ILNSS : NETWORK FOR POSITION ON LUNAR SURFACE AND INTERPLANETARY PROTOTYPE

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Abstract: Due to overpopulation, human beings need more resources. The space colonization idea has been established but there are some redundancy budgets for deep space exploration with a single purpose components. So ILNSS will be a satellite network to determine selenographic location of spacecraft from entry till takeoff from the planet including determine and broadcast signal to be used in location service for astronauts, rover, helicopter, specific site receiver using VLBI technique and Radar for ground station deployed. ILNSS not only supports multi-purpose missions for deep space missions, but also will be a prototype for reduced risk, cheaper, high precision and utilities for deep space projects in the future. This paper will include ILNSS LUCIUS operation which is the first mission for ILNSS on Lunar. Preparing for Lunar gateway before 2030.

1. INTRODUCTION

In 2030 of NASA's Moon to Mars[1] mission, progress and achievement of human beings are still discovering and exploring a new planet that can live and being a new habitat, which was detected by space exploration. One of the biggest trends is colonization on another planet due to overpopulation and insufficient amount of supply.

Colonizing on a planet or space colony is a challenge task for humans due to various environmental parameters, which humans can live in called 'Habitat Zone'. This zone has been explored by observing a planet or space area previously. Some parameters have to take a sample and precisely examine in space or are brought to a ground station, which has to land a spacecraft for human or rover exploration on the surface causing the cost of operation higher and taking more risks.

Landing on the surface is one of the most stressful parts of the mission. The latest rover landed on Mars using the EDL[2] method which is based on image processing and mathematical ideas. However, image processing could fail if a weather disaster on the surface makes the spacecraft unable to land if the surface is windy, foggy, smoked, snowed, and other technical problems with unseen terrain. Also, EDL has many components to detach before the rover successfully lands, can damage already placed buildings, previously landed spacecraft, and colonized sites. Also, EDL still has an error due to calculation and can only calculate when in a stage of landing the rover and is unuseful for multiple missions in the long term, and also not have any benefits in the core operation of the mission.

So we have an idea to create a satellite network as an orbiter to determine the position in various coordinates while using planets as references for helping spacecraft landing with lower distance error rates. To make the landing more precise, lower cost, reusable for multiple missions, durable compared to cost, and can be used as a commercial or prototype of position and navigation system for the future space colony missions.

2. MISSION OBJECTIVES

- The main objective of ILNSS is to create interplanetary prototype isolated satellite systems for spacecraft landing and space civilians navigating on planet surface or space colony. By creating a space colony we need to explore, experiment, and plan. So, we have to create a globalized system to make it easier to determine the position of the events happening on the planet, make an experiment, and devise into real space applications.
- To be an experimental way to create an isolated location-based service satellite systems on another planet. By using selenographic[4] coordinate system and cartesian coordinate system to determine the location of system's users.
- To use an experiment from ILNSS as a prototype or using ILNSS itself as a commercial location service satellite system for future space civilians in space colonies, due to over population on the earth.

3. CONCEPT OF OPERATIONS INCLUDING ORBITAL DESIGN EXPERIMENTAL CONCEPT AND SETUP

ILNSS Mission has got many factors and operation phases to make a satellite network with a reusable deployer, fast, reliable, and accurate satellite systems. ILNSS network would setup and operate as circular orbit from proximal altitude depends on shape and radius of target planet's surface. ILNSS use six orbit parameters for Kepler's orbit[3] to create an orbit, satellites should-be position by calculation which also include another operation useful for ILNSS orbital design.

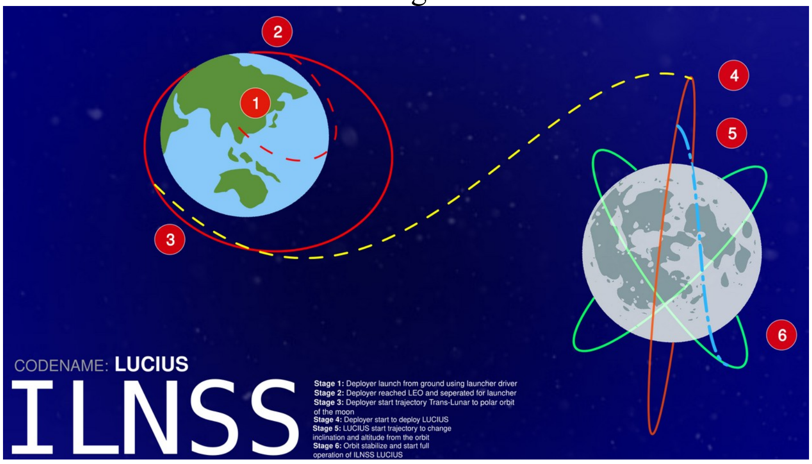

Figure 1. Overall mission and concepts for ILNSS (Object Not to scale)

The proximal use case in prototyping is ILNSS on Lunar with operation LUCIUS which also have parameter as inclination 60° degrees from Lunar equator, eccentricity is 0 because of circular orbit, RAAN is up to the number of satellite constellation in orbit divide to 360° is the different of RAAN degrees need to shift to each orbit in LUCIUS we use of 0°, 120°,and 240° RAAN, Args of periapsis is as same as 0° degrees, and true anomaly is up to the satellites in each orbit divide to 360° is the difference need to shift for placing satellite keep distance equally each other in the orbit in this case we use 7 satellites in each orbit

