

Georgia Institute of Technology

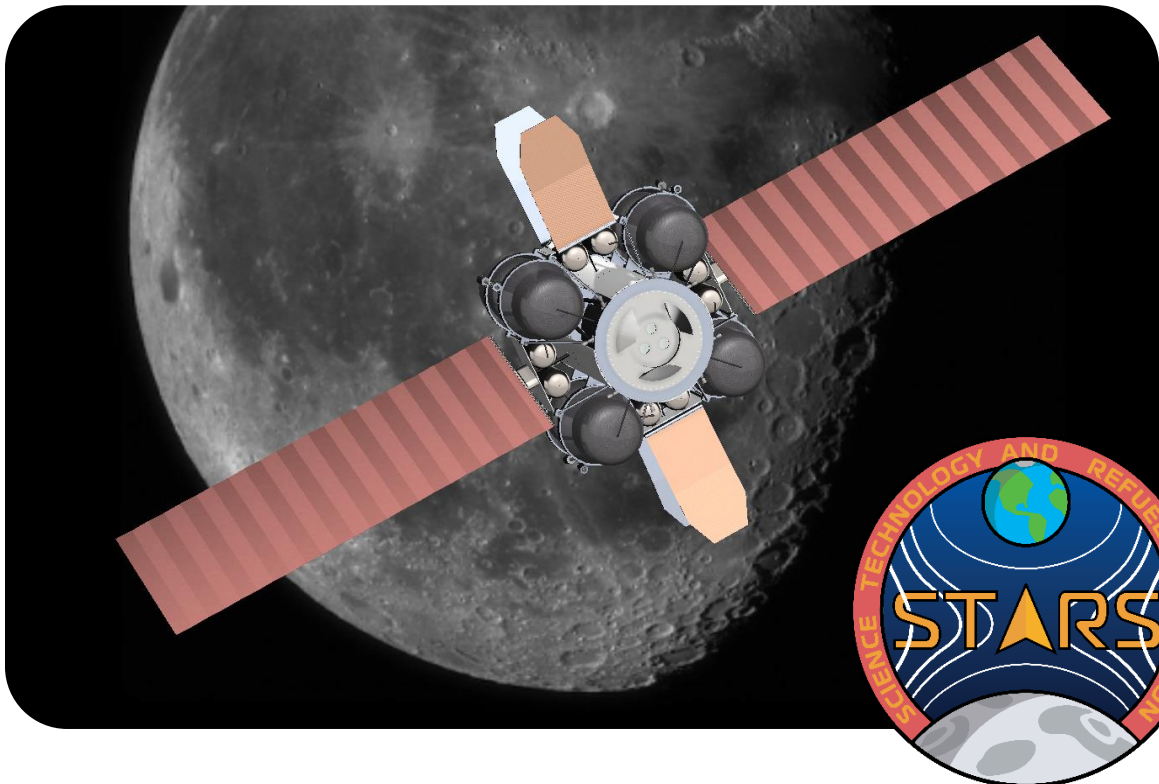
Science, Technology, and Refueling Station (STARS)

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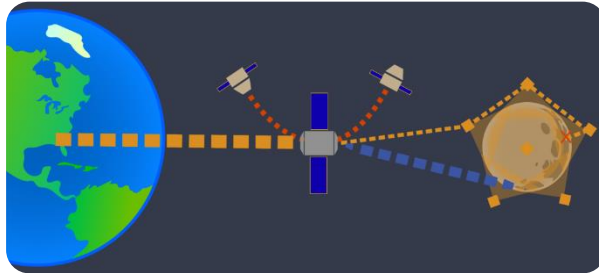
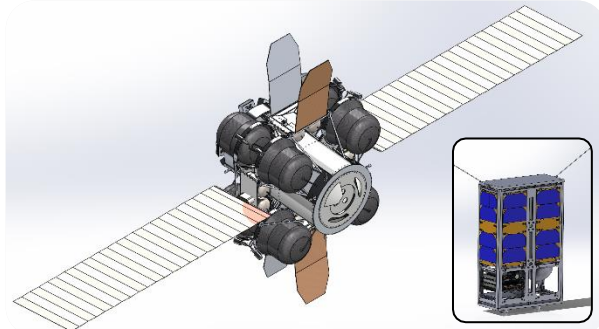
STARS Multi-Use L1 Platform



Multi-use Platform at L1

Objectives & Technical Approach:

A Multi-use platform at the Earth-Moon Lagrange point 1 will allow for refueling for medium-sized spacecraft in cislunar space as well as communication infrastructure for the moon. The platform will be in a halo orbit around the L1 point, and vehicles traveling to and from the moon will dock with the spacecraft, where propellant will be transferred through a pressure fed system. CubeSats deployed from the platform will serve as a mesh network around the moon to provide global coverage for surface-to-surface communication, keeping a low data rate link with the L1 platform while the platform will maintain a high data rate with the lunar south pole to assist future lunar operations.



Key Design Details & Innovations:

Using LMP-103S green monopropellant increases the environmental sustainability of the mission, reduces vehicle development costs, and contributes to technological innovation.

The world's first demisable composite propellant tanks with the surface tension mechanism were used for the propellant storage.

Additionally, using phased array antennas for the communication system allows the need for gimbals to be eliminated; reducing structural complexity.

Schedule:

The major milestones include successful identification of mission objectives and requirements in pre-phase A by 2022, testing within tolerance levels for phases C & D by 2029, initial spacecraft operation performance in phase E by 2040, and platform disposal completion in Phase F by 2041.

