COTS (Components Of The Shelf), the current trend towards small satellites is CubeSats. These systems allow engineers to introduce several devices in a single satellite or put a few small satellites in communication to perform a greater mission. However, this concept is still expensive since a typical 3U CubeSat has a mass of 4 kg. According to SpaceX, current launch and service costs are $60,000 per kilogram, and typical CubeSats parts costs have ranged from $40,000 to $85,000, a price far beyond most modest new entrants. For this reason, FemtoSats arose with the WikiSat, [3] and PCBSat, [4], both small satellites with a mass below 100 grams. However, these new ideas presented have thermal insulation problems and are limited by current technologies. We have proposed similar platforms that would morph shape into spheres or attain
wheels and body using inflatables and soft-body shells to overcome thermal concerns [8]. These first two types of FemtoSats propose a flat architecture that is very compact but unable to protect itself from extreme temperature conditions. In 2016, there was an initiative to create a standard for FemtoSats with the SunCube standard, where the introduced 1F model has a mass of 35 grams and a 3F with a mass of 100 grams. This project adopts this cubic architecture from the SunCube standard. FemtoSats introduce whole new application thanks to their low-mass, low-volume, low-cost, and disposability. This includes the use of FemtoSat swarms for space monitoring, multipoint observation, and tracking, in addition to the use of FemtoSats inside of science laboratories to demonstrate mobility in low-gravity environments [9, 10]. The small stowage size of FemtoSat can be a disadvantage for long-range communication, which requires high gains. Inflatables and deployables can be used to deploy large antennas that address these needs [11].

In summary, the main challenge in space exploration is cost. Finding new and cost-effective ways to explore unknown environments such as the Permanently Shadowed Regions (PSR) on the Moon could benefit from using low-cost disposable space hardware with whole new modes of exploration and discovery. For PSR exploration, we need a landed-system, and for this, CubeSats and SunCube FemtoSats are all potential candidates. CubeSats costs are still too high for university teams, high school projects, and individual initiatives. To overcome this accessibility problem, ever-smaller systems such as FemtoSats have appeared; however, some designs such as a Chipsat or Boardsat are more prone to thermal degradation and impose challenges in terms of pointing and 3-axis attitude control. Having an enclosed volume as advocated by the SunCube standard (much like CubeSats) better equips designers to deal with thermal challenges as it is a significant concern for lunar surface missions. For this mission, a 2F FemtoSat (30 mm × 30 mm × 60 mm) is sought, with a mass below 100 grams and a manufacturing cost of around $300.

**MOTIVATION AND CHALLENGES**

This paper presents a new, low-cost, low-mass FemtoSat design to explore a lunar permanently shadowed crater (Figure 1). In addition, we expect to show results on the parts that have been currently tested to prove laser charging and radio localization. The exploration of the unknown drives this project: study the Moon's PSRs and demonstrate new concepts like laser charging and radio localization in the PSRs.

For these goals to be achieved, several problems must be overcome, such as extreme temperatures and uncertainty. Temperatures range from -153°C to 127 °C, and the biggest problem is that it is unknown what the conditions will be when the lander gets to the Moon. Hence, the design must cover all possibilities. In addition, we do not yet know if the components are going to work in the given conditions.

**DESIGN**

As mentioned before, the intention is to design a 2F FemtoSat, and it's composed of four major subsystems: structure, hardware, thermal, and power. For the structure, an aluminum box will contain the hardware and the insulation. In addition, a protector case, made of a high impact absorbent material, will include the aluminum box and the solar panels. Finally, since the case is just a frame, gorilla glass will be added, separating the solar panels from the outside.

The hardware will have a Tinyduino processor board or Tinyzero board, both from Tinycircuits, [5]. In addition, the system includes an IMU, a 433 MHz radio, and a protoboard. To protect the solar panels, they are made smaller than the aluminum box. This makes a frame for the solar panels of 21 mm × 51 mm in the long faces and 21mm × 21mm in the short faces. The aluminum box outer dimensions are 24.8 mm × 24.8 mm × 54.8 mm and have a thickness of 0.5 mm. A preliminary mass, volume, and power budget are given later in Tables 4 and 5.

