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Original Article

Conceptual Design of Moon Lander Spacecraft

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Space exploration is important for the development of humankind such as for exploiting unlimited resources from the celestial body or making a new colony on outer space. The Moon exploration is chosen because it can be made as a base for outer space exploration. In this study, the main objective is to design a Moon lander spacecraft with a mission to take a photograph of the Indonesian flag with Earth's globe as the background and transmit it to the ITB ground station. Designing a Moon lander spacecraft involves many engineering fields, such as aerospace, astronautics, electrical, telecommunication, etc. In this study, the design process is based on the DRO and consists of trajectory analysis, landing site and scenario, determination of launch vehicle, subsystems analysis, and cost analysis. The subsystems are limited to propulsion, payload, power, communication, and structure. The design generates a 500-kg Moon cargo lander spacecraft with a landing site on 28° latitude. The mission will take 6 days to reach the low lunar orbit of the Moon. The propulsion analysis shows that the propellant needed is 200 kg N2O4/UDMH. The resolution of the Earth's globe is 134.3 arcseconds/pixel. The power system can provide power for 5.69 years. The spacecraft also provides 3 links to communicate between the spacecraft on the Moon and the ITB ground station. The structure's minimum factor of safety is 1.6 with its first natural frequency in the launch vehicle being 63.66 Hz. The spacecraft's expected cost is 220M USD. The results show that the Moon lander spacecraft can conduct such a mission based on the DRO. However, the payload subsystem needs to be re-evaluated.

Keywords: aerospace engineering, spacecraft, space exploration, Moon lander, conceptual design

1. Introduction

Many things regarding outer space have not been understood by humankind. Space exploration is a way to study and know more about outer space. The need for space exploration increases along with time so that the preparation of space exploration should be started early. The Moon is the nearest celestial body from Earth. It is a good destination for beginners to show the capabilities of conducting space missions.

Numerous missions to the Moon have been conducted by the following nations and entities (in chronological order): the Soviet Union, the United States of America, Japan, the European Space Agency, China, India, Luxembourg, and Israel. This study tries to stimulate space awareness, especially for the young generation, and to create networks with other space entities in the world.

In this study, the design process is limited to the conceptual design of Moon lander spacecraft that is divided into several subsystems, such as propulsion, payload, power, communication, and structure, based on [1]. This study aims to design a Moon lander spacecraft based on the design requirements and objectives (DRO) with the minimum mission to uphold the Indonesian flag, take a photograph of the flag with Earth as background, and transmit it to the ITB ground station in real-time. It is reported with design methods, results, discussion, and conclusions. Hope this study will be used for the development of space technology in Indonesia and other countries.

2. Design Methods

The spacecraft design is an iterative process. The process passes several stages to get the best design. In this study, the design stages consist of design requirements and objectives, literature study, comparative study, mission concept, subsystems analysis, and cost analysis. The flowchart of the design methods is shown in Figure 1.

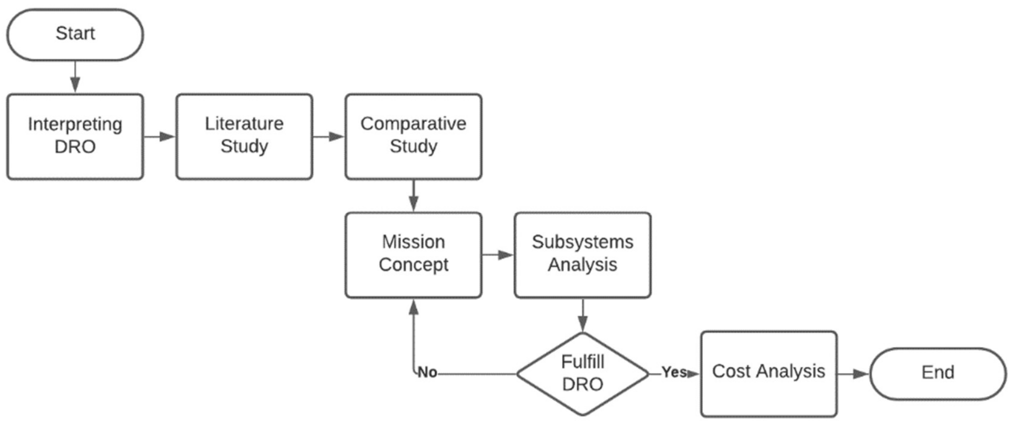


Figure 1. Flowchart of the spacecraft design process.

In the design process, the first things needed are design requirements and objectives (DRO). DRO is a tool to limit the work in designing a spacecraft. In this study, the DRO is shown in Table 1.

Table 1. Design Requirements and Objectives.

General	Subsystem
• Design a Moon lander spacecraft to uphold the Indonesian flag, take a photograph of the flag with Earth as background, and transmit it to the ITB ground station in real-time. • Use of existing commercial small lift launch vehicle. • Additional missions that can be conducted by the spacecraft become an advantage. • The spacecraft is designed to have operating life (with standard services) of 1 year.	• The spacecraft subsystems are limited to propulsion, payload, power, communication, and structure. • The subsystem requirements should be derived from the mission concept. • The spacecraft's structural natural frequency should be well above the vibrational frequencies experienced during the launch process. • Determination of battery capacity must consider the battery performance degradation during spacecraft operating life. • Solar panel degradation should be considered • The resolution of the Earth's globe image should be better than 0.1 arcsecond/pixel.

The next stage is literature and comparative study. Literature and comparative study are used as the base reference on how to design the spacecraft. This stage is done by collecting references from books, papers, articles, etc.

After that, the mission concept is determined. The mission concept is the scenario of the spacecraft through all phases, including the concept of operation, trajectory analysis, landing location and scenario, and determination of launch vehicle.