Cost:

Total proposed budget: \$2,509,460,940.

STARS Business Case

Stakeholders, Customers, & Lunar Market

The STARS L1 Platform, or Science Technology and Refueling Station, will catalyze the commercialization of space by providing spacecraft refueling capabilities and a communications link for lunar surface-to-surface operations. STARS' primary budget benefit comes from its ability to refuel spacecraft, substantially reducing the propulsion costs of future missions while also extending total mission durations. The platform also sends COMeT, Cislunar Orbital Mesh Telecommunication, CubeSats to lunar orbits as a communications link, which will increase exploratory capabilities and safety for future missions to the Moon. Overall, the STARS platform will incentivize an increased number of lunar missions from NASA and private space companies due to the potential cost reductions of future missions from propulsion mass and cost savings.

RASC-AL Theme Overview

The STARS L1 Platform addresses the “Multi-use Platform at L1” prompt for RASC-AL. The platform meets the theme by offering refueling and communication while also using new green fuel propellant. The platform also meets the operational date and lifetime requirements.

Configuration & Architecture

Operation Concept, Elements, & Phases

The Earth-Moon L1 point serves as an ideal position for STARS to achieve its mission objectives. The L1 point's fixed procession relative to the moon provides constant line of sight between both Earth and the Lunar south pole. This is a very desirable trait for STARS' communications goals – allowing constant communication for future NASA operations. Additionally, the L1 point provides access to any approach into lunar space. This means that refueling can be conducted for a spacecraft in (or bound for) nearly any lunar orbit. From these benefits, orbit about the L1 point is an optimal final destination from which the STARS platform can operate.

But first, STARS' must be delivered to L1. A concept of operations for the pre-operations STARS timeline can be found on the left in Figure 1. The first pre-operational phase of STARS' mission is a launch onboard a SpaceX Falcon Heavy Expendable launch vehicle. STARS was initially designed for the NASA Space Launch System but was downsized to utilize a launcher better suited for the cost/value of STARS' mission. The loss in value from decreasing refueling capabilities is easily offset by the much cheaper cost of launch – making the Falcon line a much better value option. The Falcon Heavy expendable's 15000kg TLI mass budget is enough to place the all-up mass of STARS into its L1 orbit. STARS is designed to utilize the maximum IML of the falcon heavy fairing diameter. STARS leaves plenty of vertical height within the fairing – using only 3.65m of the 12.5m high fairing. Some of the remaining mass and volume could allow for lunar payload ridesharing.

After entering low earth orbit, the next pre-operational phase begins. In this phase, the Falcon Heavy's upper stage performs the trans-lunar-injection burn to set the STARS platform on a lunar fly-by. The fly-by operations both provide a trajectory re-direction towards the insertion into L1 and an opportunity to stoke public interest in lunar exploration using a similar live-stream method to the current Orion capsule. After the fly-by, the Falcon's upper stage will finalize STARS' trajectory to L1, and then freely-return to Earth for disposal. Once in L1-space, STARS' onboard propulsion can begin managing its south-facing halo orbit.

In this third stage, STARS can begin releasing the COMeT network satellites towards their final frozen orbits around the moon. These eight CubeSats, built on an EnduroSat 6U structure, are designed to form a mesh network providing near-constant coverage to 80% of the lunar surface. They act to relay surface data from any covered point on the moon to each other – eventually feeding data back to the STARS platform

at L1. The COMeT satellites are orbiting so that there always exists a path within communications range allowing real-time (non-periodic) relaying from any covered lunar region.

Once these satellites are deployed, STARS can begin its operational refueling and high-data rate targeted communications phase. Phased array antennas are used to communicate both between Earth and the Lunar south pole. This choice allows communications to take place regardless of STARS' position within its halo orbit, and without requiring complex mechanical gimbaling of traditional high-gain antennas. When a high earth or lunar satellite, or a heavy mission bound for the moon, needs fuel resources, it can rendezvous with STARS dock autonomously, be resupplied, and then depart. Based on its available resources, STARS should be able to perform 3 large (800kg) refueling masses, or smaller ones.

STARS Configuration & Reasoning

The combination of both the STARS system and COMeT satellites allows one launch to establish both the high-speed communications required for NASA's future missions to the lunar south pole and global lunar coverage for any and all assisting missions. STARS is the primary platform for the mission; and its refueling and communications are built on pre-existing technologies to ensure reliability and feasibility.

STARS carries over 6000 liters of LMP-103S green propellant within its eight COBHAM high-performance composite propellant tanks. Green propellant was used because it is a simple monopropellant requiring lower complexity systems and safe to those who work with it. Eight radially mounted composite storage tanks (seen on the right in Figure 1) are used to limit the effects of propellant sloshing, and to provide redundancy in the case of tank or pneumatic damage/failure. The propulsion and propellant transfer systems are pressure fed, utilizing a PVG helium pressurant tank for each propellant tank. The lower complexity of pressure-fed systems makes them suitable for the 10-year operational time of the STARS platform. Propellant transfer is carried out autonomously through the ESA's Automatic Transfer Docking Port. This system also feeds 16 LMP-103S thrusters capable of providing translation and rotation about any axis. These low thrust engines exist to perform the low ΔV burns necessary for orbital maintenance and to dump momentum from the ADCS.

