FEASIBILITY STUDY
OCTOBER 2015
MOONSPIKE

Revision history

Version	Date	Action
1.0	Oct 1, 2015	Initial release

1 Table of Contents

2 Acronyms & Terms

CRD	Critical design review.
Delta-v	Figure of merit for booster capability (equivalent acceleration in absence of external forces). Also known as Δv.
Dry mass	Vehicle plus contents minus propellant.
FTS	Flight termination system.
GNC	Guidance navigation and control.
GPS	Global positioning system.
IMU	Inertial Measurement Unit. Electronic device measuring rocket velocity and orientation.
I_{sp}	Specific impulse. A measure of the efficiency of rocket engines.
LEO	Low Earth orbit.
LV	Launch vehicle.
Max Q	Maximum aerodynamic stress on a vehicle in atmospheric flight .
MLI	Multi layer insulation (passive thermal control layers).
PDR	Preliminary Design Review.
TLI	Trans lunar injection. Spacecraft engine burn to transition from LEO to the Moon.
TTWR	Thrust to weight ratio.
WBS	Work breakdown structure.
Wet mass	Vehicle plus contents plus propellant.

3 Document Purpose

This document is a feasibility study for project Moonspike. The focus is to outline the technical requirements for reaching the mission goal and present a starting point for ongoing research and development. The reader is encouraged to give feedback on the subject matters in case questions arise or errors are found. Please go to the feedback section for more info.

The data disseminated in this document are subject to change in the future design and development process.

For Moonspike, a key success criteria is concluding that the project is possible within the limits of available technology, rocket design physics, trajectory analysis and simulation.

This review document does not address matters like legal issues, launch site, budget, manpower, production methods and supplier availability, although these and other matters are being addressed elsewhere within the project. Detail of e.g. structural analysis, separation systems and specific subsystems design are also for future analysis, although mass estimates are explored and used as part of the mass breakdown.

Moonspike would like to stress that we will be complying with international laws and ratified treaties to reach our mission goal and we are already coordinating our activities with appropriate regulatory agencies.

4 Mission Goal

The goal of the Moonspike project is to design, build and launch a vehicle to the Earth's moon within the next 5+ years within a relatively small but still realistic financial budget. A protected payload will primarily be customer-supplied data, held in a data store weighing one metric gram. Additional payload items may be added at a later date.

The project will have its technical development center in the Copenhagen area of Denmark, where the design and production of the vehicle will take place.

The Moonspike mission provides the following challenges and potential know-how:

- Launch capabilities to LEO
- LEO satellite insertion and communication
- Trans lunar injection and communication
- Planetary spacecraft capabilities
- A complete infrastructure for near Earth space exploration

5 Design Drivers

The technical solution shall not be rated for manned space flight, nor shall we compete in making rocket engines more effective. Rather, the design philosophy will be based upon utilizing a combination of proven technology, and new technology were it is found beneficial as well as feasible to produce and test.

As such, the technical approach is therefore based on the concept of realizing the physically smallest and technically most feasible solution, which constitutes the principle design driver for the development of the project.

To achieve our goals we have been looking at certain ways to ease the process which provides a significant advantage compared to rocket development 50 or even 10 years ago.

Examples:

- Advanced materials
- Composite tanks for fuel and oxidizer
- Computer aided simulations
- Easy information access
- 3d material processing
- Pistonless turbo pumps

6 Mission Trajectory and Delta-v

When launching for the Moon, it may seem obvious to fly straight towards it, for the shortest possible trajectory. This approach, known as direct ascent, is in fact possible, but turns out to have several disadvantages. The choice of payload and trajectory determine the requirement specification for the rocket, and is described below.¨

6.1 Trajectories

For the aforementioned direct ascent, the launch position of the rocket relative to the Moon is very restricted in order for the rocket and Moon trajectories to finally intersect. This effectively limits the daily launch window to a few minutes, after which the Earth’s rotation has turned the rocket too much off course.

A limited launch window is a severe logistical disadvantage, since any launch is affected by several factors that can cause delays, and range safety may prevent launch at any specific time.

To mitigate this problem, Moonspike will follow the traditional approach of first launching into LEO, then after a few orbits accelerating towards the Moon (TLI). The advantages are:

The launch from Earth is somewhat decoupled from the trajectory to the Moon. This both allows for a much wider daily launch window (hours rather than minutes), and a less restrictive selection of launch site coordinates.
During LEO, the spacecraft can transmit self-diagnostics and its measured orbital parameters back to radio base stations on Earth. Adjustments to the timing of the TLI engine burn (onwards to the Moon) can be adjusted based on these data and uploaded to the spacecraft before it happens. This approach separates the guidance challenge into two parts: Reaching LEO, then reaching the Moon, with lower overall performance requirements than a direct ascent.

The trajectory from LEO towards the Moon is known as a Hohmann Maneuver, by which the spacecraft transitions from LEO to an elliptical orbit, intersecting the trajectory of the Moon.

In general, the LEO orbit plane will not necessarily coincide with the Moon orbit plane (it depends on launch site coordinates and/or excess delta-v capability to execute plane change maneuvers). Due to this, and to expected inaccuracy of the TLI burn, course corrections will be necessary along the path onwards to the Moon.

The figure below shows the overall trajectory from the Earth to the Moon:

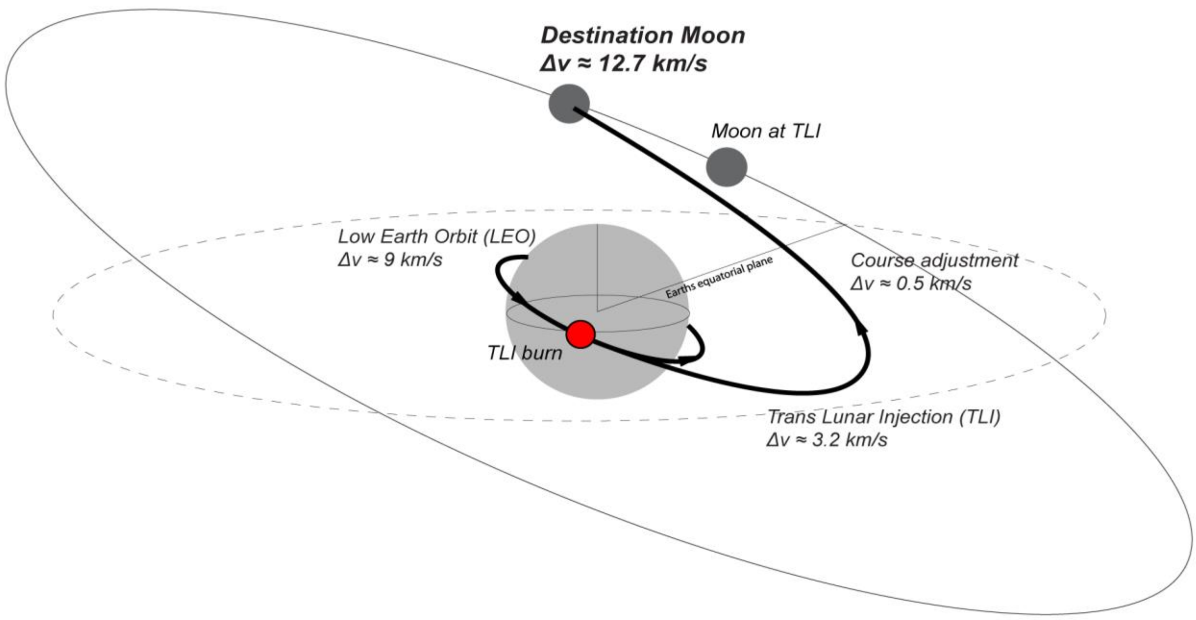


Figure 1 - Trans Lunar Injection (TLI). The TLI burn is performed at LEO to match arrival of the Moon.

The three main parameters needed for mission success are:

- Thrust, power, known as change in speed or delta-v or Δv (approx. 12.7 km/s total).
- Precision, in guidance, navigation and control, to hit a small target app. 400,000 km away (approx. 0.25 degrees pointing precision or better).
- Proof of transit to and arrival at the Moon.

6.2 Delta-v

Regarding the first bullet point above, a delta-v calculation for early estimates is done using:

$$\Delta v = ln\left(\frac{M_{start}}{M_{end}}\right) \cdot I_{sp} \cdot 9.81\frac{m}{s^2}$$ [1]

M_{start} is defined as the combined current rocket mass with propellant
M_{end} is defined as the combined current rocket mass after using propellant
I_{sp} is engine specific impulse in seconds.

6.3 Estimated Mission Delta-v Budget

Mission milestone	**Delta-v**
Launch → LEO	9 km/s
LEO → Moon	3.2 km/s
Mid course corrections	0.5 km/s
Total delta-v estimated	12.7 km/s

LEO altitude must be high enough to limit drag-induced altitude loss, e.g. 200 km. It should be noted that while this is a small fraction of the distance to the Moon, it takes most of the total delta-v to get there. The ±200 km is well below steady LEO, providing free and safe maneuvering options away from the majority of orbiting satellites.

7 Spacecraft

The spacecraft, which includes the last stage providing the TLI burn, is important to design in detail to understand all components and subsystems that drive the final mass. The mass of the full upper stage is the mass that needs to be taken into LEO, and is by definition the actual capacity of the main launcher system.

The spacecraft shall meet the following overall requirements:

- Spacecraft shall generate the necessary delta-v to reach the Moon from LEO (approximately 3.7 km/s).
- Spacecraft shall provide payload carrying capability consistent with mission objectives.
- Spacecraft shall provide proof of reaching the Moon using transmitted low bit rate imagery.
- Spacecraft shall provide attitude control capability (eg. yaw/pitch/roll thrusters), in order to control its orientation throughout the different mission phases.
- Spacecraft shall utilize a restartable main engine system to perform the TLI burn, and later mid-course corrections.
- Spacecraft shall provide suitable mechanical and electrical interfaces to the launch vehicle.
- Spacecraft shall utilize non-cryogenic propellants, as the expected nominal mission time will otherwise result in excessive propellant boil-off.
- Spacecraft shall feature microgravity compliant fuel control, e.g. tank bladder systems.
- Spacecraft shall feature thermal control.

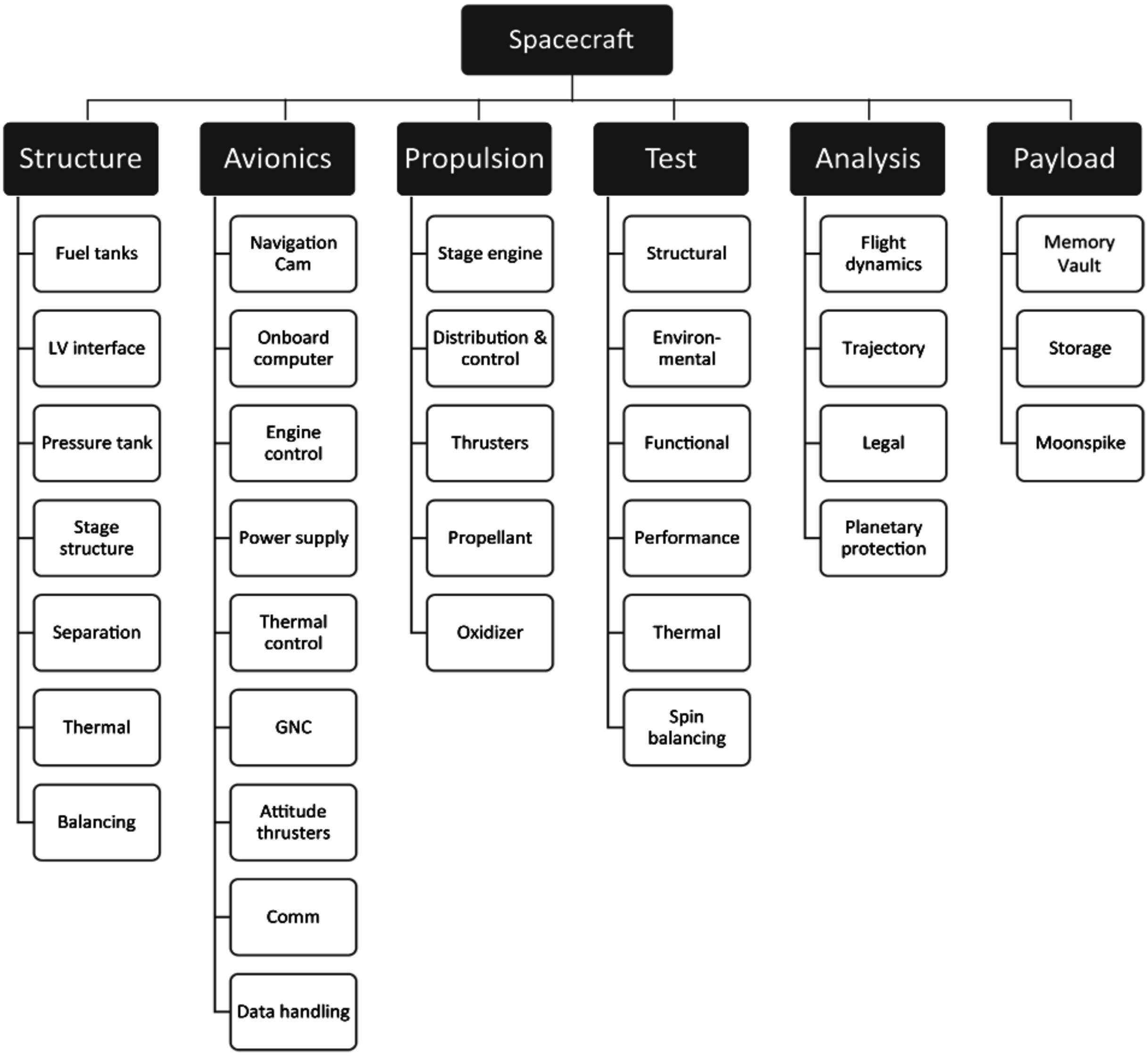
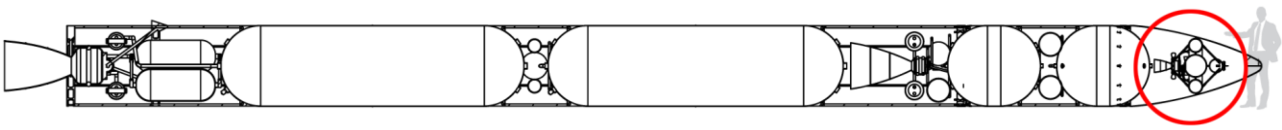