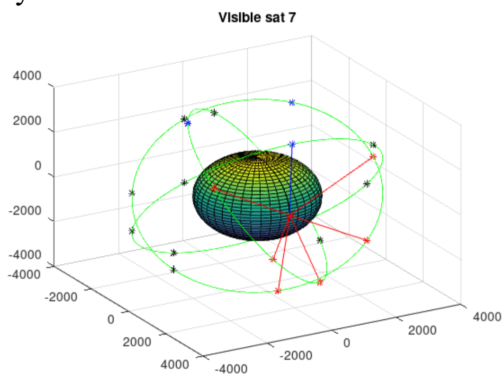

Figure 2. Orbit constellation of ILNSS LUCIUS[1]

Most of ILNSS LUCIUS mission is calculated using software GNU Octave, numerical computation software, every of orbit constellation, time dilation, coordinates systems provided and conversion source codes[2] can also be free to use under GPL3 software licenses of GNU Octave.

3.1 Concept of Operation

ILNSS mission transfer from the altitude of launcher driver velocity to deployable altitude of the target planet surface, after the orbit has stabilized using reaction wheels and propulsion. ILNSS Lucius broadcasts signal in S-Band at 2,245 MHz as center frequency which provides location of satellite and time at signal broadcasted. After he signal broadcasts through space which provide radius between starting point t_0 and end point at the moment t, speed of signal broadcasted is a constant to electromagnetic waves

1 Full animated visualization included in https://bit.ly/ILNSS-Calculated

2 The visualization and source codes is included in https://bit.ly/ILNSS-Calculated

which is equal to 299,792,458 m/s which we can find the relation of time and distance as linear movement as equation (1).

$$s = v\left(t - t_0\right) = \left(299792458\right)\left(t - t_0\right) \tag{1}$$

Which meant the speed of wave multiplied by time traveled through space to give a distance between the signal broadcast satellite and the receivers, this would cause a sphere of distance, d. Along with multiple satellites, the systems which can reach the receiver at the same time will cause multiple spheres of distances to intersect with previous received one. Which represents the sphere of intersection as Figure 3.

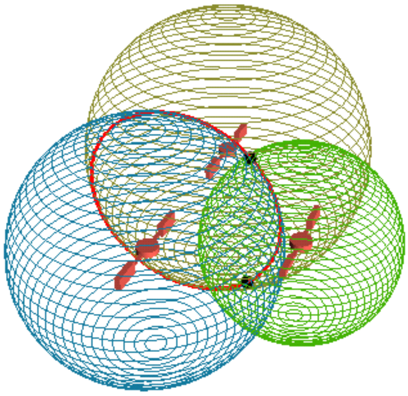

Figure 3. Sphere of intersections

Each intersection can determine the location of each satellite in the sight with a multiple radius from different location which can find the relation between the satellites, signal and receiver location by using equation to calculate an only one of intersections from all which included time dilation signal between satellite in space and receiver on the planet ground. The risk of the mission is when the satellite can't determine the exact location itself, with this, if we use only 4 satellites to determine the location will cause many problems to calculate the position of the receiver and cause more than 10 meters up to a couple of miles away. The problems of signal distortion can be solved by using more than 4 satellites to compute much more accuracy to make unique calculation of position by using the sphere of intersection represented in Figure 1.

3.1.1 Using VLBI for determining location

For determining location we used VLBI technique. Satellite will send 2 pulses of waves with time data in it. After receivers receive these pulses we use that to determine location by using the delay of the receivers receiving data because of their location. The time delay of receiver (T) and distance between two receivers (B) we can use trigonometry to find the angle of the satellite compared to the earth surface as equation (2).

$$\theta = \cos^{-1}\left(\frac{cT}{B}\right) \tag{2}$$

$$d = ct \tag{3}$$

To determine distance (d), Using the time came from the signal to calculate time that wave uses for travel from satellite to earth by the equation expressed as equation (3). When c represented the speed of light which is constant to 299,792,458 m/s, t represented the time that wave used to travel from satellite to earth , and d represented the distance between satellite and earth.

3.1.2 Sat location verifying using VLBI

To verify satellite location. Asuuming the location travelled has a coordinate latitude and longitude of zero. When the time passes the angle will increase, so, we can convert coordinates from moon to earth location and confirm satellite location using DSN VLBI array.