The ADCS relies on several attitude determination and attitude control components. For absolute attitude determination, STARS utilizes two pairs of star trackers and sun sensors. The star trackers act as the primary inertial reference due to their accuracy. These externally mounted components are placed to guarantee that at least one faces away from the Sun, Earth, and Moon at all times. The sun sensors exist as backup, allowing a rough attitude to be maintained in the event of a system failure – keeping orientation critical subsystems (such as the solar panels) pointed properly while the problem is sorted. For relative attitude determination, STARS uses a Honeywell IMU core. Integrated accelerometers and ring-laser gyroscopes are used for high-rate integrated state estimates. These measurements work in combination with the four Rockwell-Collins Space Wheels to provide pointing control. Momentum wheels are used to maintain a level of gyroscopic stiffness for the STARS platform. This helps reject adverse loads due to docking/refueling operations.

With the communications direction determined by the Earth-Moon axis, the solar panel axis can be set normal to the moon's orbital plane. This allows the single-axis gimbaled solar panels to track the sun's apparent location – negating most off-angle solar angle effects. This allows the combined 12m² STARS GaAs solar arrays to provide nearly 2400W of constant power. This power is managed with a fully regulated PPT bus - feeding both the high-power draw for the communications system and ADCS, as well as the charging of the four onboard batteries totaling 14.5kWh. These act as a power supply during periods of solar occlusion and during intermediate operations where the solar panels may not be fully utilized.

Component Selection & TRL Analysis

A table of specific subsystem components and architectures is shown in **Appendix B – Technology Readiness Level**. Down-selection criteria will be discussed in detail in each individual section. The primary result is that component selection across the board has a high TRL. While there are likely low TRL systems

in development now that will be ready by STARS' launch, using pre-existing high TRL systems ensures long-term reliability within STARS. STARS is built for reliability and redundancy. Surviving a 10-year mission requires flight-verified hardware. The most substantial lower TRL system is the propulsion engines. Choosing not to utilize hydrazine limited the number of commercially available engines. However, as hydrazine falls out of favor and green monopropellants become prioritized, modern monopropellant engines are sure to be developed that will suit STARS' needs. Additionally, careful consideration was done to ensure that both STARS and COMeT utilize as many pre-existing components as possible. Not only with this decrease the development and manufacturing costs, but it will accelerate the timeline, as less rigorous in-house testing must be done.

System Budgets & Estimates

Utilizing a combination of bottom up and statistical analysis methods, mass, volume, power, and data budgets can be estimated for the STARS platform. A summary of these analyses can be found in Table I. For mass, STARS is constrained within the 15000 kg lunar payload of the Falcon Heavy launcher. Within this weight limit, STARS is expected to utilize 11,372 kg after margin. The remaining weight opens the mission to a possible ride-sharing opportunity, offering roughly 3600 kg of available lunar payload. Within STARS, the propellant makes up the greatest mass. This propellant amount is a combination of the propellant for refueling operations and the propellant used to maintain STARS' Halo orbit. The propellant mass has a lower margin than the remaining components, as this is a well-defined value based off the volume of the storage tanks and the density of the fuel.

Volume breakdown for STARS is focused on two regions, the internal payload region, and the Falcon Heavy Fairing. STARS' internal cavity (blue in the image) houses multiple subsystems, which have volume allocations as shown. The cylindrical bounding box matching STARS greatest dimensions takes up roughly 58 cubic meters of the 151.5 cubic meter falcon heavy fairing.

Similarly, the power budget could be broken down by subsystem as follows. As expected, the high-powered communications system that allows STARS to achieve its desired throughputs consumes the most power. Further power breakdown will be discussed in more detail in sub team sections. The communications system also drives most of the data budget – with sensors from other subsystems making up a nearly negligible amount of data throughput compared to the 400 Mbps communications link.

Subsystem Feasibility & Findings

Trajectory

A Halo orbit around L1 was chosen to satisfy a key mission objective - communications. The Halo orbit affords the platform an uninterrupted view of both the near side of the Moon and the Earth, allowing for continuous communication with assets on the lunar surface as well as with the COMeT mesh network.

The specific Halo orbit selected has a radius of 3500 km, with a period of 12 days. These parameters were chosen to find a balance between the period-keeping delta-V requirements, while also allowing STARS to be accessed from a wider range of trajectories by incoming Lunar spacecraft. The orbit also possesses a worst-case occlusion of only 0.793% per orbit, meaning that solar power is exceptionally feasible for this mission. This also allows us to carry much smaller and lighter batteries, meaning we can carry more refueling propellant with us to support additional vehicles.

The COMeT CubeSats orbit the Moon in frozen orbits, so minimal orbit maintenance is required, and the satellites can last for the required 10 years. To obtain global coverage, it was calculated that 8 satellites would be needed at an altitude of 800 km; the frozen orbit was then calculated based on eccentricity and required inclination from the article, "Lunar Frozen Orbits" (Folta, Quinn, 2006). Two orbital planes would be used with 4 orbits in each plane, where pairs of CubeSats share the same orbit, half an orbit apart from each other. These orbits have an inclination of 52 degrees, an eccentricity of 0.6119, different arguments of periapsis of 90 and 270 degrees, and an orbital period of 13 hours. This arrangement is an estimate for

continuous, global coverage of the Lunar surface in addition to a line of sight from at least one CubeSat to the STARS platform at all times.