Figure 2 - Work breakdown structure, spacecraft

7.1 Spacecraft Design

The figures below show CAD drawings of the first iteration of the spacecraft design:



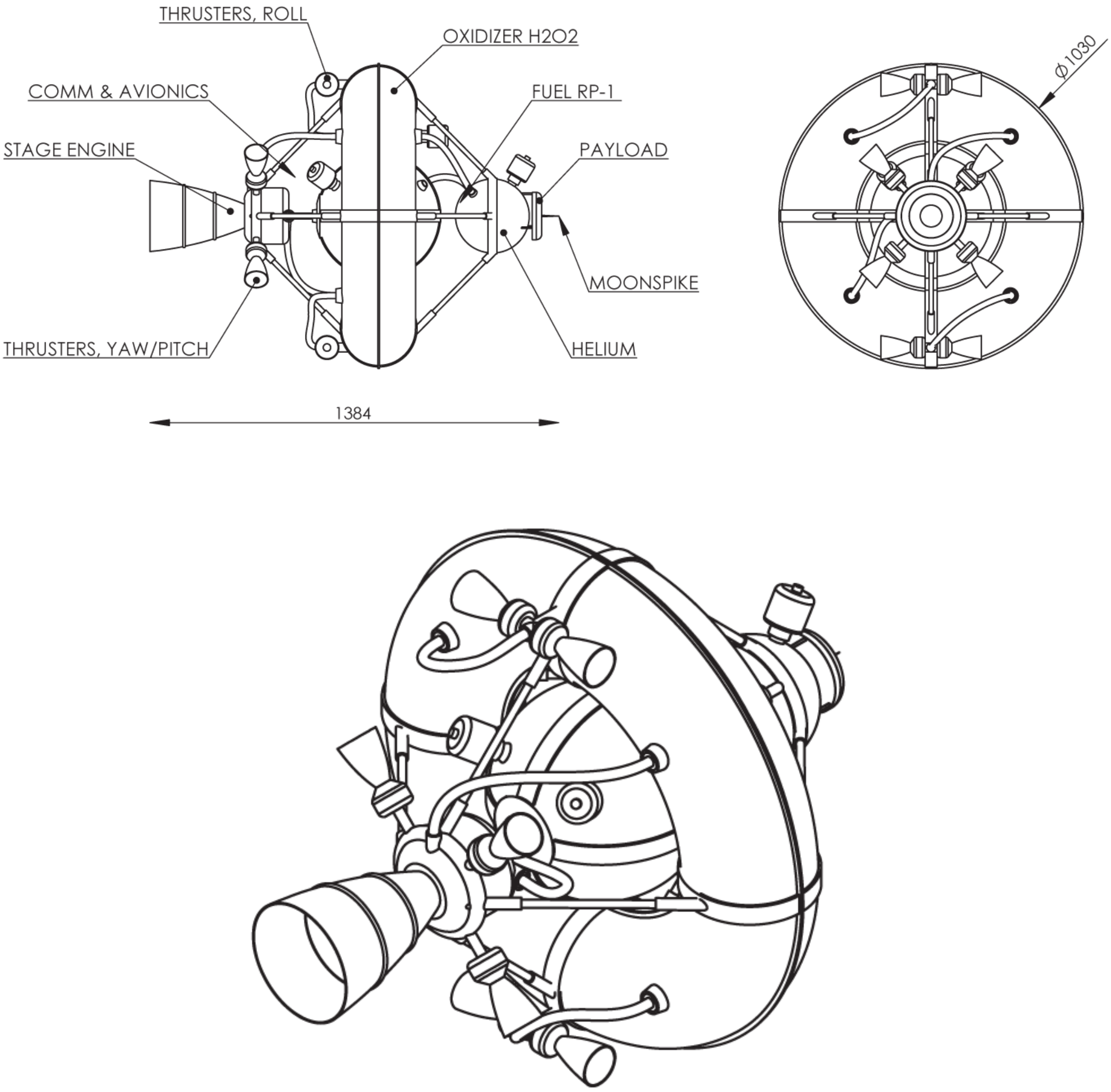


Figure 3 - First iteration of space craft design.

7.2 Propulsion

The spacecraft main engine design aims to realize a highly reliable and restartable bi-propellant propulsion system utilizing non-cryogenic propellants. The former to enable trajectory corrections after TLI and the latter to ensure thermal compliance for extended mission durations. The engine design baseline employs a propellant combination consisting of Hydrogen Peroxide (85%) as oxidizer and refined Kerosene as fuel in a nominal O/F mixture ratio of 7.5:1. The following illustration depicts the intended engine chamber and nozzle geometry while the associated table provides key design parameters.