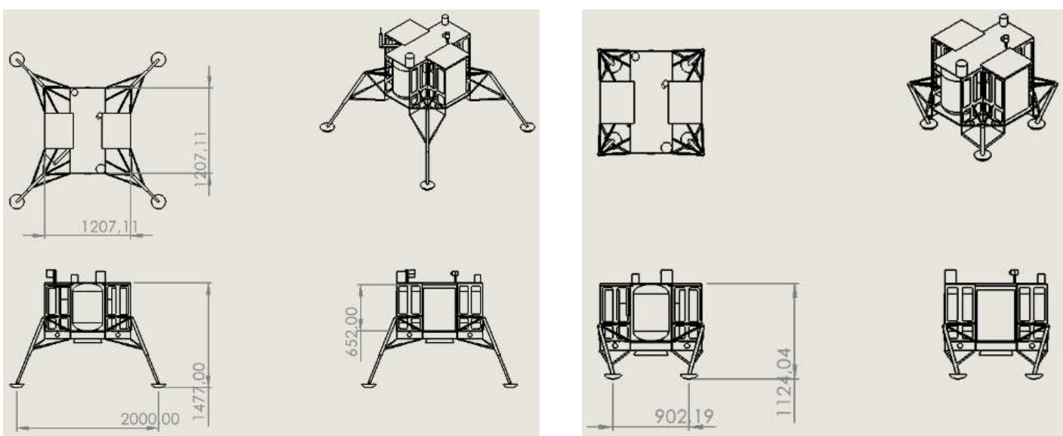
Then, subsystems analysis is conducted. The subsystems are limited to propulsion, payload, power, communication, and structure. This stage is an iterative process due to conflicts between subsystems. Hence, there will be trade-offs to achieve the best spacecraft design.

The last stage is cost estimating. The cost estimating is done to predict the cost needed to conduct the mission by using Unmanned Space Vehicle Cost Model (USCM8) [1]. The inflation rate is also considered in the cost estimating calculation.

3. Results

3.1. Mission Concept

The Moon lander spacecraft designed was named Moon Strider. There are two mission concepts in the design process: orbiter lander and cargo lander. The orbiter lander means the spacecraft consists of two parts which are an orbiter and a lander. The cargo lander means the spacecraft offers cargo to be placed on the Moon. The Moon Strider uses the concept of cargo lander due to the simplicity of design and profitability of the cargo. It should be noted that carrying cargo is an additional mission so that the design focused on the main mission. The estimation for the Moon Strider's wet mass is 500 kg. The three-view drawing of the Moon Strider is shown in Figure 2.



(**a**) (**b**)

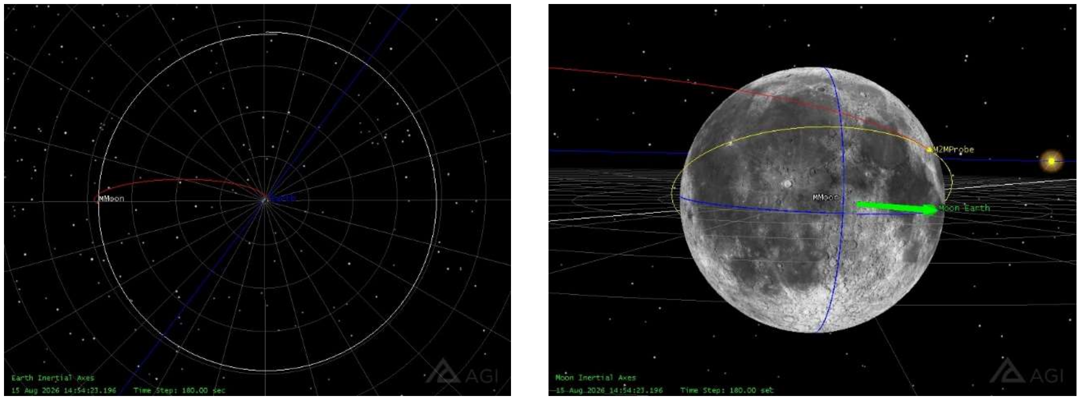
Figure 2. The three-view drawing of the Moon Strider when (a) operating on the surface of the Moon and (b) being launched in the launch vehicle.

3.1.1. Trajectory Analysis

The mission will be Indonesia's first interplanetary space probe mission. As the Indonesian flag is desired to be raised, it is highly preferable to have the lander arrive at the surface of the Moon on the Independence Day of Indonesia. Hence, the Moon Strider was designed to begin the missions on 17 August 2026 (with five years production period in mind) with the flag-raising ceremony as the first mission conducted on the lunar surface [2].

To accommodate such a plan, the Moon Strider is expected to land on 17 August 2026. The best time to take a picture with the Earth globe as the background happens near the new moon phase because, in this phase, the Earth's view is clear. In August 2026, this phase will happen between the seventh to the nineteenth [3]. The trajectory simulation on AGI STK 11.2 Figure 3 shows that the

Moon Strider will launch on 8 August 2026 at 13:46:04 UTCG and will arrive at low lunar orbit (LLO) on 14 August at 14:54:56 UTCG.



(a) **(b)**

Figure 3. Trajectory simulation (a) 6-days transfer orbit and (b) final parking orbit at LLO.

Arriving at the new moon phase, however, means the landing location will not receive sunlight. This arises a new problem to generate power for the spacecraft to function. To overcome this obstacle, the Moon Strider will orbit on low lunar orbit for approximately three days to receive solar radiation as the power supply. During the 72 hours, it is approximated that the Moon Strider will orbit the Moon 36 times before it finally descends on 17 August 2026, where the sunlight is visible on the Moon's surface and the Earth is visible.