Structure

STARS utilizes a radially symmetric geometry, where the radial and axial distributions of its mass are roughly equal. This distribution makes implementation of a 3-axis control scheme simpler. The primary structural element of the platform is a cylindrical truss composed of Aluminum Alloy 2219, which is a well-known heritage material for spaceflight structures. This alloy possesses a high strength to weight ratio which meets our needs effectively. The truss was sized for the worst-case scenario loading where all the spacecraft mass was placed directly on top of the truss. The largest loads occur during launch due to the acceleration of the Falcon Heavy, with a peak axial acceleration of 6 g's and a peak lateral acceleration of 2 g's. These load cases were taken directly from the Falcon Heavy User's Guide. The estimated factors of safety for the loads of 6g axial loading and 2g lateral loading were found to be 2.836 and 73.427, respectively. The COMeT CubeSat structure is an off-the-shelf 6U CubeSat frame, similar to those from EnduroSat.

Communications

The STARS and COMeT vehicles seek to create a dedicated communications infrastructure on the lunar surface in order to enhance the capabilities of future lunar operations, public or commercial. The overall communications infrastructure can be split into two different links: a high data rate link with emphasis on lunar south pole coverage and a lower data rate link dedicated to coverage of the entire lunar surface. These two links were designed in consideration of future lunar operations; NASA's focus on future infrastructure at the lunar south pole was the driving factor behind a high data rate link created and the exploration of the far side of the Moon by future rovers or landers was the driving factor behind the lower data rate link. The designed architecture has been proven feasible and will be explored in this section.

To begin with the simpler architecture of the high data rate link, tracking a signal as it travels through this link explains the design best. A signal is transmitted from a base or vehicle from the lunar south pole to STARS. STARS then receives this signal and amplifies it before transmitting it to a ground station on Earth. This process is the downlink of the high data rate link, the reverse process is the uplink. This extra step in the communications link that is STARS may seem unnecessary at first glance, however the presence of STARS allows for future lunar south pole missions to allocate less resources towards their communications systems, potentially reducing their mission costs. Highlighting some key design points, X-band frequencies (7.2-8 GHz) were chosen for their capability of carrying large amounts of data, which is necessary if STARS is to serve as the intermediate link. The STARS platform has an uplink data rate of 150 Mbps and a downlink data rate of 400 Mbps, so the X-band frequencies chosen to allow for this data rate to be achieved. Additionally, STARS uses phased array antennas for their reduced complexity compared to parabolic antennas that would require gimbals for pointing. These phased array antennas have a similar size and gain to the parabolic antennas, but instead use phase differences between individual antennas on the array to point the signal. The link budget for STARS to Earth can be found in **Appendix A – Link Budget**.

To ensure that the link between STARS and Earth is completed effectively, the Near-Earth Network (NEN) was chosen. NEN is run not only by NASA but also by other private and national organizations such as Kongsburg Satellite Services and the South African Space Corporation. The majority of the stations in the NEN support X-band frequencies and after plotting the ground stations in Google Earth, we determined that a NEN, Xband capable ground station would always be in view of STARS. This would make our platform less susceptible to communication blackouts, making it the perfect choice. Additionally, the G/T values of each of the Xband capable ground stations met the G/T requirements of our link budget.

Similar to the high data rate link, the lower data rate link can best be explored by following a signal through this link. Starting from a transmission from a small vehicle located anywhere on the lunar surface, the signal is received by a satellite in the COMeT mesh. Assuming the vehicle on the surface is on the far side of the Moon, most likely the receiving satellite is not in direct line of sight with the STARS vehicle. The receiving

satellite will then transmit the signal across the mesh to a satellite that is in line of sight with STARS. The STARS vehicle then transmits the signal to a receiving ground station on Earth in a similar manner to the high data rate link. As seen, this link has a much more complex architecture than the previous, so a lower data rate of 50 Mbps both up and down was chosen to account for this complexity and smaller size of each individual COMeT satellite. However, for the scale of the missions sent to the far side of the lunar surface, 50 Mbps will be a substantial increase in the data that can be sent to Earth. On each COMeT satellite, two dipole antennas are used to allow for omnidirectional antenna coverage, reducing the pointing and fuel requirements on each individual satellite. Additionally, X-band frequencies are also used in this link to maintain consistent frequencies with the high data rate link. The link budget for COMeT to STARS can be found in **Appendix A – Link Budget**.

Propulsion and Refueling

The propulsion unit for STARS enables the station to maintain its halo orbit and serves as an active attitude control system. In comparing electric and chemical propulsion, a liquid monopropellant propulsion system was ultimately chosen because it provides a much higher thrust (4.45 - 2668 N) with a duration of minutes to hours, requires much less power, and integrates with a refueling system with less complexity. The system allows for pulsing of thrusters and steady burns to maintain its orbit position and adjust its attitude. The driving propulsion requirement was the Delta-v budget from trajectory analysis, mass estimates based on the launcher capabilities, and each subsystem's mass allocation analysis. The total propellant mass for the main propulsion and refueling units were calculated to be about 4700 kg and 2600 kg respectively.