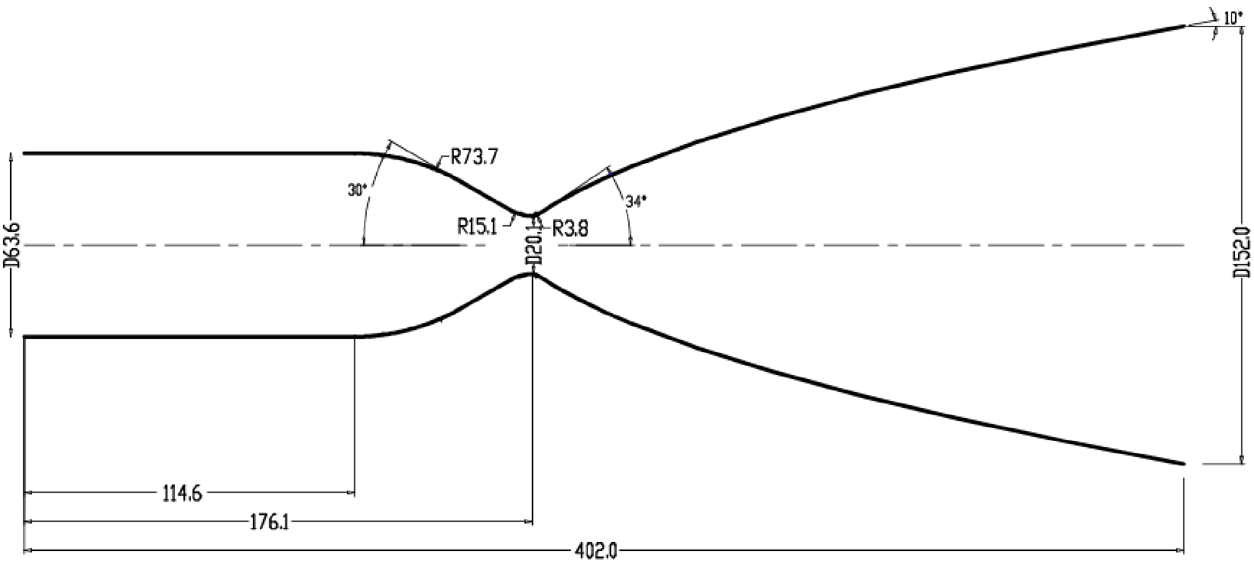


Figure 4 - Space craft main engine.

Parameter	Value	Parameter	Value
Oxidizer	H_2O_2 (85%)	Chamber pressure (nom.)	5.0 Bar
Fuel	Kerosene	Combustion Temperature	2550 K
O/F Mixture ratio	7.5:1	Effective Exhaust Velocity (vacuum)	2800m/s
Oxidizer mass flow rate	0.0946 kg/s	Specific impulse (vacuum)	286 s
Fuel mass flow rate	0.0121 kg/s	Chamber thrust (vacuum)	300 N

7.3 Spacecraft Data

Total mass: 157 kg
Max diameter: 1.03 m
Length: 1.38 m
Propellant mass: RP-1, 30 kg
Oxidizer mass: Hydrogen peroxide, 80 kg
Main engine I_{SP}: 286 s
Main engine thrust: 300N
Attitude control: Hydrogen peroxide thrusters
Fuel pressure feed: Helium
Thermal protection: Spin stabilization and MLI
Navigation: IMU and Moon, Earth and Sun proprietary digital sighting system
Course adjustment: Main engine burn & attitude control
Power: Battery
Lunar COMM: 40 cm parabolic dish
LEO COMM: low-directivity antennas

Spacecraft, mass estimate	Qty	kg	Total kg
Engine	1	2.5	2.5
Thrusters, y/p/r	8	0.25	2
Camera navigation	8	0.25	2
Pressure transducer	3	0.1	0.3
Solenoid, thrusters	8	0.4	3.2
Solenoid, engine	4	0.4	1.6
COMM, dish	1	0.5	0.5
Temp control	1	1	1
Avionics main systems	2	0.8	1.6
Wire, electronic	10	0.2	2
Piping, propellant	4	0.3	1.2
Battery	1	3	3
Tank, propellant	1	3	3
Tank, H202	1	5	5
Tank, helium	1	1.5	1.5
Helium	575	0.179	0.103
Structure	1	6	6
Margin 15%			5.475
Propellant,	1	30	30
Oxidizer, H2O2	1	85	85
Total dry mass, stage 3			**41.98**
Total wet mass, stage 3			**156.98**

The spacecraft shall be autonomous in the sense, that it shall be possible but not required to send telecommands from Earth to the spacecraft in order to perform TLI or any mid course corrections.

In addition the spacecraft shall have an autonomous attitude control system enabling the possibility of automatic pitch, yaw and roll control.

7.4 Spacecraft Components

Figure 5 provides an overview of the spacecraft main components and their interfaces.

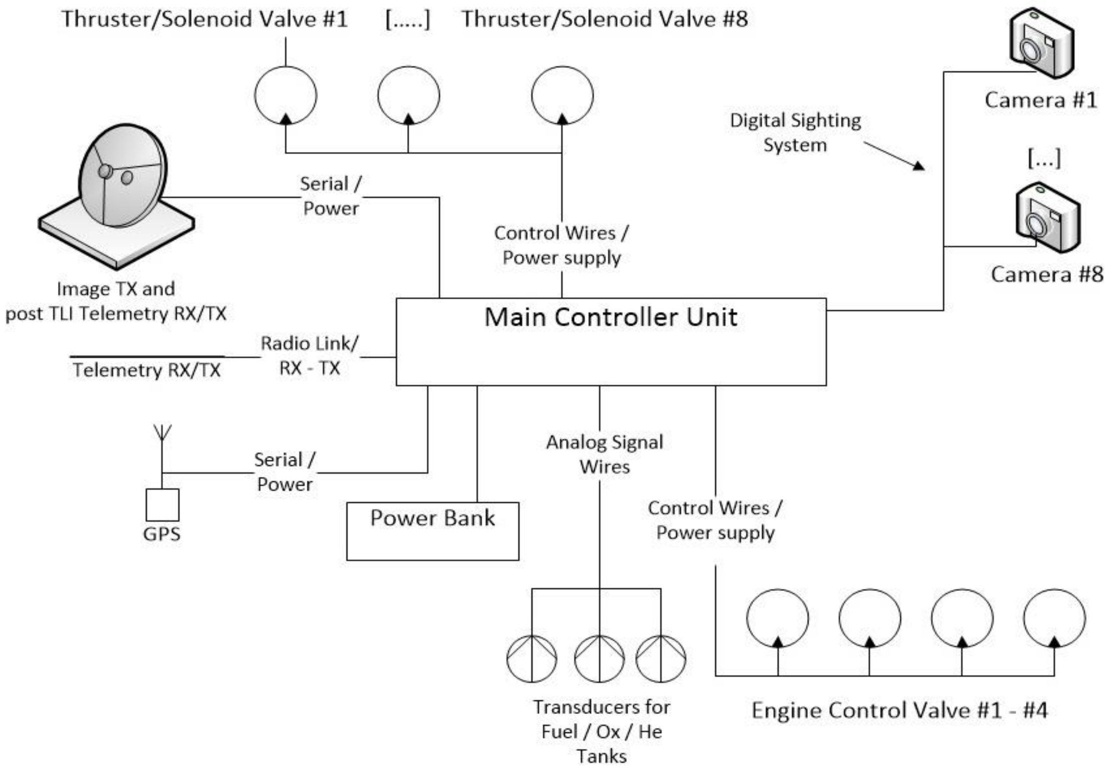


Figure 5 - Preliminary Spacecraft avionics logical overview.