3.1.2. Landing Location and Scenario

The landing location selection depends on the scientific constraint (geomorphologic and geological characteristic, distribution of lunar resources, gravity, magnetic field, and demand) and engineering constraint (topographic slope and terrain obstacle, communication ability, and temperature). Based on the scientific constraint, high altitude (above 60°N or °S) is not recommended because it exceeds high SEA (Solar Elevation Angle). Lower altitude (below 30°N and °S) is also not recommended because it has relatively high-temperature variation. This thermal condition will need more complex thermal control. However, higher altitude will give a better result for the main mission demand which is a photo of the Indonesian flag with Earth globe as background. For the engineering constraint, the landing site needs to be safe for soft landing based on a steep slope, craters, boulders, and rock that is located on the lunar nearside with the low latitude (below 45°N and °S). The near side of the Moon with low altitude also gives a better option for communication subsystem because the communication can directly travel from the Moon to the Ground Station at a shorter distance [4]. Therefore, the landing location for the Moon Strider mission is located at 30°N latitude on the nearside of the Moon.

For the landing scenario, after reaching enough power needed, the spacecraft will slow down its orbit velocity to land safely at the landing site. The Moon strider will land with the main thruster as the main landing engine and several smaller thrusters as a stabilizer. To navigate the landing location, LIDAR (Light Detection and Ranging) will be used to determine the contour of the landing site and the spacecraft altitude [5].

3.1.3. Determination of Launch Vehicle

Based on the DRO, a commercial small lift launch vehicle would be used to place the Moon Strider into low lunar orbit. The launch vehicle was determined by using a parametric study (shown in Figure 4) which relates the spacecraft's mass with the launch vehicle's mass of several existing missions to the Moon.

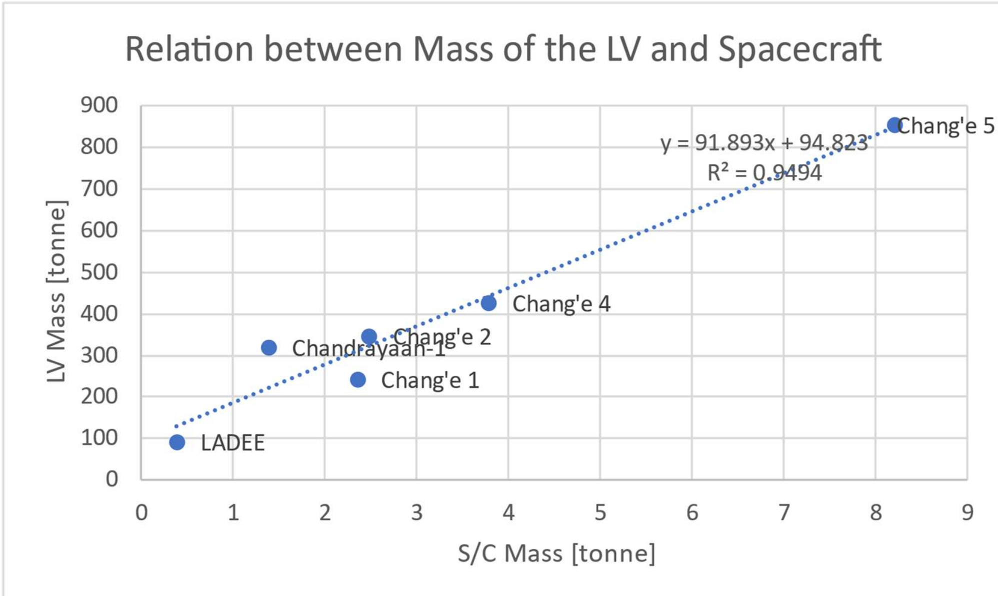


Figure 4. The relation between the launch vehicle's mass and the spacecraft's mass of several existing missions to the Moon.

By using the linear regression equation, the required mass of the launch vehicle for 500-kg-spacecraft can be estimated to be 140 tonnes. One of the commercial small lift launch vehicles which would have approximately adequate required mass is the Vega launch vehicle [6]. The Vega launch site is located at the Guiana Space Centre (CSG) in French Guiana.

3.2. Subsystems Analysis

3.2.1. Propulsion Subsystem

In this study, the launch vehicle is assumed to be capable of carrying the Moon Strider spacecraft into low lunar orbit. Thus, the spacecraft's propellant is used only for the landing phase. The discussion of the propulsion subsystem is limited to the design of the main engine and propellant.

The main engine refers to the Lunar Module Descent Engine (LDME) which is used by NASA in its Apollo program [7]. The engine provides thrust during the descent phase to maintain the spacecraft's attitude. The overall length of the main engine will be 600 mm with a 400 mm diameter and a thickness of 2.5 mm and weigh 12.5 kg. It uses unsymmetrical dimethylhydrazine (UDMH) as the fuel and dinitrogen tetroxide ($N2O4$) as the oxidizer.

The main engine uses UDMH because the engine's ignition is hypergolic in the combustion chamber. UDMH also provides the highest engine operation, theoretical, and density ratio from operation mixture ratio, compared to the usual MMH and hydrazine [8]. UDMH is also stable and can be kept loaded in rocket fuel systems for long periods, which makes it appealing for use in many liquid rocket engines, despite its cost. By using the Tsiolkovsky rocket equation, the propellant needed for landing on the Moon is 200 kg.

3.2.2. Payload Subsystem

From the DRO, the resolution of the Earth's globe image should be better than 0.1 arcsecond/pixel. However, it was considered impractical to achieve a single image with such resolution due to the high field of view needed. To reduce the field of view, the spacecraft needs to land on high latitude. However, the landing site is suggested to be on low latitude. Thus, there needed to be trade-offs between these two variables.

Since the camera used for spacecraft is subjected to harsh environments, such as extreme temperature difference and radiation, the sensor needs extra requirements compared to ground-based cameras. Therefore, the sensor chosen needs to be a space-grade camera. The XGS 12000 was chosen as the sensor used in the Moon Strider. The XGS 1200 is a space-tested sensor with small power consumption and sensor size, which are 0.9 W and 3.2 μm respectively [9]. With a 90° vertical field of view, the XGS 12000 has an angular resolution of approximately 134.3 arcseconds/pixel. Based on the Rayleigh diffraction criterion with the quality factor of 1, the aperture diameter of the camera system is 1.87 mm. The illustration of the picture taken is shown in Figure 5.