The first step in system sizing was choosing a propellant. Hydrazine was the first choice due to its common use in missions since the 1960s. However, it is an extremely toxic chemical that imposes high safety risks and costs during testing stages. The ADN-based LMP-103S green monopropellant was chosen instead of hydrazine due to its higher specific impulse and density, a potential increase in the mission's environmental sustainability, reduction of vehicle development costs, and contribution to technological innovation. Being a newly developed propellant, it doesn't pose high risks for mission delay or failure since it has been qualified for space flight and thus is suitable for STARS. LMP-103S has a higher TRL compared to other green monopropellants, and the alternative AF-M315E propellant was not selected due to its high ambient flame temperature (2166 K) compared to hydrazine (1170K) and LMP-103S (1903.15 K) which reduces the lifetime of engines and propulsion system. Since LMP-103S has a higher ambient flame temperature than hydrazine the catalyst bed and other materials of the engine must qualify and pass the fatigue and thermal load tests. To mitigate the risks of the catalyst bed sintering entirely before the end of the mission, a catalyst metal with higher melting and sintering temperatures than those of iridium must be selected. Rhenium was determined to be an adequate substitution for iridium since it has melting and sintering temperatures of 3453 K and 2071.8 K. Since the performance and compatibility of the rhenium and LMP-103S for prolonged missions have not been vigorously explored, numerous tests must be designed and performed to ensure that the maturity of the technology satisfies all the requirements.

For the fuel transfer system, we have also selected the LMP-103S. Considering the fuel will be used for future lunar missions, the propellant selection will be critical for both this mission and the space industry. Since fuel transfer system has limited history and TRL, many experiments will be needed to increase the maturity. Especially for this huge scale, there has not been a single mission that has performed a fuel transfer and plenty of tests will be required. To limit the cost of those experiments, we have restricted our propellant options to green monopropellant and excluded hydrazine which has the highest TRL within any propellant. When we look at the specific impulse, density and TRL, there are mainly three green monopropellant candidates which are LMP-103S, AFM315M and HNP221. LMP-103S has a good performance characteristic and it has flown with several space missions such as the PRISM and SkyBox which justifies the maturity and TRL if the propellant for our mission. Due to the 10-year operational life, the system needs to be simple with minimal number of moving parts and to satisfy these requirements a pressure-fed system was selected for the propellant transfer.

Next step was to determine the total number of engines needed to satisfy thrust, Delta-v and attitude control requirements. Assuming that the duration of a burn for station-keeping and orbit maintenance is around 35 seconds and the total number of burns per day was around 0.56 with a safety factor of 20%, the total thrust per burn was estimated to be around 183 N. An additional 5 N were added to account for the desaturation of the reaction wheels. To broaden thrust capabilities of STARS thrusters with magnitudes of 5N, 10N and 22 N were integrated into the design. The number of thrusters was determined with the thrust per burn needed divided by the average thrust provided by a single engine (12N). Thus, the total number of thrusters was calculated to be 16 with the following magnitude distribution: four 22N thrusters, eight 10 N thrusters, and four 5 N thrusters. The thrusters were positioned such that individual thrusters will have control over multiple axes.

Finally, the tanks were sized by searching for commercially available tanks for propellant and helium. The carbon-fiber propellant tanks found have a surface tension mechanism that mitigates sloshing problems and allows the fuel to stay at the bottom of the tank in microgravity conditions. The ideal gas law was used to check that 8 estimated helium tanks with maximum operating pressure of 4496 psia can supply the engine feeding pressure of 275psia, based on the commercial hydrazine engines, where P_f is the feeding pressure, V_f is the total volume of propellant tanks, and V_i is the volume of the selected helium tanks.

$$V_i P_i = V_f P_f \quad (1)$$

Overall, 8 propellant tanks were needed to store all the propellant and 8 tanks to pressure the propellant. The last step in finalizing the propulsion system design was constructing the propulsion schematic shown in Figure 1.

ADCS

The internal components of the ADCS include the following: four Rockwell Collins TELDIX RSI 68-75/60 Space Wheels, two Redwire Space Hi-REL Sun Sensors, two Ball CT-2020 Star Trackers, and one Honeywell Miniature IMU. Momentum wheels were chosen for gyroscopic rigidity to allow for maximum possible light to reach the gimbaled solar panels; modified TELDIX fly wheels were used by the James Webb Space Telescope, which has a similar mass and orbit to the STARS system. A star tracker and sun sensor are positioned on the side facing the sun as well as the side opposite. The CT-2020 star trackers have a 33° standard sunshade and can track at high rates during slew maneuvers. This positioning allows one star tracker to always have a complete field of view; and the sun sensors act during modes with lower pointing requirements. Table – specifies the masses and power of the ADCS system, with the thrusters considered a part of the propulsion system. The momentum wheels consumed the most power; however, at steady state nominal speed, three wheels consume 60 W, much lower than the 270 W required for the worst case of maximum torque at nominal speed. The wheels were also the heaviest components, as expected, followed by the star trackers. These sensors were chosen for their accuracy of less than 1.5 arcsec, which accommodates the large uncertainty in docking pointing requirements.

ODH

To begin our onboard data handling requirement calculations, we started by looking at the worst-case situation for both COMeT and STARS. Starting with COMeT, we determined that the worst-case scenario for a single CubeSat would be that it should be capable of storing all the data that it would otherwise communicate if it was in view of STARS. There are 8 total CubeSats that make up COMeT but in this worst-case calculation, we assume that there is only one CubeSat. Using the CubeSat's orbit, we performed a calculation to determine that time that one CubeSat would be in the dark aka not in view of STARS. By multiplying this time by the throughput rate of the CubeSat and applying an overhead percentage, we came to the value of 32GB of storage per CubeSat. Now moving on to STARS, the worst-case scenario would be when it would need to store all the data stored on all of the CubeSats as well as some data from the lunar bases. To estimate the data storage from the lunar south pole base, we used the data throughput to storage ratio for the CubeSat to relate the lunar base's throughput rate to a data amount. After applying overhead

to the data requirements of the CubeSat and lunar base, the STARS data storage requirement comes out to be 2256GB.