8 Launch Vehicle

The Moonspike launch vehicle constitutes a rocket system capable of transporting the spacecraft to a suitable LEO parking orbit, from where the spacecraft will subsequently inject into a Moon intersecting trajectory. Ascending from ground to LEO represents a technical challenge requiring high performing rocket engine systems and minimal structural mass in the vehicle itself, wherefore we plan to make use lightweight composite technologies to a significant extent.

Overall, the launch vehicle shall meet the following requirements:

- Vehicle shall generate the necessary delta-v to reach the LEO (app 9 km/s) while carrying the Moonspike spacecraft.
- Vehicle shall achieve an initial thrust to mass ratio > 1.3, thus increasing the envelope of acceptable launch weather conditions.
- Vehicle shall contain a 3-axis attitude control system (gimbal for yaw/pitch and thruster for roll). In order to control the vehicle during the ascent phase and compensate for environmental disturbances such as wind or atmospheric density variations, an attitude control system is necessary.
- Vehicle shall feature an aerodynamic shroud to protect the spacecraft during the trans-atmospheric part of the ascent to LEO..
- Vehicle shall interface to launch ground systems.

- Vehicle shall include propellant pump system to achieve necessary engine I_{sp} performance.
- Vehicle shall include ullage engine systems or similar functionality to prevent flame out of second stage engine upon ignition.
- Vehicle shall include a fault tolerant flight termination system (FTS).

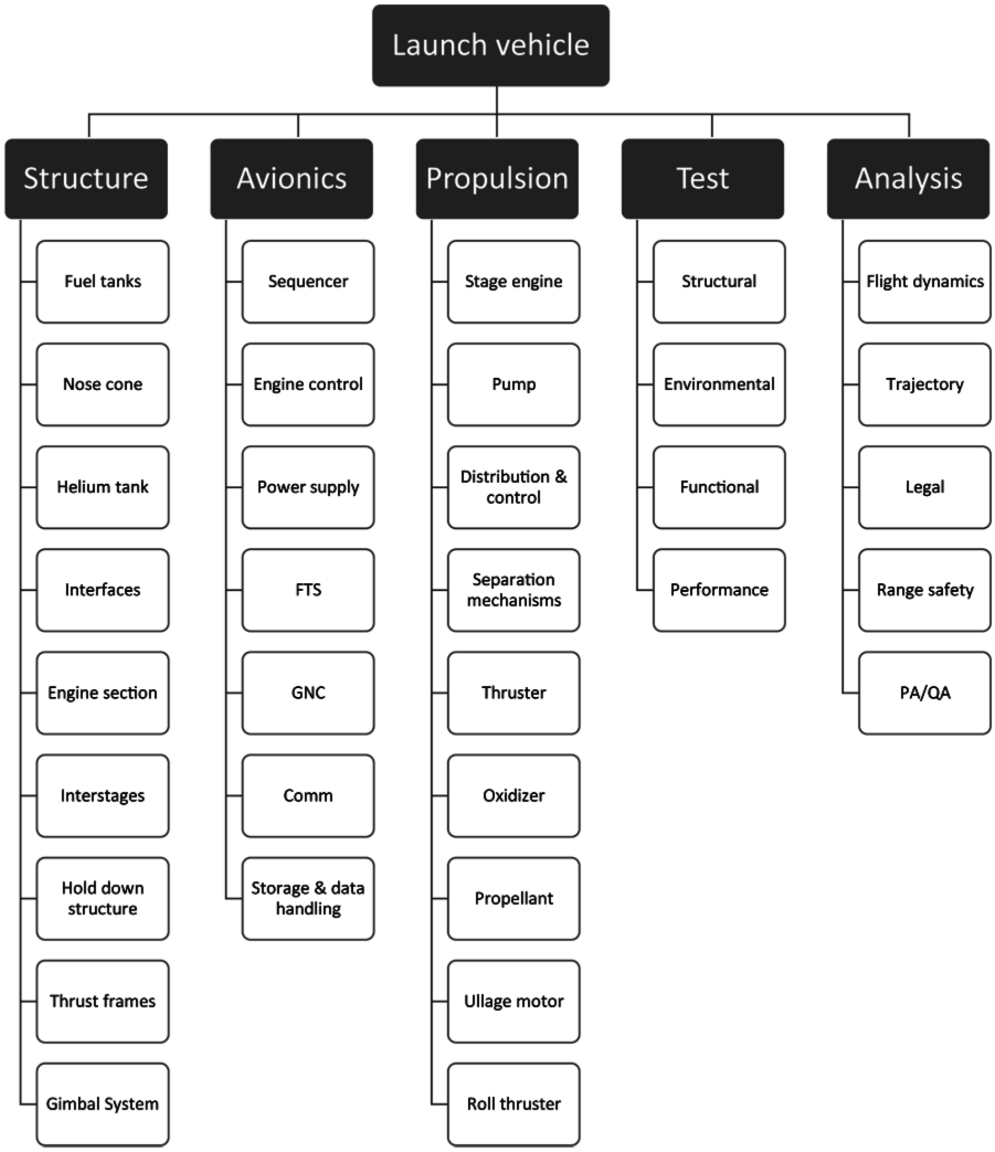


Figure 6 - Work breakdown structure, combined launch vehicle

8.1 Staging Philosophy

Launch vehicles mostly employ 2 or 3 booster stages to reach LEO. Using 3 stages has the theoretical advantage of accelerating lower average mass, as empty fuel tanks are jettisoned (slightly) more often. This approach comes with the disadvantage of increased total mass for an extra engine and interconnect structure. The delta-v budget for Moonspike has been calculated for both a 2 and a 3 stage design, and turned out in favor of two stages (lowest launch mass).

8.2 Propulsion

The current baseline design of both stages in the Moonspike launch vehicle makes use of a single cryogenic rocket engine burning liquid oxygen (LOX) as oxidizer and Ethanol as fuel. The propellant combination, although not the most energetic available, has the very significant advantages of being non-toxic as well as comparatively easy to handle, transport and procure. (However, it should be noted that the trade on LOX/Ethanol vs. LOX/Kerosene is still open at this time).

The baseline first stage main engine design is pump-fed (pistonless) and makes use of an ablatively cooled chamber and nozzle with the following geometric and operating point characteristics: The engine is designed to operate at medium chamber pressures to reduce pump, structural and engine cooling requirements while allowing industrial standard valves, actuators, sensors etc. to be utilized in the design of the engine feed system.