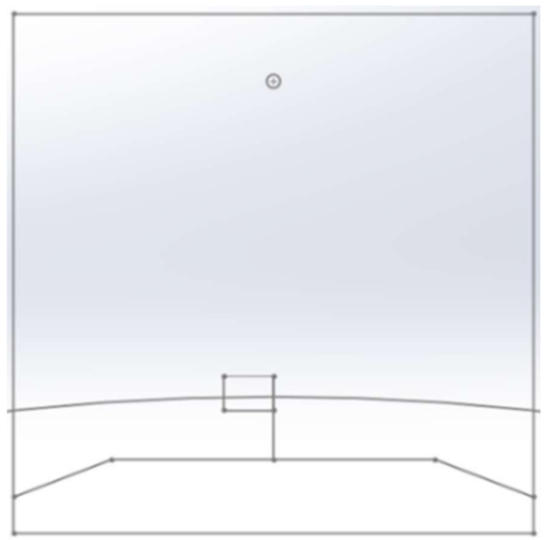


Figure 5. The illustration of the picture taken by the Moon Strider.

The other payload to consider is the Indonesian flag. The flag must be able to maintain its color from the harsh environment. Therefore, the flag's storage is needed to provide sufficient protection. The flag is designed to have a size of 15 cm × 10 cm with a deployment mechanism to deploy the flag from the storage.

Based on the DRO, there is no specific requirement needed for the Attitude Determination and Control System (ADCS) or as known as Guidance, Navigation and Control (GNC) subsystem. However, the spacecraft will need several ADCS components for the lunar lander mission. First, the star tracker is used to determine the spacecraft attitude in the star's inertial reference with the earth as the reference point. In addition, a sun sensor is also used to detect the direction of sunlight to meet the needs of the solar panel. Inertial Measurement Unit (IMU) is used for the spacecraft to determine its acceleration and angular rate. For the landing phase, LIDAR (Light Detection and Ranging) is needed to determine the altitude of the spacecraft and the terrain condition for landing location [5]. To control the spacecraft attitude, the thruster will be chosen as the main actuator because it can provide a huge amount of torque at an instant. However, the reaction wheel is needed to stabilize the huge moment amount from the thruster and can be used for rotating the spacecraft in a small portion. All the ADCS components above have several dimensions, mass, and power variance that will affect its performance [1]. For the Moon Strider spacecraft, the ADCS system has a maximum 22 W power budget and an 18 kg mass budget. Furthermore, the ADCS subsystem analysis is still needed to determine the ADCS dimension and placement location.

3.2.3. Power Subsystem

To accomplish the mission, the power subsystem needs to be designed to accommodate the power requirement of each subsystem. A power system generates, converts, or stores the power and energy required for the entire one year of mission life. A spacecraft power system typically consists of solar arrays and batteries [10].

The Moon Strider's power requirement is classified into two based on the mission phases: before and after reaching low lunar orbit. The former phase solely depends on batteries as a power source since the solar array is yet to be available (before being deployed out of the launch vehicle), whereas the latter combines the two components based on the time of the lunar day. Phase one requires total energy of 5360 Wh and phase two requires maximum power of 93 W [1]. These requirements will be used to size the respective component.

The Moon Strider uses ISIS Cubesat as solar panels and two batteries, namely Pylontech 2000 and 3000. The solar panels' total area is 0.4 m^2 which weighs 25.3 kg and can generate a peak power of 94 W and average power of 48 W. The batteries weigh a total of 56 kg and can store 5400 Wh of capacity [11]. This sizing is also restricted to the maximum dimension and mass available.

The spacecraft will stay in low lunar orbit for 72 hours to wait for 'lunar dawn' on 17 August 2026 because the main mission requires the presence of solar radiation to execute. During the orbiting, the battery is supposed to be fully discharged and the solar panel begins generating power to power the required subsystems during the waiting time. For a two-hour orbiting period, the spacecraft is expected to revolve approximately 36 times before finally beginning to descend.

3.2.4. Communication Subsystem

For communication, the Moon Strider spacecraft is designed with three links consisting of TT&C (Telemetry, Tracking, and Command) uplink as link 1, TT&C downlink as link 2, and payload downlink as link 3. Under ITU regulation, TT&C communication links use S-band frequencies (2.1 GHz for link 1 and 2.2 GHz for link 2) because it is commonly used in Moon lander mission TT&C link [12]. For payload downlink transmission, X-band will be chosen due to its speed. All those three links will be analyzed with the communication link budget and the result will depend on the communication components specification (transmitter, receiver, and antenna) and atmospheric condition.

There are some requirements for communication link analysis such as the gain margin and typical data rate. For satellite communication, the TT&C link must have a margin above 6 dB while payload transmission must have a margin above 3 dB. There is also a typical data rate needed for moon lander mission such as 8 kbps for TT&C uplink and 20 kbps for TT&C downlink. In the first communication link budget iteration, all three links still had margin values below the requirements (0 dB for link 1, -3.5 for link 2, and -24.53 for link 3) which means the communication cannot be conducted. To overcome this problem, the communication subsystem components need to be readjusted by changing the transmitter, receiver, and antenna component selection, redesigning Ground Station (GS) antenna, and changing the data rate for the transmitter.

The Moon Strider uses STC-MS03 with 14 W maximum power and 6.25 Mbps maximum data rate as TT&C transceiver because it has a high-power output and can be used as a transmitter and receiver at the same time [13]. For payload transmission, the transmitter component is changed from Swift-XTS to Swift-XTRX (45 W maximum power and 100 Mbps maximum data rate) because it has more power output [14] [15]. Omnidirectional antennas with 0 gain are chosen for all three links antennas in the spacecraft because they have a wide beamwidth (360°) and have the simplest structure design. For GS antenna, a parabolic reflector is also chosen for both TT&C and payload communication link because it has high gain value, is relatively easy to design, and has a wide beamwidth (180°). The gain of the parabolic reflector depends on the value of the frequency link and the reflector diameter. For the Moon Strider mission, ITB GS needs to build a 10 m diameter parabolic diameter to provide 44.25-dB gain for link 1, 44.66-dB gain for link 2, and 55.87-dB gain for link 3 [1].