Once we have the data throughput requirements (from link budget) and the data storage requirements (from the above calculations), the ODH architecture can then be selected. Starting with the CubeSat, the EnduroSat on board computer was selected as it was already sized to the 6U CubeSat platform and could process the required communication throughput rates (can support up to 125Mbps). Regarding STARS, the RAD750 onboard computer was chosen as it both supports a 400Mbps throughput and has been flight proven. Similarly, the SpaceWire PCIe board was chosen to interface the antennas with the CPU as it can support X band frequencies and is a tried-and-true system.

Thermal Control

Passive systems will be used for nominal operations with actives systems taking effect during eclipse periods and emergencies. Heat piping and insulation through the central systems will be used to distribute heat as needed. Louver systems will be implemented on radiators for emergency heat venting. The main body of STARS will be coated with white paint while propellant tanks will be coated with a black paint to maintain temperatures. Thermal analysis of this configuration indicates that STARS will experience temperatures between –120 and –7 degrees Celsius with black paint reaching temperatures of 39 degrees Celsius. Patch heaters will be utilized for active heating on necessary systems with cartridge heaters used for propellant heating for burns. This system should allow function over the 10-year lifespan even with degradation of coating materials experienced in this time frame

Timeline, Cost, & Risk

Development Timeline

Because STARS aims to fill infrastructure needs in cislunar space and is mostly comprised of off-the-shelf components, the development timeline can be accelerated since instruments and experiments do not need to be designed. Knowing this, the timeline allots 2 years for Phases A & B, 4 years and 3 quarters for Phases C & D, 10 years for Phase E, and 2 quarters for Phase F. The largest concern for this timeline is the development of green monopropellant thrusters capable of providing the required thrust to control STARS. The next step of STARS mission development would include the creation of a technology advancement plan to ensure the thrusters would be mature enough by the required launch date.

Estimated Budget

The STARS budget is developed using version 8 of the Unmanned Space Vehicle Cost Model (USCM8) in conjunction with launch costs from the Falcon Heavy Launcher. Subsystem masses from Table I were used in the USCM8 to calculate a total component cost of \$2.3 Billion, and the launch costs totaled to \$190 Thousand when adjusted for 2030 inflation. Total projected costs of STARS sum up to \$2.5 Billion, with component breakdown shown in Table III. This cost can be lowered substantially with ridesharing, mass reduction, faster development, and additional cost-effective materials in the future.

Program Operational Risk

STARS' largest operational risks come from its defining characteristics. Namely, propellant sloshing induced instability, propulsion system failure due to novel green monopropellant thrusters, and consequential single-event upsets (SEU) aboard any mission vehicles. These risks were identified early in the design phase and mitigations were integrated in the design process. To mitigate sloshing effects, the number of propellant tanks was increased, the tank sizes were decreased, and flexible plastic baffles were introduced in each tank. For the propulsion system, redundant thrusters and piping were included to ensure necessary performance from the system for the duration of the mission. As for the risk of SEU's, radiation hardened components were selected along with redundant hardware.

Tables & Figures

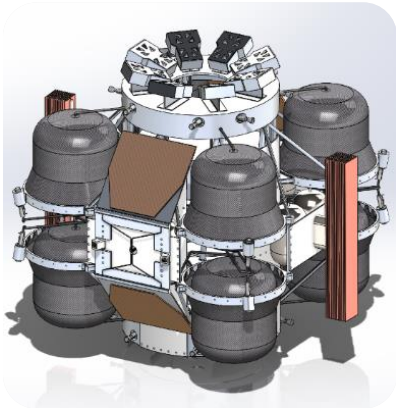
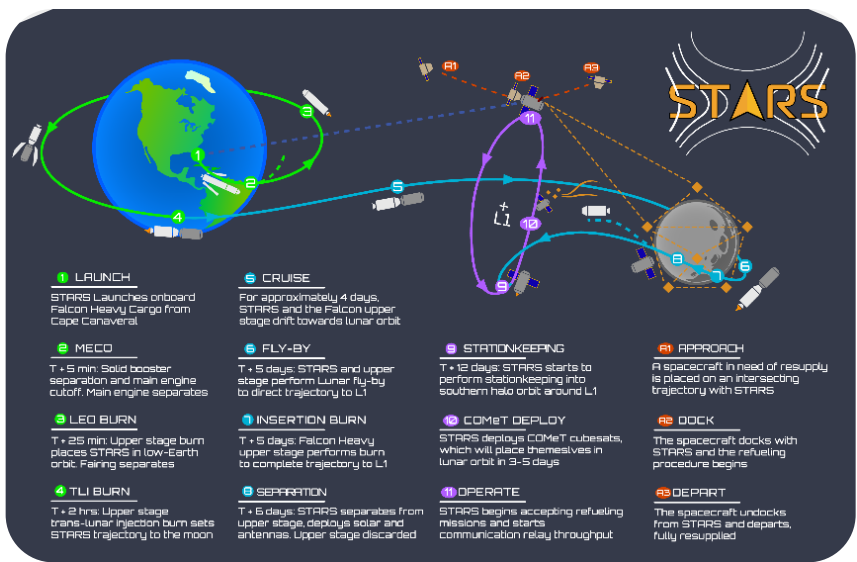