3.1.3 Providing ILNSS signal

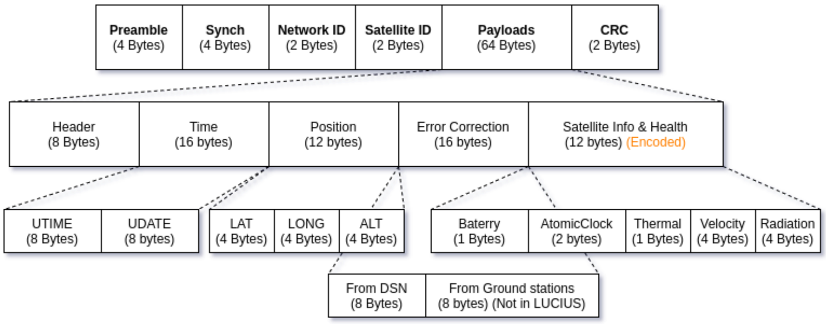

Figure 4 ILNSS Data packet send through S-Band signal

ILNSS packets would be broadcasting in the format of Figure 4. Providing information for determining a location for users as civilian uses and satellite status data to planet ground stations as encoded format due to security purposes.

3.1.4 Coordinate systems in ILNSS

Using the Polar coordinate system to determine where the satellite is and then change Polar coordinate system to Cartesian coordinate system and use coordinate conversion to change Earth coordinate to Moon coordinate then change the Cartesian coordinate system back to Polar coordinate system.

With knowing the angle, distance of the satellite is by changing their coordinate system which is Polar coordinate system to Cartesian coordinate system and using matrix equation to change from Earth Cartesian coordinate system to Moon Cartesian coordinate system after that change Cartesian coordinate back to Polar coordinate system. This calculation can be found in the source code mentioned before.

3.1.5 Mission Area Coverage

ILNSS mission is designed to target entire planet coverage location services but if the budget is too high we can choose only specific sites for single or specified time by period and intersection of the sphere can be covered as shown in Figure 1. The main idea to choose the proximal altitude of satellites if Half power beamwidth (HPBW) then the altitude is up to the antenna system attached on the satellites In case of LUCIUS, 100° degrees of HPBW and the semi-major axis of planet is 1738.1 kilometers at equator which make the calculation as triangle for the altitude using law of sine as

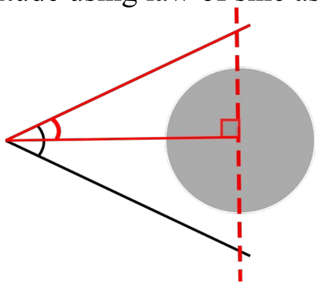

Figure 5. Altitude Calculation

By using semi-major axis as reference for full coverage would make LUCIUS can coverage 50% of Lunar surface (in case, line of sight is cleaned) which can be calculated as equation (4)

$$H = \frac{a}{\tan(\theta)} \tag{4}$$

When H represented altitude of satellites, a, the semi-major axis, and Θ represented half of HPBW as the half angle of black line in figure 5. In case of Lunar, LUCIUS have altitude of 2500 kilometers from surface which meant we can cover more than radius about 3,312.7 kilometers from the surface of Lunar which make spacecraft can use benefit about couple of thousand kilometers from the surface to navigate itself before landed on the Lunar's surface.

3.1.6 Determining User Location

Each satellite would broadcast the signal of ILNSS format packets with S-band frequency reached to the surface and upto 100 km underground with stable RF budgets.

To determine location we use the intersection point of the sphere to decide where the receiver should be in the intersection point and radius computed by capture signal decoded by receiver IC. Our recommended sensitivity for receiver antenna is -160 dBm which corresponds nearly to GPS on the earth for stability and for performance of usage which the margin is upto 21 dBm at the surface of Lunar in LUCIUS missions.

3.1.7. Time realization

Definition of system time

The system time is a time measure in SI unit second which is time inside satellite therefor no leap second in the system time.

The system time is realized in a conception of composite clock and maintained by a time and frequency system located at master control station.

The system time and station time are synchronized via Two-way satellite time and frequency transfer(TWSTFT)

3.1.8 Time dilation

Due to Einstein theory of relativity time is relative to different observer and the time delay that came from relativity might made error on the location determination because of time tick faster to the frame further the earth of the gravitation field and since the satellite have a velocity relative to earth time tick slower for earth perspective relative to moon time and also tick slower due to moon gravitation field which is the result of Lorentz factor which can be separate into three part.

1.Inverse Lorentz factor due to earth gravitational field shown as equation (5)

$$\frac{1}{\gamma_e} \approx \frac{r_{se}}{2r_e} - \frac{r_{se}}{2r_{ce}} \tag{5}$$

When r_{se} represented a Schwarzschild radius of earth, r_e represented a distance from earth center to earth surface, r_{ce} represented a distance from earth center to satellite.

2.Inverse Lorentz factor due to moon gravitation field shown as equation (6)

$$\frac{1}{\gamma_m} \approx 1 - \frac{r_{sm}}{2r_{cm}} \tag{6}$$

When r_{sm} represented a Schwarzschild radius of moon, r_{cm} represented a distance from moon center to satellite.

3. Inverse Lorentz factor due to velocity relative to earth shown as equation (7)

$$\frac{1}{\gamma_v} \approx 1 - \frac{v^2}{2c^2} \tag{7}$$

When v represented a relative velocity of satellite relative to earth centered, c represented the speed of light which is constant to 299,792,458 m/s

Sum all of Inverse Lorentz factor which care only about fraction part and multiply by time (nanosecond/day) which shown as equation (8)

$$\Delta t = 86400\left(\frac{r_{se}}{2r_e} - \frac{r_{se}}{2r_{ce}} - \frac{r_{sm}}{2r_{cm}} - \frac{v^2}{2c^2}\right) \tag{8}$$

When Δt represent how much time tick faster.