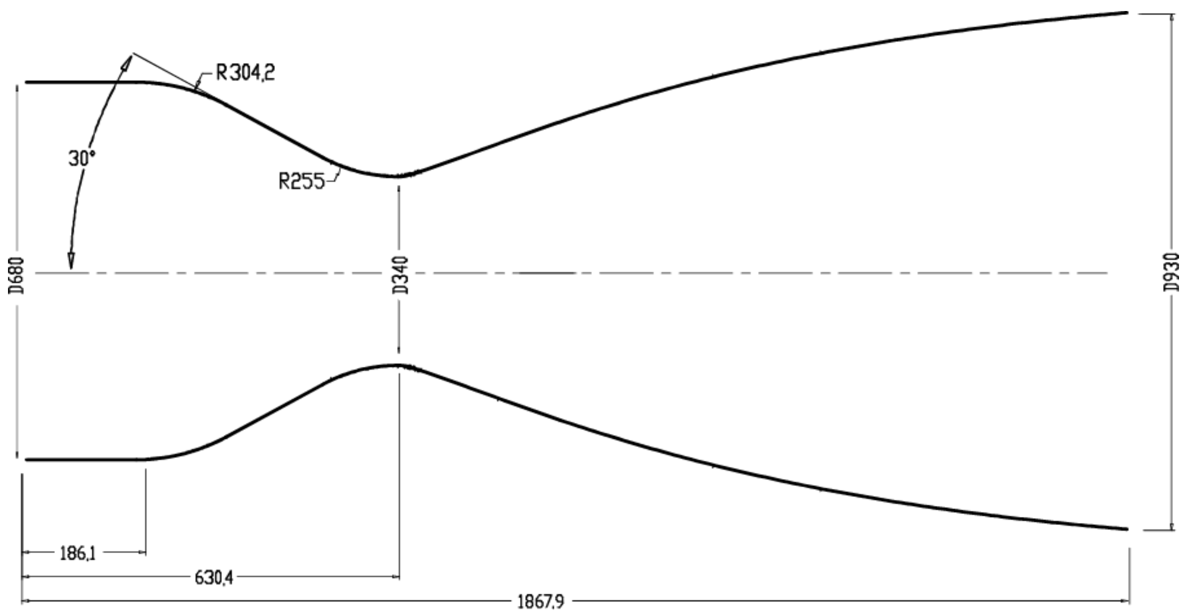


Figure 7 - First stage main engine.

Parameter	Value	Parameter	Value
Oxidizer	LOX	Chamber pressure (nom.)	25 Bar
Fuel	Ethanol (95%)	Combustion Temperature	3210 K
O/F Mixture ratio	1.5	Effective Exhaust Velocity (vacuum)	2825 m/s
Oxidizer mass flow rate	80.1 kg/s	Specific impulse (vacuum)	288.2 s
Fuel mass flow rate	53.4 kg/s	Chamber thrust (vacuum)	377 kN

The second stage engine likewise incorporates ablative cooling in the design of both chamber and nozzle. As the engine will be optimized towards operating in near-vacuum high altitude conditions it makes use of a nozzle system with significantly larger expansion ratio, when compared to the first stage engine. Similarly, the operating environment for the second stage engine allows lower chamber pressures to be utilized while maintaining high efficiency, which in turn opens the possibility for building the second stage of the launch vehicle as a pressure-fed rather than a pump-fed system. At present, both solutions are being evaluated, but for the current analysis a pump-fed solution has been utilized. The baseline second stage engine geometry and operating point characteristics are as follows:

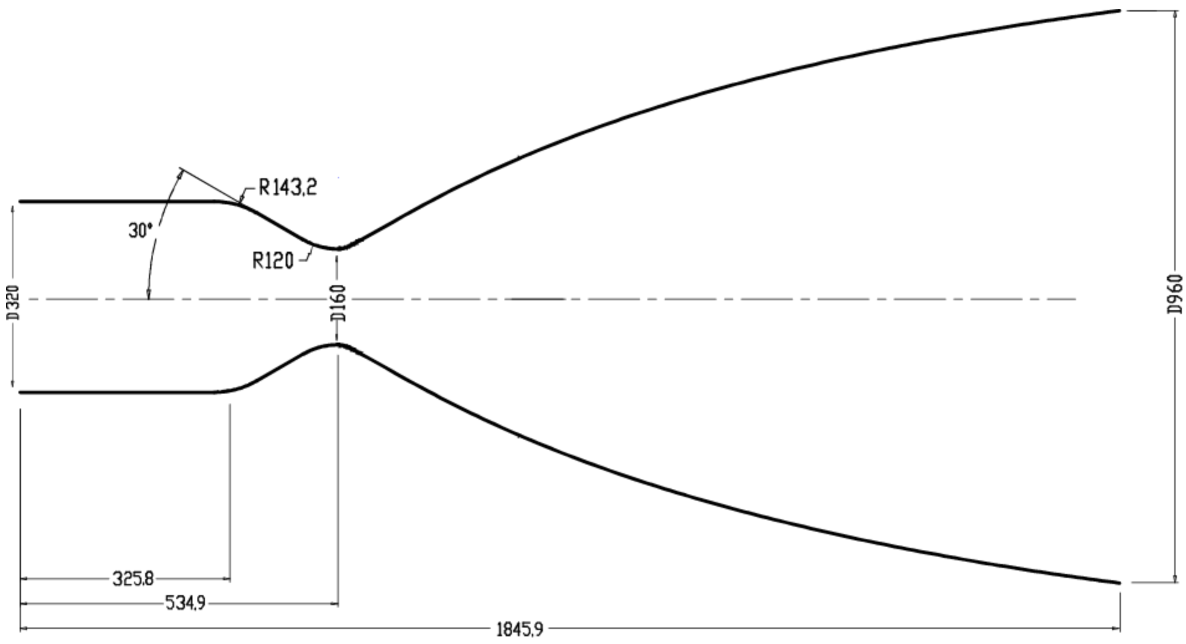


Figure 8 - Second stage main engine.

Parameter	**Value**	**Parameter**	**Value**
Oxidizer	LOX	Chamber pressure (nom.)	8.0 Bar
Fuel	Ethanol (95%)	Combustion Temperature	3100 K
O/F Mixture ratio	1.5	Effective Exhaust Velocity (vacuum)	3080 m/s
Oxidizer mass flow rate	5.8 kg/s	Specific impulse (vacuum)	306 s
Fuel mass flow rate	3.8 kg/s	Chamber thrust (vacuum)	29 kN