The intended MLI thickness is 0.5 mm for the thermal insulation, and adding aerogel is under consideration. The size of the hardware elements is 20 mm × 20 mm. This leaves an almost 3 mm gap inside for wiring and other possible elements. We will use solar panels designed for space applications on the power system side and will account for the increase in current and voltage due to temperature.

This section is divided into three parts and includes experiments, analysis, and communications. In addition, the experiments include ballistic impact tests, power systems, and radio localization testing.
Impact Survival

The first experiment, impact survival, is developed based on the conservation of energy model as there is no air resistance on the Moon (Figure 2). FemtoSats will be ballistically launched within a range of 2 to 20 meters and from a height of 1.9 meters. They need to be designed so that they survive the impact with the lunar surface.

Figure 2. Sketch for the energy model. The lander is represented in red, and the FemtoSat is described in yellow. \( \alpha \) represents the mean slope between the lander and the FemtoSat and \( \delta \), the launching angle. \( V_0 \) represents the initial velocity of the FemtoSat, and \( h_0 \) represents the vertical distance from the FemtoSat to the lander base.

The system has one DOF and parameters, \( \delta \) and \( \alpha \) respectively, hence optimization can be done, resulting in just one local minimum or maximum. In addition, since the number of equations and variables are the same, there is an analytical expression that calculates the launch angle, \( \delta \) as a function of the parameter \( \alpha \):

\[
\sin 2\delta (20 \tan \alpha + 1.9 + 20 \tan \delta) = 20
\]  

(1)

This expression provides a launch angle that minimizes the impact energy. However, there is a general expression for similar types of experiments:

\[
\sin 2\delta (x_0 \tan \alpha + z_0 + x_0 \tan \delta) = x_o
\]  

(2)

Where \( x_0 \) and \( z_0 \) represent the desired range or the horizontal flying distance of the FemtoSat and the height measured from the lander’s base from where it is launched (same as \( h_0 \)), respectively.

Solar Array Design

The part presented intends to estimate what \( V_{MP} \) and \( I_{MP} \) are needed for the solar panels from DHV technologies. The equations used for these calculations are as follows:

\[
I_{MP}^{STD} = \frac{\Phi}{n_1} - \frac{\delta I}{\delta T} (T - T_{STD}) A
\]  

(3)

\[
V_{MP}^{STD} = \frac{V_{MP}}{n_2} - \frac{\delta V}{\delta T} (T - T_{STD})
\]  

(4)

Where \( n_1 \) is the number of cells in parallel and \( n_2 \) is the number of cells in series. Results for just one cell are optimum since the solar cell could work in both hot and cold conditions. The desired voltage ranges between 4.64 and 4.99 V, and the current would be 28 mA. However, there is no cell with those capabilities in the market due to the voltage being too high. For that reason, there is a need to use at least two cells in series.

Thermal Design

The thermal analysis (Figure 3) provides an idea of the type of insulation needed. It shows if passive methods are enough or if the mission also requires the use of active techniques to regulate the temperature. The advantages of the former are their simplicity and the lack of power consumption.

Figure 3. An electrical analogy for the thermal analysis.

On the other hand, the latter allows the designers to control the FemtoSat temperature precisely. For the thermal model, an electrical analogy is used. The system is composed of the hardware (\( E_{bh} \)), the MLI (\( E_{bMLI} \)), the aluminum box (\( E_{bAl} \)), the solar panels (\( E_{bSP} \)), the glass (\( E_{bg} \)), the Lunar surface (\( E_{bsoil} \)), and the outer space (\( E_{b2} \)). The following resistances are defined as follows:

\[
R^i = \frac{1 - \varepsilon_m}{A_m \varepsilon_m}  
\]  

(5)

\[
R^e = \frac{1}{A_h F_{p-q}}  
\]  

(6)

\[
R^c = \frac{\varepsilon_m}{A_m \varepsilon_m}  
\]  

(7)

Where \( R^i \) indicates the internal resistor of the element due to radiation, \( \varepsilon_m \) indicates the emissivity of the material or element “m,” \( A_h \) represents the radiation-emitting area and \( F_{p-q} \) shows the sight facto between element “p” and element “q.” \( R^e \) indicates the radiation resistor between two elements connected, while \( R^c \) represents the resistor due to conductivity. The thickness of the element “m” is indicated
by $t_m$ and $k_m$ is the thermal conductivity coefficient. The resistor values used are shown in Table 1.