For moon surface, Change the Inverse Lorentz factor due to moon gravitation(6) by add moon surface term and don't add lorentz factor due to earth gravitation field because it's so small that we can assume that is zero and change relative velocity in equation(7) into velocity relative to moon surface. Which is shown as equation(9).

$$\Delta t = 86400\left(\frac{r_{sm}}{2r_m} - \frac{r_{sm}}{2r_{cm}} - \frac{v^2}{2c^2}\right) \tag{9}$$

When v represented a relative velocity of satellite relative to moon centered, r_m represented a distance from moon center to moon surface.

3.2 Trajectory Design

Stage 1: Trajectory from ground via Launcher Driver

In this phase we need to bring deployer modules with payload included with Launch Driver. In case of Lunar, LUCIUS operation need to send the mass of LUCIUS-D along with LUCIUS. We need to send 7 satellites each of 3 orbits which mean to 21 satellites must be put into the payload of LUCIUS-D, each LUCIUS satellite have 3.3±0.1 kilograms weight, then we need to put the 69.3 kilograms of mass in to the payload, and LUCIUS-D have weight of 30.01 kilograms, so all of the mass to be eject with Lauch Driver is 99.31 kilograms.

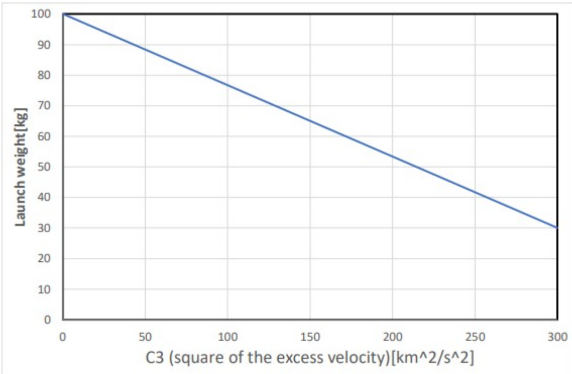

Figure 6. C3-Launch weight relation provided by SpaceMIC

By this we can calculate the excess velocity given by figure 6 which calculate to velocity by square root of C3 to get the velocity of the orbit as equation (10)

$$f(\boldsymbol{\alpha}) = \frac{-30}{7}(\boldsymbol{\alpha} - 100) \tag{10}$$

Which □ represented to launch weight, while the launch weight is 90 kilograms which mean C3 is about 2.95 by computing hypothesis, with excess velocity is the square root of C3 so excess velocity is about 1.719 km/s for LUCIUS-D deployer with 24 LUCIUS as payloads.

Stage2: Trajectory to Moon deployer's orbit

Assume that the initial orbit of satellite speed is equal to the launch driver speed at sent which in circular orbit so the velocity can calculated by equation (11) as

$$v_{cs} = \sqrt{\frac{G(M+m)}{r}} \tag{11}$$

When the velocity of LUCIUS-D in circular orbit, v_{cs} . Newton's universal gravitational constant, G, Mass of the planet, M, mass of LUCIUS-D with payload, m, and the displacement of center of mass between M and m. With m << M, the mass of the LUCIUS-D causes very little change in total mass so we calculate only the main mass of the object which is M, so the altitude from the earth surface is 128,423 km. After making the orbit stabilize enough before the trajectory, use Hohmann transfer orbit to transfer LUCIUS-D from that altitude to target moon orbit at altitude of 5000 km above Lunar surface with low propellant maneuver produce changes of velocity for trajectory between earth and moon

Δv_1 = 2.413543555550986 m/s $\quad$ Δv_2 = 1.488274532063803 m/s

Δv_{total} = 3.901818087614790 m/s

$$F = \frac{-m\Delta v}{\Delta t} \tag{12}$$

$$t_H = \boldsymbol{\pi}\sqrt{\frac{(r_1 + r_2)^3}{8\mu}} \tag{13}$$

The propellant and propulsion is calculated by the third Newton's Law via equation (12). So, the propellant must eject as 239.34 NΔt which is so this makes more proximal for propulsion decision for LUCIUS-D transfer to the moon and 147.87 NΔt to enter the moon orbit at 5000 altitude above the moon surface. Time is coming in to the scene, by using Hohmann transfer orbit which similar to elliptic orbit start from angle 0 radians, end at $\boldsymbol{\pi}$ radians. So we can calculate one half of elliptic period using equation (13)

When t_H represented one half of the t_H orbital period for the whole ellipse, a_H represented the length of the semi-major axis of the Hohmann transfer orbit, r_1 and r_2 represented as radius from the center of mass to start orbit altitude and end orbit respectively. After calculate by using previous info we get time to travel as which is equal to 16814482.78083998 seconds , after change we get the readable time is about 6.4 months.