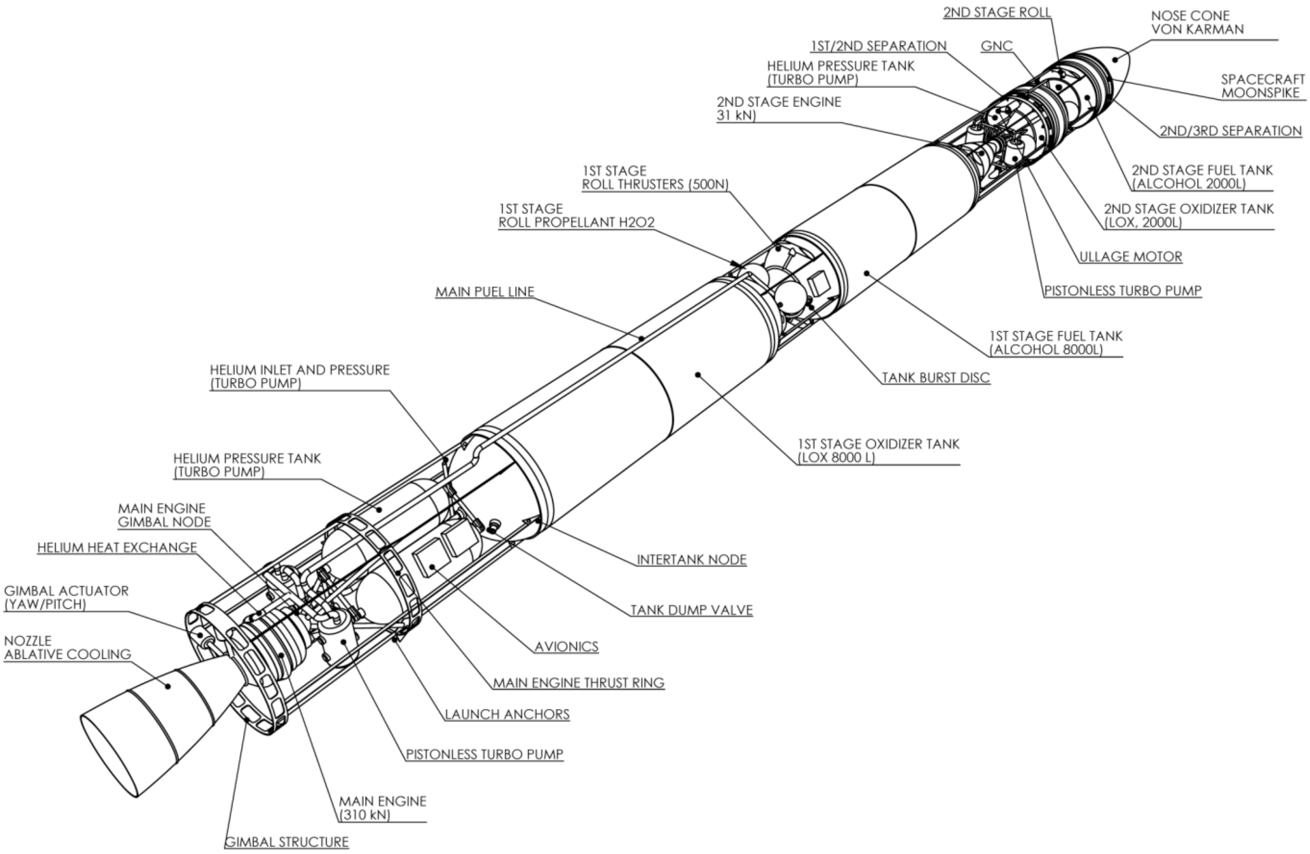


Figure 9 - Perspective of the total rocket. Total length 23.4 meters and diameter 1.5 m.

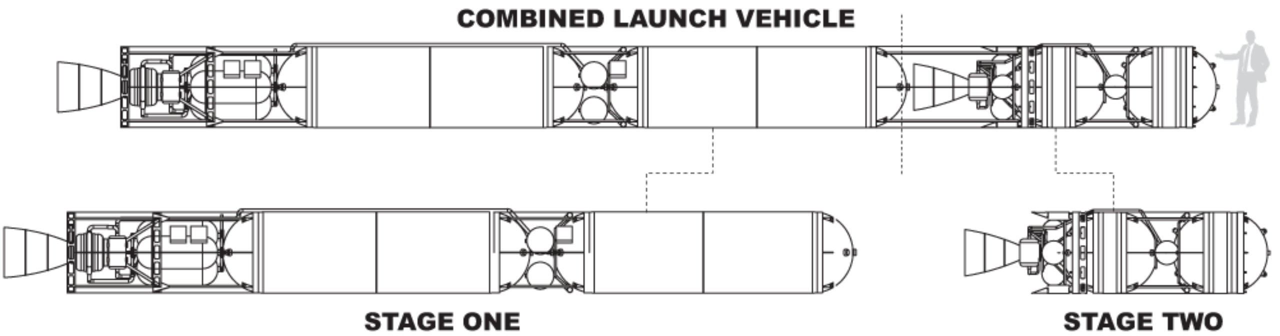


Figure 10 - Combined launch vehicle

8.3 Launch Vehicle Data

LV Stage One Data
Total mass: 18,012 kg
Max diameter: 1.5 m
Total length: 15.7 m
Propellant mass: Alcohol, 6,400 kg

LV Stage Two Data
Total mass: 4,460 kg
Max diameter: 1.5 m
Total length: 5.6 m
Propellant mass: Alcohol, 1,600 kg

Oxidizer mass: LOX, 9,600 kg
Main engine I_{sp} (vacuum): 288 s
Main engine thrust (SL): 377kN
Main engine burn time: 120 sec
Yaw/pitch control: Gimbal
Roll control: Hydrogen peroxide thrusters
Fuel pump system: Piston-less turbo pumps
Fuel pressure feed: Helium, 59 kg

Oxidizer mass: LOX, 2,400 kg
Main engine I_{sp} (vacuum): 306 s
Main engine thrust (vacuum): 29kN
Main engine burn time: 420 sec
Yaw/pitch control: Gimbal
Roll control: Hydrogen peroxide thrusters
Fuel pump system: Piston-less turbo pumps
Fuel pressure feed: Helium, 17,6 kg
GNC: IMU
Power: Battery

Stage 1, Mass estimate	**Qty**	**kg**	**Total kg**
Engine	1	500	500
Turbo pump piston less	1	40	40
Thrust Vector Control, actuator	2	5	10
Thrust Vector Control, structure	1	30	30
Avionics engine sections	1	8	8
Roll, thrusters	4	2	8
Roll, tanks	1	15	15
Roll, propellant	1	20	20
Tank LOX (carbon)	10.56[1]	38[2]	401.28
Tank Propellant (carbon)	7.04	38	267.52
Tank 200 b helium (carbon)	1.65	100[3]	165
Helium HP turbo pump	330	0.179	58.94
Piping	25	0.5	12.5
Avionics top section	1	7	7
Additional structure	4	30	120
Nose cone	1	100	100
Margin 15%			249.5
Propellant, Alc (40%)	1	6,400	6,400
Oxidizer, LOX (60%)	1	9,600	9,600
Total dry mass, stage 1			**2,012.72**
Total wet mass, stage 1			**18,012.72**

[1] LOX volume added 10% ullage. Same for “Tank Propellant”.
[2] Estimated 38 kg low pressure structure per m3 content. Same for “Tank Propellant”.
[3] Estimated 100 kg high pressure structure per m3 content.