**Table 1. Resistors values.**

| Resistance Variable | Value (Ohms) |
|---------------------|--------------|
| $R_0^e$             | 0            |
| $R_1^e$             | 376          |
| $R_2^e$             | 0            |
| $R_3^c$             | 0            |
| $R_4^c$             | $3.58 \cdot 10^{-4}$ |
| $R_5^e$             | 64.52        |
| $R_6^e$             | 193.57       |
| $R_7^e$             | 2436.2       |
| $R_8^c$             | 59.6         |
| $R_9^c$             | 0.47         |
| $R_{10}^e$          | 1102.3       |
| $R_{11}^e$          | 0            |

The equation for the resistors calculations can be found in the Appendix. Properties of the materials used can also be found in the Appendix. Finally, to calculate $P_{dis}$, which stands for dissipated power, we need to analyze what is the output power of the battery and what is the power consumption of the elements.

$$
P_{dis} = P_{dis}^{bat} + P_{dis}^{proc} + P_{dis}^{IMU} + P_{dis}^{333} \quad (8)
$$

$$
P_{dis} = 10.99 \, mW
$$

Now that all the numeric values are obtained, it is just a matter of solving the circuit in Figure 3. Two temperatures are essential to get: the hardware temperature and the temperature of the solar panels.

$$
\frac{E_{bMLI}-E_{bh}}{\sum_{i=0} R_i} + P_{dis} = 0 \quad (9)
$$

$$
P_{dis} = \frac{E_{bMLI}-E_{bsp}}{R_3+R_4} = \frac{T_{MLI}-T_{SP}}{A_{MLI}R_{MLI}+A_{Al}R_{Al}} \quad (9)
$$

$$
E_{bsp} \left( \frac{1}{R_3} + \frac{1}{R_4} \right) = \left( \frac{E_{bMLI}}{R_3} + \frac{E_{bAl}}{R_4} \right) + P_{dis} \left[ 1 + \left( R_3 + R_6 \right) \left( \frac{1}{R_3} + \frac{1}{R_4} \right) \right] \quad (10)
$$

$$
E_{hi} = \sigma T_i^4 \text{ (for radiation)} \quad (11)
$$

Where $\sigma$ is Boltzmann's constant. The system, although it seems coupled, can be solved easily in the order (10), (9), (8). The outer space temperature can be considered 0 K, thus, $E_{hi}$ is zero. Since we are designing for cold conditions, the regolith's temperature is -153 °C (122.15K).

**Radio Frequency Localization**

The great difficulty is that it is impossible to use GPS tracking as it is done on Earth. If there were a satellite constellation providing GPS localization, it would be much easier. However, this problem may be overcome by using the Decawave DWM 1000 (Figure 5). This device allows localization with 10 cm precision. Power consumption is comparable with the rest of the hardware, and it consumes between 86 mW to 220 mW. The device also fits in the 2F FemtoSat (23 mm x 13 mm x 2.9 mm). Finally, the operating temperature range is also in accordance with the rest of the devices (-30ºC to 85ºC).

**Figure 5. DWM 1000 device from Decawave. Source [6].**

**Overall System**

The overall architecture and packed FemtoSat is shown in Figure 6 and 7, respectively. The FemtoSat circuitry comprises several layers, each 25 mm x 25 mm in the area and packed together as shown.
RESULTS

This section will gather the results obtained from the design process and then compare the experiments' measurements.

Impact Survival

Table 2. Impact results for a 12º mean slope.

| Variable     | Value  |
|--------------|--------|
| \(v_0\) (m/s) | 4.97   |
| \(t\) (s)    | 5.01   |
| \(\delta\) (º) | 36.45  |
| \(h_0\) (m)  | 4.25   |

These results were validated with two numerical analysis. The former implemented optimization, and the second one just calculated the energy for every combination of \(\delta\) and \(\alpha\). With these results now it is easy to compute the energy per mass unit and the equivalent height on Earth for testing,

\[
H_E = \frac{E_m}{mg_E} \cdot SM = 2.18 \text{ m}
\]

Where \(H_E\) is the height on Earth, \(E_m/m\) is the mechanical energy per mass unit, \(g_E\) is Earth's gravity, and \(SM\) is the safety margin, and it has been considered 10%. Figure 8 represents the mechanical energy variation with the mean slope and confirms that most critic case is the one with a bigger slope.