Another useful variable to decide the proximal time to transfer the LUCIUS-D is angular alignment at the time of start between the source object and the target object will be which is Lunar is calculate by equation (14) and the target velocity being at earth perspective expressed as equation (15)

$$\omega_2 = \sqrt{\frac{\mu}{r_2^3}} \quad (14) \qquad \alpha = \pi\left(1 - \frac{1}{2\sqrt{2}}\sqrt{\left(\frac{r_1}{r_2}+1\right)^3}\right) \quad (15)$$

After applying the equation to be calculated, angular velocity is 8.345456586439956e-08 rad/s and angular alignment is 1.738347292880371 radians respectively.

Stage 3: Trajectory of satellite after deployed

After deploying LUCIUS satellites, it was deployed at inclination 90° because LUCIUS-D was orbiting as polar orbit. LUCIUS need some amount of propellant to change to make direction from inclination 90° to 60° which can be calculated by using equation expressed as equation (16)

$$\Delta v_i = \frac{2\sin\left(\frac{\Delta i}{2}\right)\sqrt{1-\epsilon^2}\cos(\omega+f)na}{1+\epsilon\cos(f)} \quad (16)$$

Where $\boldsymbol{\varepsilon}$ is the orbital eccentricity, $\boldsymbol{\omega}$ is the argument of periapsis, f is the true anomaly, n is the mean motion, a is semi-major axis. But we need to reduce altitude from the center of moon from 5000 to 2500 km above surface which need to do as same as horman transfer orbit which needs to use another equation for delta-V to reduce propellant usage which expressed as equation (17)

$$\Delta v = \sqrt{V_1^2 + V_2^2 - V_1 V_2 \cos(\Delta i)} \quad (17)$$

As V_1 and V_2 represented velocity of orbit before and after change the speed by Δv respectively. After calculate using second equation resulting Δv_i 1.043946891326670 m/s which is more efficient to do change in transfer altitude with inclination of the orbit because of the stimulating trajectory in 2 dimensions. Most of LUCIUS satellites propellant waste in this stage which is the huge change for deployed ILNSS in to the planet not only Lunar but also another planet due to gravity of the planet where ILNSS deploys; make human needs one more step to approach the challenge in insufficient propellant usage if this stage cause an unexpected error when deploying the system. Other way we think to solve the problem is to divide LUCIUS-D in to sub-orbit; In case of 21 LUCIUS must be deployed to 3 orbits around planet, we separate LUCIUS-D in to 3 orbits with 7 LUCIUS each to solve the problem of propellant usage in this stage and use of the sub LUCIUS-D as and ground station if it could to to cause an lower error rate.

Stage 4: Trajectory of satellite orbit

After trajectory transformation the orbit still same but we need to make sure LUCIUS satellites can maintain their aligned using left propellant from previous stage in case of Lunar we use 2500 km as altitude, this isn't fixed altitude to another planet due to link budget is enough for size of Lunar we set this high for easily to track and not to far for 3U CubeSat in space.Use of circular orbit to reduce propellant use while orbit around the Lunar or use eccentricity of the planet instead if planet has more than 0.01 of eccentricity in this case if we do eccentricity applied to ILNSS on Lunar will cause 0.0012 and calculation is differ from perfectly circular which we wrote a simple software to compute possible visible satellite, orbit trajectory in Planet-centered, Planet-fixed coordinates and also useful parameters for elliptic orbit as Kepler's laws and six orbit elements. The velocity of satellites is 1.075476817874790 km/s.

Stage 5: Trajectory of deployer after verify Stage 3 success

LUCIUS-D would take back to the earth LEO in case of moon because we can reuse them in another case; send it to higher altitude and mark as out-of-service satellite or transfer from the orbit the planet polar or near next mission landing site and use as ground station for reducing error rate from space time dilation.

In case of Lunar to LEO we use Hohmann transfer as it is a low-thrust use orbit which requires more time and waste of propellant more than take down to the ground. Transfer orbit uses calculations as written before in stage 1. But parameters changed is gravitational force due to the mass of reference object is changed from earth to moon which has much lower mass compared to an earth which cause Lunar-LEO take more time than LEO-Lunar representing differ delta-V budget, time, and angular velocity but angular alignment is still cause of as same as distance to LEO-Lunar transfer. For landing strategy we also use Hohmann transfer to reduce use of propulsion to keep propulsion left enough for jet landing above 50 meters or more as gravity resistance for softer landing and not to break the modules of the antenna. The LUCIUS-D now being used as ground station to calculate the error from ILNSS LUCIUS with more reliable than calculated from VLBI But only for specific site for in period of LUCIUS orbit (In case of Lunar, period is about 1.4 hours at 2500 kilometers above surface)

4. KEY PERFORMANCE PARAMETERS

ILNSS has many stage of missions which is the chain stage, most of things are important to the mission cause this mission has removed redundant in many options except propellant which still left in stage 4 because after a time LUCIUS satellites can have and orbit distortion due to inaccurate deployment in very little amount cause some sensors can not detected the error or due to unexpected events in space some of propellant will be use to improve accuracy in determining location for ILNSS.