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Several conditions also needed to be considered in designing the communication links such as losses due to atmospheric conditions, polarization, and antenna miss alignment. Atmospheric condition is mostly affecting the communication link budget because the Moon Strider is located so far away from GS. In addition, the modulation type and the needed margin value also affect the communication link. For the Moon Strider mission, BPSK modulation with 9.6 dB Eb/No will be used for all three links. Then, a 6 dB margin is needed for the TT&C link and a 3 dB margin is needed for the payload link. Finally, the data rates of communication will be adjusted based on communication link budget analysis according to the equation in the SMAD to get the most effective and efficient communication link [1]. The detail of the communication link budget is shown in Appendix 1.

Based on the communication link budget analysis, all the three communication link margins have values more than the requirement which means the communication can be done. For payload transmission, with a 45 MB payload data size, the data transfer can be finished in 47 minutes. It means for 1 day communication time with 12 hours range assumption, the spacecraft can transmit up to 30 photographs to the ITB ground station.

The mass budget and dimension of the communication subsystem are listed detailed in the structure subsystem part. The minimum power required for communication subsystems is only determined by TT&C downlink (14 W) so the GS can still give a command and get the feedback from the spacecraft then. However, the maximum power for the communication system is when all three communication links are active simultaneously with a maximum power of 63 W.

3.2.5. Structure Subsystem

The design of the structure subsystem is limited to the design of the spacecraft's bus and the landing gear. The requirements of the structure subsystem are mainly derived from the selected launch vehicle. The usable volume inside the Vega fairing has a diameter of 2,426 mm and a height of 5,965 mm with a conical shape at the top. The spacecraft's design limit load factors are -7.5 in the longitudinal direction and 0.9 in the lateral direction. To prevent dynamic coupling with fundamental modes of the Vega, the spacecraft should be designed with a structural stiffness that ensures that the first natural frequency is greater than 60 Hz.

The materials for the spacecraft's structure are selected primarily based on the specific strength and the specific rigidity. The most common structure which is implemented to the spacecraft's structure is honeycomb sandwich panels and shells. The honeycomb structure provides a material with minimal density and relatively high out-of-plane compression and shear properties. Below is the list of materials used for typical space applications [16].

Table 2. The List of Materials for Typical Space Applications.

Material	ρ [kg/m^3]	σ_T [MPa]	E [GPa]	Specific Strength [MN·m/kg]	Specific Rigidity [GN·m/kg]
Aluminium 2024-T3	2780	485	72.4	0.174	0.0260
Aluminium 6061-T6	2700	310	69.0	0.115	0.0256
Aluminium 7075-T73	2810	570	72.0	0.203	0.0256
A-286 2-in.	7940	970	201.0	0.122	0.0253
Inconel 718 4-in.	8220	1280	203.0	0.156	0.0247
AZ31B H24	1700	270	45.0	0.159	0.0265
Ti-6Al-4 V annealed	4430	1050	104.8	0.237	0.0237
AMS 7906	1850	320	290.0	0.173	0.1568
T800H/epoxy [0, ±45, 90]	1600	820	60.0	0.513	0.0375
Kevlar 49/epoxy [0, ±45, 90]	1400	500	30.0	0.357	0.0214
E-Glass/epoxy [0, ±45, 90]	2200	300	10.0	0.137	0.0045

The Aluminium 2024-T3 is chosen to be the material used in the Moon Strider due to its high specific strength and specific rigidity. Besides, aluminium has high thermal conductivity so that it can discharge heat efficiently.

To design the landing system, it is important to choose a suitable landing gear type design. There are two common designs for lunar landers which are the cantilever and the inverted tripod landing gear design [17]. The difference between these two designs is in the connection of the secondary struts. The cantilever design has the secondary struts connected to the lower end of the primary strut upper section whereas the inverted tripod design has the secondary struts connected to the footpad. The cantilever design is chosen because the structure is lighter and the risk of interference with obstacles around the footpad can be reduced.

To start the structural design, the spacecraft's bus took the idea from the Apollo Lunar Module with a hexagon shape. The Apollo Lunar Module is a manned mission to land on the Moon so that the structure is designed to protect all the payloads including the men. Hence, for the Moon Strider, the structure can be reduced to achieve lighter spacecraft. The dimensions of the spacecraft are driven by the payload fairing and the dimensions of the payloads. After that, the structure's natural frequency is checked. The natural frequency depends on the mass and stiffness of the spacecraft. To increase the natural frequency, the structure needs to be shorter and thicker due to higher stiffness and vice versa. The final layout of the Moon Strider is shown in Figure 6 with payloads consisting of thermal control, ADCS, on-board processing, S-band transceiver, and X-band transmitter.