Figure 8. Minimum Theoretical mechanical energy at \(\delta=36.54\) and for varying \(\alpha\).

Solar Array

Results with four cells, two in parallel and two in series, are good but still present some problems. It was discovered that the greater the number of cells, the more distortion between hot and cold conditions exist; therefore, it is essential to minimize the number of cells in series.

Table 3. Maximum power current and voltage for different radiation and temperature

| \(\Phi\) | \(T\) | Imp | Vmp |
|--------|------|-----|-----|
| 1322   | 107  | 72.75 | 3.1 |
| 4444   | 107  | 244.59 | 3.1 |
| 4444   | -153 | 81.63 | 6.01|
| 1414   | 107  | 77.82 | 3.1 |

From observing Table 3, it would be a great advantage to have a \(V_{MP}\) of almost 6V in cold conditions and at least obtain 3.7V in warm conditions, which is the nominal value of the battery and would allow the battery to charge up to a decent percentage. If we get to raise the panels' temperature to -98ºC in cold conditions, we get the desired condition.

Thermal Analysis

Using a standard MLI and using all the thickness available, of 2.3 mm, with a thermal conductivity coefficient of 0.01, the dissipated power to achieve the temperature requirements is 290 mW. This results in panel temperatures of the following:

\[
T_{SP} = -67 \text{ ºC}
\]
\[ T_h = -19^\circ C \]

We would like to have the solar panels at around -95\(^\circ\)C. In addition, the battery lasts 1.5h under these conditions. Solutions to improve this are by decreasing the thermal conductivity of the MLI and probably combining it with aerogel. The optimum result could be achieved by having a thermal conductivity coefficient of 0.001, which is not that far from current results and maybe doable by combining aerogel and MLI.

\[ T_h = -18^\circ C \]

\[ T_{sp} = -93^\circ C \]

In this optimum result, the dissipated power required is 155 mW, and if using 150 mA, the working time is three hours. If changing the battery to one with 500 mA, the mission would be ten hours.

### Mass Budget

| Group    | Element                        | Mass (g) | Volume (mm\(^3\)) |
|----------|--------------------------------|----------|--------------------|
| Hardware | Processor board (TinyZero)     | 1.4      | 1160               |
|          | IMU                            | 1        | 2044               |
|          | Radio + Antenna                | 1.41     | 2044               |
|          | ProtoBoard                     | 0.85     | 2044               |
| Power system | Battery (500 mA, 3.7V)    | 9.3      | 5890               |
|          | Charger Controller             | 0.54     | 466.2              |
|          | Solar Panels 4x                | ~27      | 6854               |
| Structure | TPU case                       | 9        | 54000              |
|          | Box (aluminum)                 | 9        | 33704              |
| Insulation | MLI                            | TBD      | 12509              |
|          | Aerogel                        | ~0       | TBD                |
| Total    |                                | 54       | 54000              |

### Power Budget

| Group       | Element                        | Temperature Range (\(^\circ\)C) | Current (mA) | Voltage (V) | Power (mW) |
|-------------|--------------------------------|---------------------------------|--------------|-------------|------------|
| Hardware    | Processor board (TinyZero)     | [-40, 85]                       | 4            | 2.7 – 5.5   | 10.8 – 22  |
|             | IMU                            | [-40, 85]                       | 4.6          | 3.5 – 5     | 16.1 – 23  |
|             | Radio + Antenna                | [-40, 85]                       | 18.5 (RX)    | 3.5 – 5     | 65 – 93    |
|             |                                |                                 | 30 (TX 13 dBm)|            | 105 – 150  |
|             |                                |                                 | 85 (TX 20 dBm)|            | 298 - 425  |
|             | ProtoBoard                     | Unspecified                     |              | 3.7         | TBD        |
| Power system| Battery (500 mA, 3.7V)         | [-20, 130]                      | 30-500       |             | 1.85 Wh    |
|             | Charger controller             | [-40, 85]                       | 15-500       | 4.2 – 6     | -          |
|             | Solar Panels                   | Unspecified                     | 65           | 4.66        | -          |
DISCUSSION

Outer space missions are complex, and therefore a lot remains to study about FemtoSats, but some relations can be drawn from this preliminary study. If FemtoSats are to be ejected from a lander, we must design a trajectory that minimizes the impact energy since the equipment is sensitive. The FemtoSats need thermal protection, and thus, studying and selecting the proper insulation materials is essential. For this reason, maybe it is a better option to substitute the aluminum box for a different material with a lower thermal conductivity coefficient.