Table 1 Key performance parameters

Key Performances	Value	Expectations
Deployers able to deploy payloads in the correct direction.	< 5% of error compared to the opposite target direction vector	Orientation, time, and position must be correct and verified before deployment.
Deployers transfer the orbit back to LEO or land on the planet's surface.	Lunar's LUCIUS-D safely landed on the surface and was used as a ground station to improve accuracy of ILNSS.	Satellites can de-orbit themselves using propellant, transfer back to earth in case the target planet isn't land-able (Gas planet, too high temperature and pressure, etc.), or land on the planet if it can.
DSN can control Deployers before operation execution.	Verify, Spectate, Cancel, Start signal, and Change the plan (In case of unexpected and emergency).	DSN can control to make ILNSS under the situation manually if it needs to.
Satellites determine and align orientation.	< 10 meters error before calibration	Using VLBI in the first attempt. In another way, use ground stations instead.
Satellites can communicate with DSN.	-	Satellites can keep connected to DSN for error correction due to orbit alignment in half of the period.
DSN can calculate visible sat location	DSN can use VLBI method on nanosatellites	Coordinates conversion works across 2 planet references.
Satellites can provide ILNSS format signal in S-Band	Signals can be broadcast in a sphere-like shape around the planet.	Transmitter and Transponder signals can receive and transmit simultaneously in single satellites.
Satellites can maintain orbit around to target planets.	LUCIUS can keep the orbit error realignment process from DSN's VLBI info.	ILNSS satellites can maintain orbit using left propellant referenced by DSN or planet ground station info.
Receiver/Spacecraft able to receive signal from ILNSS	-	Receiver can receive and calculate position from ILNSS network.
Signals from ILNSS can be calculated using a specific receiver.	ILNSS format signals can be computed by the FPGA on the receiver correctly.	Code, subframe and data from ILNSS network can computed to measure the error and location of signal correctly.
Signal reached the planet's surface.	Spacecrafts can receive signals until they land on	Signals can reach the planet surface and up to -100 km from the planet

	the planet's surface.	surface.
Satellites maintain the failure systems	Provide multiple same operation modules	Multiple hardware in case of systems failure for stability of ILNSS overall.
Operation Uptime	24/7 of Moon day.	More than 95% uptime.
ILNSS can calculate a more; precise location.	-	Can give a more widely accurate location for spacecraft than image processing or previous method used.
VLBI method can be used in verifying location of satellites	24 satellites or more	VLBI technique works in the case of very small objects such as the 3U CubeSat.
The space missions cost of landing, maintaining, and determining the trajectory on the planet.	Lesser cost with more accurate when compared to previous method	Improve accuracy and benefits compared to another landing method for spacecraft landed before.
ILNSS be used as prototyping another deep space colonization	ILNSS used as prototype or more technically used.	Human beings can explore easier, faster, cheaper in deep space exploration missions.

5. SPACE SEGMENT DESCRIPTION

5.1 Satellites design (LUCIUS)

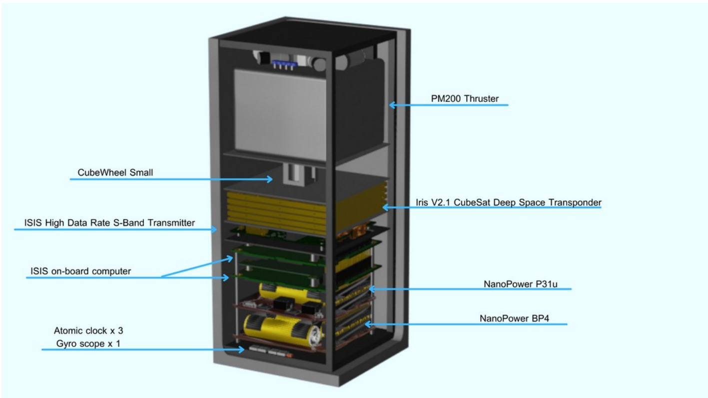

Figure 7. LUCIUS Satellites

Table 2. LUCIUS Component specifications

Component	Detail	Description
Atomic Clock	SA.45s Space CSAC	**Radiation Tolerance** 20 kRad **Power consumption** <120 mW **Frequency** 10 MHz
Star Tracker	arcsec Sagitta Star Tracker	**Accuracy** 2 arc seconds (1 sigma) cross-boresight, 10 arc seconds (1 sigma) around boresight **Update Rate** up to 10Hz **Lost-in-space availability** > 99% of the night sky. **Nominal power consumption** < 1 W
Gyro Sensor	Sensenor STIM377H	**Range gyro** ±400 °/s **Range Accelerometer** ±10g **Optional ranges** ±5g and ±30g **Data Output** Digital 24 bit RS422 **Angular Random Walk** 0.15°/√h **Sampling frequency** 2000 SPS
Propulsion system	Hyperion Technologies B.V. PM200	**Thrust Power** 0.5 N **Power Consumption** < 6 W **Total Impulse** >850 N s **Specific Impulse** >285 s
Main Processor	ISIS On board computer	**Processor** 400MHz 32-bit ARM9 processor