Figure 1: STARS OV-1 Concept of Operations and Configuration

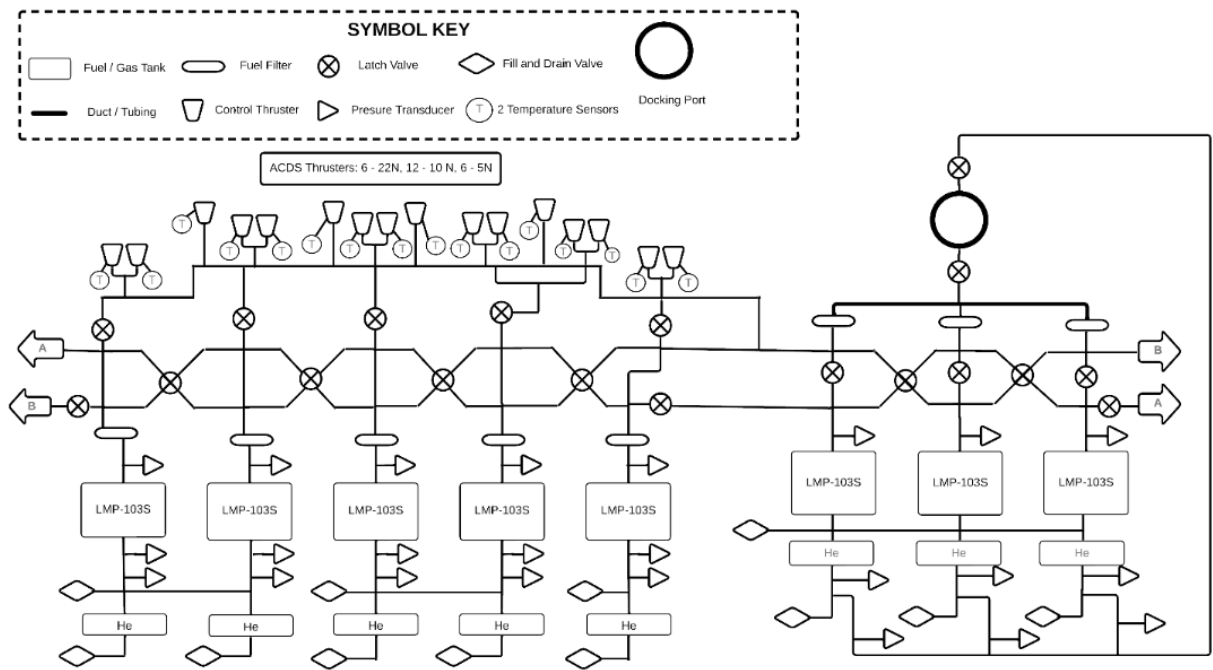


Figure 2: STARS Propulsion Diagram

Table I: STARS Mass, Volume, Power, and Data Budgets

<i>Subsystem</i>	<i>Mass Estimate</i>	<i>Margin</i>	<i>w/ Margin</i>		<i>Segment</i>	<i>Volume</i>
Communications	540.19	30%	702.25	kg	<u>STARS Internal Volume</u>	4.63 m ³
Propellant Mass	4696.4	15%	5400.86	kg	- Communications	0.79 m ³
Structure + Mechanisms	2099.67	30%	2729.57	kg	- Power	0.93 m ³
Thermal Control	423.96	30%	551	kg	- ADCS	0.50 m ³
Power	390	30%	507	kg	- Tubing	0.07 m ³
OBP	140	30%	182	kg	- Thermal	0.31 m ³
ADCS	56	30%	72.8	kg	<u>Remaining Volume</u>	2.03 m ³
Propulsion	592.35	30%	776.95	kg	STARS Prop. Storage	6.18 m ³
Other	330	30%	429	kg	STARS OML Volume	58.05 m ³
TOTAL	9273.91		11351	kg	Falcon Fairing IML	151.5 m ³
Falcon Heavy M _{TLI}			15000	kg		
Rideshare Capacity			3648	kg		

<i>Subsystem</i>	<i>Power Estimate</i>	<i>Margin</i>	<i>w/ Margin</i>		<i>Subsystem</i>	<i>Data (kbps)</i>
1. ADCS	356	30%	462.8	W	<u>Communications</u>	<u>440011</u>
2.1 UHF TX1	1000	30%	1300	W	Data Throughput	
2.2 UHF TX2	3.5	30%	4.55	W	Performance Sensors	
3. CDH	29.6	30%	38.48	W	<u>Thermal & Power</u>	<u>22</u>
4. Propulsion	416	30%	574.6	W	Performance Sensors	
5. Battery Charging	23	30%	29.9	W	<u>ADCS & Propulsion</u>	88
6. Thermal	80.6	30%	104.8	W	Sensors	
Total	1803.9		2514.24	W	Data Filtering	110
					Total (kbps)	440231

Table III: STARS Budget Breakdown

Subsystem	Cost in 2010 \$K	Cost in 2022 \$K	Cost in 2030 \$K
Structure	\$ 644,883.12	\$ 881,555.23	\$ 1,119,575.14
Thermal Control	\$ 164,830.41	\$ 225,323.17	\$ 286,160.43
ADCS	\$ 23,587.20	\$ 32,243.70	\$ 40,949.50
EPS	\$ 32,600.10	\$ 44,564.34	\$ 56,596.71
Propulsion	\$ 61.10	\$ 83.52	\$ 106.07
TT&C	\$ 22,916.00	\$ 31,326.17	\$ 39,784.24
Communications Payload	\$ 74,160.00	\$ 101,376.72	\$ 128,748.43
Integration, Assembly & Test	\$ 140,213.41	\$ 191,671.73	\$ 243,423.10
Program Level (Comms Satellite)	\$ 152,192.42	\$ 208,047.03	\$ 264,219.73
Aero Ground Equipment	\$ 80,293.99	\$ 109,761.88	\$ 139,397.59
Component Development Cost		\$ 1,825,953.49	\$ 2,318,960.94
Falcon Heavy Launch Cost		\$ 150,000.00	\$ 190,500.00
Total STARS Cost		\$ 1,975,953.49	\$ 2,509,460.94