This paper has shown that the power system design is coupled with thermal design. For that reason, it is recommended to make a second-order analysis or use finite element methods to have more accurate results. Moreover, this gains importance since even minor changes could make the difference between having sufficient available volume or not. Transmitting and receiving data using the radio consumes a significant amount of power; hence their duty cycle needs to be minimized when experiments are carried out.

When considering launch angles, for a given maximum mean slope of 12º provided by Colorado School of Mines [7], the ejecting angle minimizes the mechanical energy is 36.45º and reaches 20 meters horizontal range. This means that the initial speed at which the FemtoSat is launched is almost five meters per second, and therefore the system that launches the FemtoSat will have to be designed in accordance with this value. To test if the solar cells and the hardware inside would survive, the FemtoSat needs to be dropped from a 2 m height, and accounting for a 10% safety margin; it is recommended to drop it from 2.2 m.

For use to effectively charge the solar panels in both hot and cold conditions, we need two cells in series per solar panel so that the final output voltage at maximum power in standard conditions is 3.98 V. At the same time, a well-insulated system that can keep the solar panels at a temperature of -95 °C or less is required. This also allows dissipating less power to warm hardware and battery. Finally, radio power consumption is high; therefore, the duty cycle should be a minimum.

CONCLUSIONS

Overall, we show it is feasible to deploy FemtoSats from the lander into the PSR to provide exploration data. For the FemtoSat deployment problem, the ejecting energy should be calculated for the maximum mean slope expected. In addition, the problem presents one degree of freedom; therefore, an analytic solution can be found. Current commercial solar panels are too sensitive to temperature because it is not recommended to use them in space if our battery charging voltage is limited. To avoid this problem, the main solution is to minimize the number of cells in series. Power analysis and thermal analysis are highly coupled; therefore, higher-order analysis or finite element methods should be used. However, the linear method used here provides a quick insight into the requirements. The radio duty cycle should be minimized to maximize mission life since its power consumption is very high.

APPENDICES

A. Thermal properties

| Table 6. Different materials emissivity. |
|-----------------------------------------|
| Material   | Emissivity |
|------------|------------|
| Aluminum   | 0.04       |
| Glass      | 0.94       |
| Copper     | 0.95       |
| White paint| 0.97       |
| Black paint| 0.75       |

| Table 7. Different elements thermal conductivity coefficient, k. |
|-----------------------------------------|
| Material   | Conductivity |
|------------|--------------|
| Aluminum   | 209.3        |
| Glass      | 0.8          |
| MLI        | 0.01         |

| Table 8. Sight factor between components. |
|-----------------------------------------|
| Components | Sight Factor |
|------------|--------------|
| Fh−MLI     | 1            |
| FSP−g      | 1            |
| Fg−soil    | 0.3          |
| Fg−2       | 1 − Fg−soil  |

| Table 9. Component area (mm²). |
|--------------------------------|
| Component | Area (mm²) |
|-----------|------------|
| Ah        | 2658.4     |
| AMLI      | 6254.64    |
| AAl       | 6666.24    |
| ASP       | 5166       |
| Ag1       | 1071       |
| Ag2       | 3025       |
| Asoil     | ∞           |
B. Resistors detailed calculation

Table 10. Internal and external radiation resistor and thermal conductivity resistor.