		Volatile Memory 64MB SDRAM **Code Storage** 1MB NOR Flash **Critical Data Storage** 512kb FRAM On-board temperature sensor
Antenna Systems	IQ spacecom X Band Antenna	**Operation frequency** 8.025-8.400 7.145-7.250 GHz **Maximum gain (main direction)** 11 dBi **Half power beam width** 40° **RF power input** < 2 W **Impedance** 50 Ω **Polarization:** RHCP (opt. LHCP)
	ISISSpace S-band Patch Antenna	**Frequency Range** 2200-2290 MHz **Gain in boresight (centre frequency)** 6.5 dBic **Half Power Beam Width** 100 ° (degrees) **Bandwidth** > 100 MHz **Axial Ratio** < 3 (for ±100° degrees) dB
Transponder	Iris V2.1 CubeSat Deep Space Transponder[5]	Please check in the references on this component due to a very specific large amount of details for Iris V2.1.[6]
	ISIS TXS S-Band Transmitter	**Frequency** 2200 to 2290 MHz **Transmit Power** 27 to 33 dBm **Modulation** OQPSK **Data Rate** 3.4 Mb/s **RF Output Impedance** 50 Ohms
Power Source	GOMspace NanoPower BPX	**Technology** Lithium-ion **Capacity** 2600 mAh 77 Wh **Configuration** 2S-4P: 6 - 8.4 V & 10.4 Ah
	GOMspace NanoPower P110	Due to multiple configuration and wiring problems please check NanoPower P110 details in references.[7]
Aligning system	CubeWheel Small	**Speed range** ±8000 RPM **Speed control accuracy** <5 RPM **Max torque** 0.23 mNm **Momentum storage (@8000 RPM)** 1.7 mNms

LUCIUS is made in size of 3U cubesat which causes the LUCIUS satellites to fit in most of the launcher and deployer with square limit volume for launch and give most volume to put the payloads inside. The problem for LUCIUS is it needs to contact DSN on earth ground station to verify the location themselves but the antenna must be separate because frequencies for DSN are near the center frequency of ILNSS missions which can cause the signal distribution. Then we place the antenna most far away to help the signal process separately (At least to reduce) at different sides of the satellite which face on the ground and face to the earth when LUCIUS can be visible by DSN ground stations.

5.2 Deployer design (LUCIUS-D)

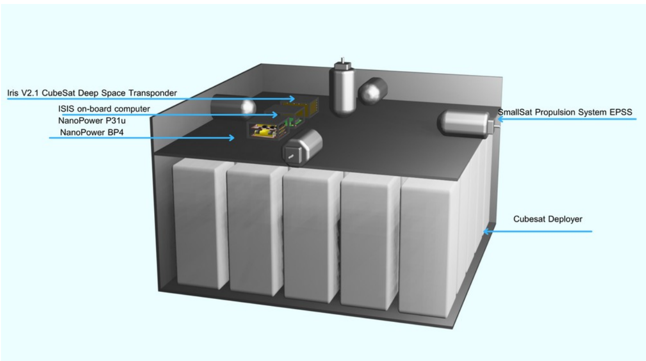

Figure 8. Deployer design

Table 3. LUCIUS-D Additional Component specifications

Component	Detail	Description
Deployer Modules	PicoSatellite Launcher (PSL) 3U Configuration	**Max Payload Mass** ≤ 4.0 kg **Mass** < 1.63 kg **Typ. deployment energy** 3.1 J **Actuating pulse** 28 ± 4 VDC / 1s
Main Propulsion system	Bradford ECAPS 1N GP Thruster	**Inlet Pressure Range** 4.5 to 22 bar **Thrust Range** 0.25 to 1 N **Nozzle Expansion Ratio** 100:1 **Steady State Isp** 1900 to 2230 Ns/kg **Specific Impulse** 194 to 227 s **Density Impulse** 2318 to 2721 Ns/L **Minimum Impulse Bit** < 70 mN s

5.3 Link Budget

Table 4 Link budget in different communication

Description	DSN Uplink	DSN Downlink	LUCIUS-D - Sat
Distance (km)	3.8463e+05	3.8804e+05	7.2996e+03
Elevation (deg)	64.2063	21.1103	-1.1569
Tx EIRP (dBW)	65	37	9
Polarization loss (dB)	3.0103	3.0103	3.0103
FSPL (dB)	221.2843	222.7232	176.6016
Total propagation losses (dB)	221.2843	222.7232	176.6016
Received isotropic power (dBW)	-162.2946	-191.7335	-172.6119
C/No (dB-Hz)	89.3045	59.8656	78.9873
C/N (dB)	21.5230	-14.1138	15.9770
Received Eb/No (dB)	19.3045	16.8553	19.9564
Margin (dB)	7.3045	4.8553	7.9564

Table 4 assumes there is no cable loss in satellites because of the very short length of cable in satellites. DSN antenna also provides multiple size antennas, ILNSS chose a 34 m diameter HEF subnet antenna for X-Band for uplink (Earth - ILNSS) and downlink (ILNSS - Earth) operation in case of LUCIUS missions. With verifying the location of satellites, inside the packets send the location (With error or as satellites think they are), network ID, and satellites ID to DSN to calculate the error from the exact location they should be. For ILNSS LUCIUS Downlink from Lunar surface to satellites, we assume the antennas have 20W of Transmit EIRP, diameter of antenna is 1.5 meters and antenna aperture efficiency is 0.65. Also The deployer does not require to downlink with the surface because the deployer is on the planet's ground or had trajected back to Earth LEO.