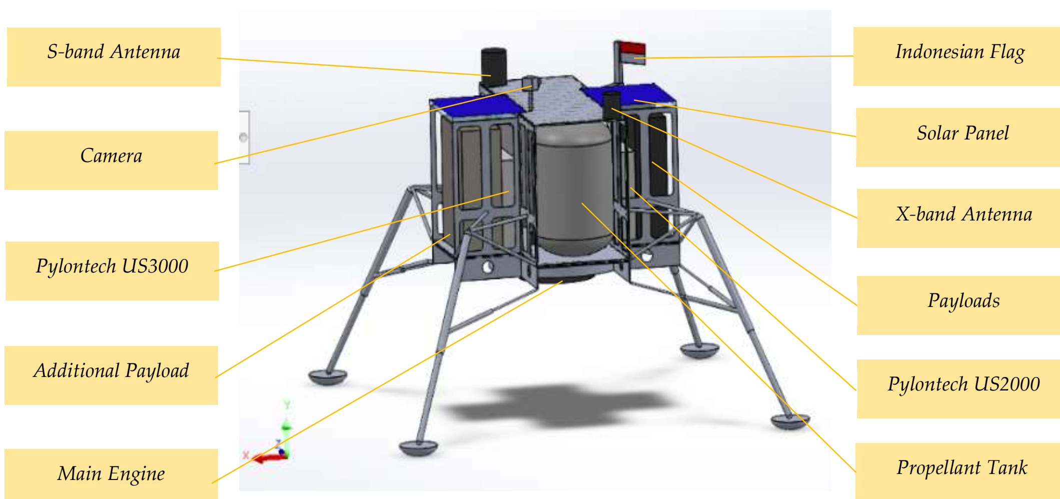

Figure 6. The layout of the Moon Strider.

The center of gravity of the Moon Strider is -57.18 mm (x-axis), 954.80 mm (y-axis), and 0.70 mm (z-axis). It should be noted that this center of gravity is an approximation due to several components and payloads that are not designed in this study. The spacecraft is designed symmetrically between xy- and yz-plane so that the center of gravity is around the center of the spacecraft. The center of gravity on the y-axis should be as low as possible so that the spacecraft is stable when landing. However, it cannot be too low due to the engine clearance.

The list of components' dimensions and mass used in the Moon Strider are shown in Table 3. Several subsystems are not analyzed in this study, such as thermal control that is used to maintain the temperature of the spacecraft, attitude and determination control system that is used to make sure that the spacecraft is on the right track and position, onboard processing that is used to process the data collected by the spacecraft, etc. In this study, they are not analyzed in detail. However, they are assumed to work well with no failure and are listed below to calculate the total mass of the spacecraft.

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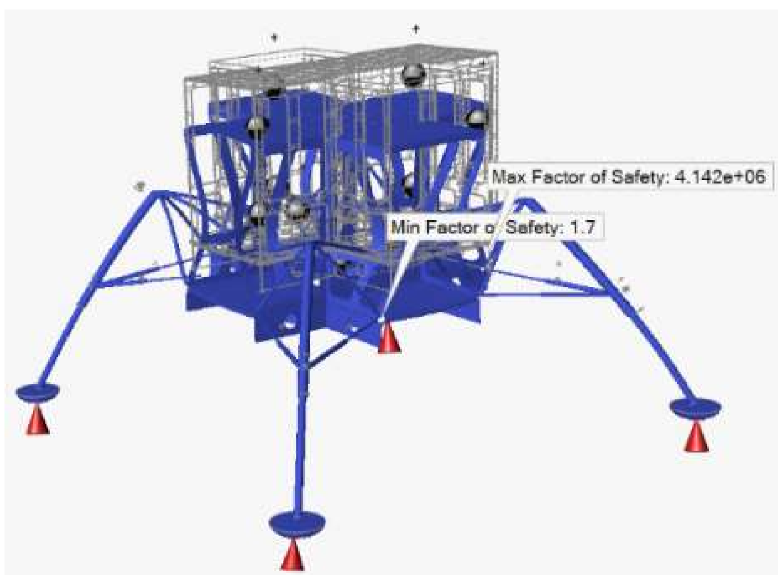
Table 3. The List of Components' Dimension and Mass in the Moon Strider.

Component	Qty.	Dimension [mm]	Mass [kg]
Spacecraft Bus	1	1207 (l) × 1207 (w) × 792 (h)	51.24
Landing Gear	4	20 (d) × 1070 (h)	23.93
Propellant Tank	2	420 (d) × 670 (h)	51.86
Propellant	2	-	200.00
Main Engine	1	400 (d) × 600 (h)	12.50
Camera	1	90 (l) × 60 (w) × 160 (h)	0.50
Flag mechanism	1	300 (l) × 40 (w) × 240 (h)	0.30
Additional payload	1	450 (l) × 211 (w) × 600 (h)	30.00
Thermal Control	1	-	18.00*
Attitude Determination & Control System	1	-	18.00*
Onboard Processing	1	-	12.00*
Pylontech US2000	1	442 (l) × 410 (w) × 89 (h)	24.00
Pylontech US3000	1	442 (l) × 420 (w) × 132 (h)	32.00
Solar Panel	2	500 (l) × 400 (w) × 25 (h)	10.00
S-band transceiver	1	160 (l) × 110 (w) x 44 (h)	1.25
X-band transmitter	1	86 (l) × 86 (w) x 50 (h)	0.50
S-band antenna	1	130 (d) × 160 (h)	0.50
X-band antenna	1	90 (d) × 115 (h)	0.54
Other	1	-	9.00*
TOTAL			**496.12**

l, w, h, d stands for length, width, height, and diameter; *mass value is assumed based on [1]

After making the CAD, the structure must be proven to survive in all the phases of the mission. The analysis is conducted with the finite element method to simulate 3 phases: static analysis on the Earth, static analysis in the launch vehicle, and modal analysis. The result is shown in Figure 7.