| Resistor  | Value                                                                 |
|-----------|-----------------------------------------------------------------------|
| $R_0^i$   | $\frac{1 - \varepsilon_h}{A_h \varepsilon_h} = 0$                   |
| $R_1^i$   | $\frac{1}{A_h F_{h-MLI}} = \frac{1}{A_h} = 376$                     |
| $R_2^i$   | $\frac{1 - \varepsilon_{MLI}}{A_{MLI} \varepsilon_{MLI}} = 0$      |
| $R_3^i$   | $\frac{t_{MLI}}{A_{MLI} k_{MLI}} = 0$                                 |
| $R_4^i$   | $\frac{t_{AI}}{A_{AI} k_{AI}} = 3.58 \cdot 10^{-4}$                 |
| $R_5^i$   | $\frac{1 - \varepsilon_{SP}}{A_{SP} \varepsilon_{SP}} = 64.52$     |
| $R_6^i$   | $\frac{1}{A_{g1} F_{g-2}} + \frac{1}{A_{g2} F_{g-soil}} = 2436.2$  |
| $R_7^i$   | $\frac{1}{A_g \varepsilon_g} = 59.6$                                 |
| $R_8^i$   | $\frac{t_g}{A_g k_g} = 0.47$                                        |
| $R_9^e$   | $\frac{1}{A_{g2} F_{g-soil}} = 1102.3$                               |
| $R_{10}$  | $\frac{1 - \varepsilon_{soil}}{A_{soil} \varepsilon_{soil}} = 0$   |

C. Hardware dissipated power calculation

\[
P_{\text{dis}} = P_{\text{bat}}^{\text{dis}} + P_{\text{proc}}^{\text{dis}} + P_{\text{IMU}}^{\text{dis}} + P_{\text{433}}^{\text{dis}}
\]

\[
P_{\text{dis}} = I^2 R_{\text{bat}} + V_{\text{proc}} l_{\text{proc}} + V_{\text{IMU}} l_{\text{IMU}} + V_{\text{433}} l_{\text{433}}
\]

\[
P_{\text{dis}} = 27.1 \times 0.027 \times 0.2 + 0.4 \times 4 + 0.4 \times 4.6 + 0.4 \times 18.5 = 10.99 \text{ mW}
\]

\[
\frac{1}{R^*} = \frac{1}{R_g + R_g} + \frac{1}{R_{10}}
\]
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BIOGRAPHY

Álvaro Díaz-Flores Caminero earned his M.S in Aerospace Engineering from Universidad Politecnica de Madrid (UPM) in 2020, an M.Eng in Aerospace Systems from the University of Arizona in 2020, and his B.S. in Aerospace Engineering from Universidad Politécnica de Madrid (UPM) in 2018. He is pursuing his Ph.D. in Aerospace Engineering at the University of Arizona. Álvaro has interned with the Spanish Air Force and developed maintenance and design tools. He is currently working on the FemtoSat project to explore the Permanently Shadowed Regions of the Moon at the Space and Terrestrial Robotic Exploration (SpaceTREx) Laboratory in collaboration with the Colorado School of Mines. His research interests include space exploration, swarms, high-speed docking, aircraft conceptual design, aerodynamics, and structures.

Leonard Dean Vance is a Ph.D. candidate in Aerospace and Mechanical Engineering at the University of Arizona, and a retired Senior Fellow from Raytheon Missile Systems. His 33 year career at Hughes Aircraft and Raytheon Missile Systems includes Systems Engineering lead, Chief Engineer and Program manager for a variety of forward leaning projects, including the LEAP kinetic kill vehicle, the AIM-9X sidewinder missile, Kinetic Energy Interceptor and program manager for the SeeMe imaging microsatellite. He holds a masters of Engineering from Harvey Mudd College, six patents and is currently in his 3rd year of the Ph.D. Program at the University of Arizona.

Himangshu Kalita received a B.Tech. in Mechanical Engineering from National Institute of Technology, Silchar, India in 2012. He is presently pursuing his Ph.D. in Mechanical Engineering from the University of Arizona in the Space and Terrestrial Robotic Exploration (SpaceTREx) Laboratory. His research interests include dynamics and control, space robotics, machine learning and automated design.

Jekanthan Thangavelautham has a background in aerospace engineering from the University of Toronto. He worked on Canadarm, Canadarm 2 and the DARPA Orbital Express missions at MDA Space Missions. Jekan obtained his Ph.D. in space robotics at the University of Toronto Institute for Aerospace Studies (UTIAS) and dis his postdoctoral training at MIT’s Field and Space Robotics Laboratory (FSRL). Jekan Thanga is an assistant professor and heads the Space and Terrestrial Robotic Exploration (SpaceTREx) Laboratory at the University of Arizona. He is the Engineering Principal Investigator on the AOSAT I CubeSat Centrifuge mission and is a Co-Investigator on SWIMSat, an Airforce CubeSat mission concept to monitor space threats.