5.4 Propulsion (Error correction and delta-V budget)

5.4.1 LUCIUS-D

Stage 2 needs propulsion to transfer to Lunar orbit and enter the polar orbit of Lunar at altitude 5000 km. Needs to use the total delta-V as 271 NΔt. In **stage 3** the orbit is still in the polar orbit and no change in any delta-V so propulsion uses very little propulsion to accurately determine the direction. Lastly **stage 5**, DSN needs to decide and execute with left propellant of the LUCIUS-D to deorbit, transfer back to LEO or land on the surface which is up to the operation and brought to Lunar propellant.

5.4.2 LUCIUS

Stage 3 needs propulsion to change the inclination from 90° degrees to 60° degrees of the Lunar which need large amount of propellant as calculated we need up to 1630.8978783 NΔt in this stage for **stage**

4 and 5 the orbit still same in circular orbit we don't need to change in any delta-V after stage 3 so the propellant left is use for correct the error using information from DSN via VLBI method if the error is too huge when calculate the position of ILNSS users while using reaction wheel for small amount of orientation needs for better accuracy than use propellant.

5.5 Satellite (LUCIUS) Clocks and Time

LUCIUS consists of 3 atomic clocks inside to handle failures in cases like the Galileo GPS satellite on earth. Due to the size of LUCIUS is very small compared to GPS satellites we must reduce the size of atomic clock to chip scale atomic clock (CSAC) which has nearly performance with GPS output to CMOS connection ports about 10 MHz vs GPS as 10.23 MHz and have radiation tolerant as 20krad but with P code and C/A code on GPS. ILNSS doesn't have a purpose to reduce accuracy on determining location so we could give full performance with P code for ILNSS fully functional at 10MHz of CSAC speed.

5.6 Energy Balance

$$\Delta E = Pt_{genterate} - Pt_{consume} \tag{18}$$

Which we nned to calculate power consumption as equation (18) from the orbit period but power generation time is the half of the orbit because our main power resources is came from the SAP which produce energy by Sun. We assume we can use all of the power so the $\Delta E = 0$ then we calculate only max power consumption for the satellites which hace 3 modes as Table 5.

Table 5 Power Consumption & Energy Balance trends

	Initial Mode	Trajectory Mode	Mission Mode
Description	Only connected to DSN	Inclination Maneuver	ILNSS role operation
Power Consumption (mW)	36640	42640	65700
Power Regenerate (mW)	67200	67200	67200
Power Balance (mW)	30560	24560	1500
Energy balance trends (Wh)	+	+	+

As in the table, in Mission mode is calculated power consumption in term of full orbit. Satellites can only regenerate power using SAP in half of orbit period, but satellites components consume power all time in orbit period except DSN connection which only half orbit operation because there's very low possiblility to receiver the signal from DSN so we reduced power consumption by half to make energy balance for satellites enough

6. SPECIAL ASPECTS OF THE MISSION

ILNSS is a network that helps spacecrafts, astronauts, rovers, and helicopters in other deep space exploration missions define the location on an interplanetary planet with lower cost, fast, reliable, and stable when needed to explore on the planet surface intensively. From landing till exploring, they can use our network to determine location and convert to coordinates from planet across planet to the earth without problems when referenced by ILNSS. ILNSS not only help in specific space missions but also a prototype network for small satellites mission in commercial use case of geolocation or internet service which require more satellites than ever for space colonization due to overpopulation, limited and insufficient resources on the earth which also support Moon to Mars mission by NASA directly in long terms.

7. CONCLUSION

ILNSS would improve accuracy on landing spacecraft at first sight on the Lunar in mission codename called LUCIUS and will be ported to Mars with another mission codename. Not only for landing a single spacecraft but also multiple projects as long as ILNSS lifetime, including on surface usage to determine selenographic location for astronauts, rovers, and for civilians in space colonies.

8. REFERENCES

[1] Lambright, W. H. (2014). Why Mars: NASA and the politics of space exploration. JHU Press.

[2] Braun, Robert D., and Robert M. Manning. "Mars exploration entry, descent and landing challenges." 2006 IEEE Aerospace Conference. IEEE, 2006.
[3] Nefedyev, Y. A., Andreev, A. O., Petrova, N. K., Demina, N. Y., & Zagidullin, A. A. (2018). Creation of a global selenocentric coordinate reference frame. Astronomy Reports, 62(12), 1016-1020.
[4] Lissauer, J. J., Jontof-Hutter, D., Rowe, J. F., Fabrycky, D. C., Lopez, E. D., Agol, E., ... & Welsh, W. F. (2013). All six planets known to orbit Kepler-11 have low densities. The Astrophysical Journal, 770(2), 131.
[5] Kobayashi, M. (2017). Iris deep-space transponder for SLS EM-1 CubeSat missions.
[6] https://www.jpl.nasa.gov/cubesat/pdf/Brochure_IrisV2.1_201611-URS_Approved_CL16-5469.pdf
[7] https://gomspace.com/UserFiles/Subsystems/datasheet/gs-ds-nanopower-p110-210.pdf