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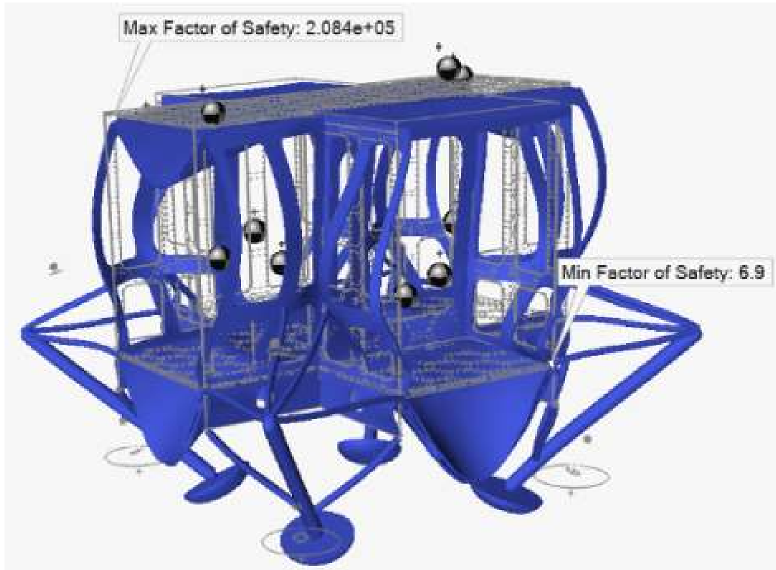

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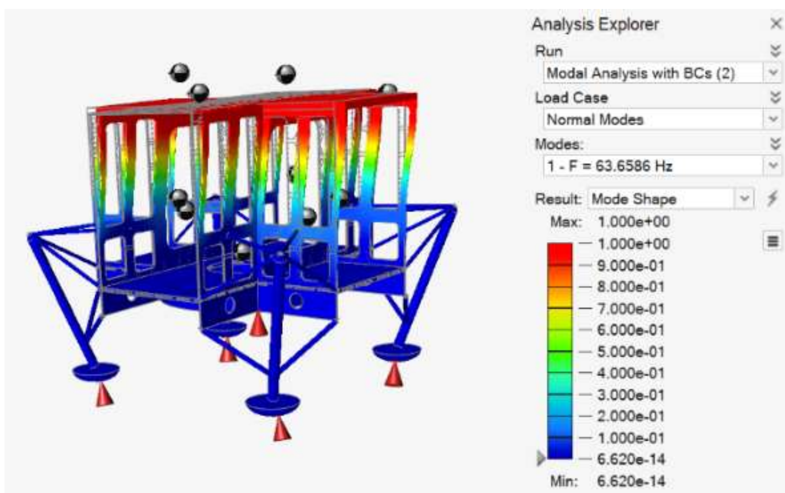

(c)

Figure 7. The results of (a) static analysis on the Earth, (b) static analysis in the launch vehicle, and (c) modal analysis by using finite element analysis software.

Based on the finite element analysis, the spacecraft's structure was proven to survive in two phases, which are the ground phase and launch phase, with a minimum factor of safety is 1.7 and the first natural frequency is 63.66 Hz.

3.3. Cost Estimating

Cost estimating is the iterative process of calculating the expected cost of a space mission. The cost models emulate the design process, production activities, and operations of space systems. The cost estimating tool used in this study is the Unmanned Space Vehicle Cost Model (USCM8). The USCM8 was developed by Tecolote Research for the US Air Force [1].

There are two types of costs in a space mission: non-recurring and recurring costs. Non-recurring costs are associated with the labor and material associated with designing, developing, fabricating, and testing a space vehicle qualification test model plus program-peculiar ground support equipment whereas recurring costs are associated with labor and material of fabricating, manufacturing, integrating, assembling, and testing of follow-on space vehicle flight hardware plus the effort associated with launch and orbital operations in support of the program [1]. For the Moon Strider mission, the cost estimation is based on the non-recurring cost because there will be only one spacecraft in this mission.

The cost estimation consists of the cost from spacecraft bus, payload, spacecraft integration, assembly, and test, program level, and launch vehicle. Spacecraft bus cost depends on each spacecraft subsystem hardware and software excluding the payload. Similar to spacecraft bus cost, the payload cost also depends on the payload (camera and flag mechanism) and its communication system hardware and software. However, the software cost for spacecraft bus and payload are excluded in this report. Integration, Assembly, and Test (IA&T) cost is the cost for integrating, assembling, and testing all space subsystems into an operable spacecraft and does not include the costs of integrating the bus with the payload. The program level cost is the cost needed for system engineering, program management, product assurance (quality control), government (regulation), and other customer oversight. The launch vehicle cost is the cost needed for using the launch vehicle including launch vehicle production, integrating with spacecraft and testing, and tracking launch vehicle performance during the mission. The transport cost from Jakarta, Indonesia to Kourou, French Guiana is included in the launch vehicle cost. The cost estimation is presented in Table 4.

Table 4. The Cost Estimating based on the USCM8.

Subsystem		Cost [×1,000 USD]
Spacecraft Bus	Structure and Thermal Control	14,362.04
	Attitude Determination and Control System	5,832.00
	Electrical Power System	5,594.10
	Propulsion	6,406.43
	Telemetry, Tracking, and Command	26,916.00
Communications Payload		16,326.81
Spacecraft Integration, Assembly, and Test		14,710.29
Program Level		32,182.72
Launch Vehicle	The Vega Launch Vehicle Services	37,000.00
	Shipping (from Jakarta to Kourou)	45.85
TOTAL		**159,376.24**

The expected cost for the Moon Strider mission is 160M USD. It should be noted that the cost estimation is calculated in FY2010. By assuming the inflation rate to be 3%/year, the cost in FY2021 is 220M USD or 3T IDR.

4. Discussion

The Moon Strider spacecraft has the main mission to take the photograph of Indonesia's flag with Earth globe on the background. Moon Strider uses the lander mission concept, so the entire spacecraft section lands on the Moon without using orbiter due to the simplicity of design and lower cost needed. This spacecraft will launch on 8 August 2026 and orbit at LLO on 14 August 2026 before descending and landing on 17 August 2026 to do its mission. The landing site is located at the nearside of the Moon with 30°N latitude based on scientific and engineering constraints. In carrying out its mission, Moon Strider consists of several subsystems such as propulsion payload, power, communication, and structure.

LDME is used for Moon Strider's main engine, which has been tested and used on NASA's Apollo program. The spacecraft also uses UDMH propellant which is ideal and optimal for lunar lander missions. However, the overall propulsion system could be simulated and designed in more detail to attain the most realistic representation.