Appendix A – Link Budget

STARS to Earth Budget	Value	Units
Uplink Frequency	7.20	GHz
Gateway Terminal		
Antenna Diameter	4.00	m
Beamwidth	0.73	deg
Transmit Gain	47.00	dBi
Transmit Power	300.00	W
Backoff and Line Loss	-4.00	dB
EIRP, Gateway	67.77	dBW
Propagation Range	24759929.27	km
Space Loss	-197.47	dB
Atmosphere Loss	-15.23	dB
Net Path Loss	-212.70	dB
Satellite Antenna		
Coverage Area	2.03	deg ²
Antenna Efficiency	70	%
Receiver Gain	41.52	dBi
Line Loss on Satellite	-2.00	dB
Received Carrier Power, C	-109.06	dBW
System Noise Temperature	27.90	dB-K
G/T	13.62	dB/K
Receiver C/No	93.65	dB-Hz
Data Rate	79.72	dB-Hz
Available Eb/No, Uplink	13.93	dB
Link Margin, Design	3.43	dB
Link Margin, Required	3.00	dB
Downlink Frequency	8.00	GHz
Satellite Antenna		
Coverage Area	2.03	deg ²
Antenna Efficiency	70	%
Antenna Gain	41.52	dBi
Satellite Transmit Power	1000.00	W
Backoff and Line Loss	-2.00	dB
EIRP, STARS	67.77	dBW
Propagation Range	24759929.27	km
Space Loss	-198.38	dB
Atmosphere Loss	-15.23	dB
Net Path Loss	-213.61	dB
Gateway Terminal		
Antenna Diameter	4.00	m
Beamwidth	0.66	deg
Receiver Gain	47.91	dBi
Backoff and Line Loss	-4.00	dB
Received Carrier Power, C	-97.82	dBW
System Noise Temperature	27.90	dB-K
G/T	20.01	dB/K
Receiver C/No	96.86	dB-Hz
Data Rate	83.98	dB-Hz
Available Eb/No, Uplink	12.88	dB
Link Margin, Design	3.18	dB
Link Margin, Required	3.00	dB

COMeT to STARS Budget	Value	Units
Uplink Frequency	8.00	GHz
STARS Terminal		
Coverage Area	21.21	deg ²
Antenna Efficiency	70	%
Antenna Gain	31.34	dBi
Satellite Transmit Power	5.00	W
Backoff and Line Loss	-4.00	dB
EIRP, STARS	34.33	dB
Propagation Range	64912.83	km
Space Loss	-146.76	dB
Atmosphere Loss	0.00	dB
Net Path Loss	-146.76	dB
COMeT Antenna		
Antenna Efficiency	70	%
Receiver Gain	0.091	dBi
Line Loss on Satellite	-2.00	dB
Received Carrier Power, C	-114.34	dBW
System Noise Temperature	25.00	dB-K
G/T	-24.91	dB/K
Receiver C/No	89.26	dB-Hz
Data Rate	75.44	dB-Hz
Available Eb/No, Uplink	13.82	dB
Link Margin, Design	3.32	dB
Link Margin, Required	3.00	dB
Downlink Frequency	8.00	GHz
COMeT Antenna		
Antenna Efficiency	70	%
Antenna Gain	0.091	dBi
Satellite Transmit Power	3.50	W
Backoff and Line Loss	-2.00	dB
EIRP, COMeT	3.53	dB
Propagation Range	64912.83	km
Space Loss	-146.76	dB
Atmosphere Loss	0.00	dB
Net Path Loss	-146.76	dB
STARS Terminal		
Coverage Area	21.21	deg ²
Receiver Gain	31.34	dBi
Line Loss on Satellite	-4.00	dB
Received Carrier Power, C	-111.88	dBW
System Noise Temperature	25.00	dB-K
G/T	6.34	dB/K
Receiver C/No	87.72	dB-Hz
Data Rate	74.95	dB-Hz
Available Eb/No, Uplink	12.77	dB
Link Margin, Design	3.07	dB
Link Margin, Required	3.00	dB

Appendix B – Technology Readiness Level

Purpose	Component	TRL
Propellant Storage	COBHAM 772L Tank	8
Pressurant Storage	PVG Family 40-75L Tank	9
Propellant	LPM-103S Monopropellant	8
Engines	22N, 10N, & 5N Thrusters	6
Power Generation	Fourth-Junction GaAs Panels	7
Power Storage	Lithium Ion Cells	9
IMU	Honeywell Miniature IMU	9
Star Tracker	Ball CT-2020	7
Sun Sensor	RAD-HARD Hi-Rel	9
Reaction Wheels	RC TELDIX RSI 68-75/60	9
Docking Port	ESA Auto Transfer Port	9
COMeT Launchers	Motorized Lightband	9
Communications	Phased Array Antennas	9
Cooling	Louver System	9
Flight Computer	RAD750 Computer	9

Appendix C – Citations

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