The Moon Strider payload subsystem consists of Indonesia's flag and a camera that generate a picture with 134.3 arcseconds/pixel Earth globe size on it. This result does not meet the DRO requirement that stated Earth globe size should be better than 0.1 arcseconds/pixel. This happens due to the landing location of the satellite which is at a low latitude and the limitation of camera specification. Therefore, the iteration needs to be done with another camera or change the landing location into the upper latitude but still under the landing site recommendation.

Moon Strider is equipped with a solar panel that can provide 93 W and 88 W of peak power at the beginning of life and end of life. respectively. The batteries are sufficient to power the spacecraft from launch to LLO with a capacity of 5400 Wh. Considering the degradation factor, the power system can sustain power for 5.69 years, which is 569% of the mission lifetime expectancy. Future research might require other variables such as orbit eccentricity, inclination, power distribution and utilization, and more specific elements to obtain more detailed and accurate results.

The Moon Strider communication system consists of two S-band frequency links for uplink and downlink TT&C and one X-band frequency link for payload data downlink that is connected directly in real-time with the new ground station infrastructure at ITB. This communication system is designed based on its main mission to provide data transmission from spacecraft to ground station, therefore for the additional mission, the communication system only provides links for payload TT&C and data transmission too. Moreover, the extra link must be included for a more complex additional mission, for example, a rover exploration mission.

The structure subsystem requirements are mainly derived from the Vega user's manual. The finite element analysis shows that the first natural frequency is greater than the requirement stated. Furthermore, the static analysis on the Earth and launch vehicle has been done and showed that the structure is safe. However, the impact analysis when landing on the Moon has not been done and must be done next because the landing impact load could be the critical load for the Moon Strider's structure. Then, the detailed design, such as joint, mechanism, etc, is also important to be done for future works. The material could be optimized to get lighter weight by using composite materials.

The Moon Strider provides the opportunity to carry additional payloads and missions to execute. There is specifically 30 kg of mass within a dimension of 450 mm × 211 mm × 600 mm to fit inside the Moon Strider. They will also be able to utilize power up to 1 year with 88 W of peak power at EOL and even longer. The communication system provides links 1 and 2 for uplink and downlink payload TT&C and link 3 for payload data transmission with a maximum data rate of 129 kbps.

The expected cost for the Moon Strider's mission is 220M USD or 3T IDR. The cost value is reasonable compared to other Moon landers. However, the cost can be reduced by reducing the weight of the Moon Strider subsystem. The cost estimation has also included the cost of flight software and the income from the additional payloads entrusted to the Moon Strider spacecraft.

5. Conclusion

The conceptual design of the Moon lander spacecraft has been discussed from the mission concept, subsystems analysis, and cost analysis. That analysis is done based on the literature study and comparative study done before. Overall, the design fulfills the DRO except for the payload subsystem. It happened due to the quality of the picture is not good enough. Hence, there must be a further study in the payload subsystem. Many research and discussions can be built based on this study to make sure that the mission can be done safely in a more accurate way.

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Appendix

Appendix 1 shows the communication link budget calculated for the communication subsystem of the Moon Strider.

Appendix 1. The Communication Link Budget.

Design Element	Symbol	Units	Link 1	Link 2	Link 3
Link Frequency	f	GHz	2.1	2.2	8.0
Transmitter Power	P_{tx}	W	14	14	45
Transmitter Power	P_{tx}	dB·W	11.46	11.46	16.53
Transmitter					
Antenna Gain	G_{tx}	dB	42.25	0.00	0.00
Antenna Transmitter Losses	L_{tx}	dB	-0.5	-0.5	-0.5
Antenna Beamwidth	q_{tx}	°	180.0	360.0	360.0
Antenna Misalignment	a_{tx}	°	18	5	5
Alignment Loss	L_{qtx}	dB	-0.12	0.00	0.00
Equivalent Isotropic Radiated Power	$EIRP$	dB·W	53.10	10.96	16.03
Losses					
Propagation Path Length	S	km	384,400	384,400	384,400
Space Loss	L_s	dB	-211.00	-210.99	-222.21
Atmospheric	L_a	dB	-0.1	-0.1	-0.1
Polarization Loss	L_p	dB	-2.2	-2.2	-2.2
Total Losses	L	dB	-212.89	-213.29	-224.51
Receiver					
Antenna Gain	G_r	dB	0.00	44.66	55.87
Antenna Receiver Loss	L_r	dB	-0.5	-0.5	-0.5
Antenna Beamwidth	q_r	°	360.0	180.0	180.0
Antenna Misalignment	a_r	°	5	18	18
Alignment Loss	L_{qr}	dB	0.00	-0.12	-0.12
Total Receiver	G	dB	-0.50	44.04	55.25
Sky (Antenna) Noise Temperature	T_a	K	127	31	31
Receiver Temperature	T_r	K	60	85	85
System Noise Temperature	T_s	K	187	116	116
Receiver Merit	G/T	dB/K	-23.22	23.39	34.61
Powers					
Power Flux Density	f	dB·W/m^2	-127.59	-171.73	-166.66
Power Flux Spectral Density	f_f	dB·W/ (m^2·kHz)	-129.35	-175.71	-178.73
Carrier Power Received	P_{rx}	dB·W	-158.30	-158.29	-153.22
Noise Spectral Density	N_o	dB·W/Hz	-205.88	-207.95	-207.95
Carrier to Noise Density	P_{rx}/N_o	dB·Hz	47.58	49.66	54.73
Rates					
Data Rate	R	kBps	1.5	2.5	16.1
Data Rate	R	kbps	12	20	129
Eb/No	E_b/N_o	dB	15.82	15.68	12.66
Required Eb/No	-	dB	9.6	9.6	9.6
Required Margin	-	dB	6	6	3
Margin	-	dB	0.22	0.08	0.06

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