Stage 2, mass estimate	Qty	kg	Total kg
Engine	1	80	80
Turbo pump pistonless	1	20	20
Guidance	1	10	10
Engine Avionics	1	2	2
Roll, thrusters	4	0.5	2
Roll, tank	1	5	5
Roll, propellant	1	5	5
Tank, propellant (carbon)	1.76	40	70.4
Tank, LOX (carbon)	2.64	40	105.6
Tank, Turbo pump (carbon)	0.413	80	33
Helium, for turbo pump	99	0.179	17.68
Structure	2.5	20	50
Margin 15%			60.10
Propellant (40%)	1	1600	1600
Oxidizer (60%)	1	2400	2400
Total dry mass, stage 2			**460.78**
Total wet mass, stage 2			**4,460.78**

Combined delta-v budget

Stages	Mass combined	Kg	Δv km/s	Total Δv km/s
LV stage 1	Wet	22,474		
	Dry	6,474		
			3.5	
LV stage 2	Wet	4,618		
	Dry	618		
			6.0	**9.5**
Spacecraft	Wet	157		
	Dry	42		
			3.7	**13.2**

9 Avionics

The Moonspike avionics comprise a set of generic subsystems which to a large extent can be utilized in both the Spacecraft and the launch vehicle, thus minimizing development costs.

The Spacecraft carries a range of electrical systems in accordance with figure 5, whereas the launch vehicle contains an additional set of avionics as described below:

9.1 Computers

Engine control units (ECU) for 1st and 2nd stage engines are an integral part of each engine. This minimizes wiring between the ECU and engine sensors and valves, including pistonless turbo-pumps (but not gimbal actuators), and is necessary for the engine to be developed and tested as a unit.

Guidance and Navigation Computer (GNC) for the ascent to LEO (1st and 2nd stage) is located in the 2nd stage, and controls the engine gimbals of both 1st and 2nd stage engines. This is to reduce the overhead of a separate 1st stage system, and to avoid a navigation switch-over when the 1st stage is jettisoned.

9.2 Navigation

Both the ascent GNC (located on launch vehicle 2nd stage) and spacecraft GNC will incorporate inertial measurement units. After 2nd stage jettison the spacecraft GNC will take over, use GPS[4] for position calibration, and a proprietary digital sighting system to compensate gyro bias drift. After TLI, navigation drift is compensated solely based on camera images, which besides orientation can provide coarse position estimates based on the size of the Moon and Earth in the images.

9.3 Communications

The spacecraft radio link has two modes

	LEO	**After TLI**
Antenna	low directivity	dish antenna
Bit rate	60 kbit/s	3 kbit/s
TX power	5W	50W
Range	~2,000 km	~400,000 km

The LV will employ a separate radio system, with specifications similar to the spacecraft LEO mode.

[4] Using techniques to mitigate legacy height/velocity limits on GPS

During ascent and LEO, quarter-wave dipole whip (or similar) low-directivity antennas are used. This allows for arbitrary angles between the LV / spacecraft and receiver antenna on Earth, and for maximum communication time per overpass during LEO.

After the TLI burn, the spacecraft radio (TX and RX) switches to a directional dish antenna. The attitude of the spacecraft is controlled to keep the dish pointing approximately towards the Earth (note this does not imply a specific direction of travel), and transmit power is switched from 5W to 50W. This, combined with a low bitrate, will enable the Moonspike spacecraft radio to achieve the necessary 400,000 km range (link budgets below).

LEO link budget

Transmitted power	+37 dBm	5W effective
TX antenna gain	0 dBi	Insensitive to orientation
TX losses	-1 dB	
Path loss	-158 dB	2,000 km @ 900MHz
RX antenna gain	+18 dBi	Moonspike enthusiast friendly (ø1.2m dish, $300)
RX losses	-1 dB	
Fading margin	-6 dB	RX ant. misalignment, atmosphere
Received power	-112 dBm	
Receiver noise floor	-121 dBm	100kHz BW, 3 dB NF, room temp
Receiver SNR	9 dB	

Moon trajectory link budget

Transmitted power	+47 dBm	50W effective
TX antenna gain	+8 dBi	40 cm parabolic, aperture efficiency 0.5
TX losses	-1 dB	
Path loss	-204 dB	400,000 km @ 900MHz
RX antenna gain	+34 dBi	e.g. ø8m dish, 53% eff.
RX losses	-1 dB	
Fading margin	-9 dB	TX and RX ant. misalignment, atmosphere

Received power	-126 dBm	
Receiver noise floor	-136 dBm	4kHz BW, 2 dB NF, room temp
Receiver SNR	10 dB	

Until and including the first day after TLI, the telemetry can be received with relatively inexpensive hardware. As the spacecraft approaches the Moon, collaboration with at least 3 facilities around the world with larger dish antennas will be necessary.

9.4 Power Supply

The power budgets do not rely on solar panels. Initial estimates indicate that the spacecraft power budget (below) can be covered entirely by batteries, at a lower mass penalty than the combination of a smaller battery pack and a solar charging system. Like the ascent GNC, batteries for 1st and 2nd stages are located in the 2nd stage. Placing part of these batteries in the 1st stage would theoretically reduce vehicle mass after 1st stage jettison, but the gain would be small due to the short burn time of stage 1. Conversely, placing all batteries in stage 3 (the spacecraft) by the same argument, would be too wasteful especially because stage 1 and 2 need much higher power capability for the pistonless turbopumps.

9.4.1 Spacecraft power budget

Spacecraft power usage is controlled by switching subsystems ON and OFF as needed during the mission. During most of the ~4 day journey from LEO to the Moon, only basic navigation and monitoring systems need to be active (idle power mode). At fixed intervals, cameras will be turned on, and their pictures processed to compensate navigation drift (normal power mode). Radio transmitters are turned on for frequent telemetry transmissions during ascent and LEO, and less frequent during TLI, at the respective transmit power levels. Following each telemetry transmission, the radio receiver is kept ON for a fixed duration, to listen for commands from the earth.

Mission Phase →		**LEO**	**After TLI**	
Power mode ↓	watts	50 hours	96 hours	notes
Idle	1	40	88	Inertial navigation, monitoring
Normal	10	5	4	Normal w. cameras, image processing
Radio RX	15	1	1	Normal +5W radio RX power
Radio TX (LEO)	21	2	-	Normal +5W TX power @ 45% efficiency

Radio TX (TLI)	121	-	3	Normal +50W TX power @ 45% efficiency
Total Wh		147	506	

Spacecraft main engine and thruster burns have relatively short duration and little effect on the power budget. Power for pumps, controls and gimbal actuators on the 1st and 2nd stage engines are supplied from 2nd stage batteries.

Adding 10% conversion losses and another 10% margin to the 147 + 506 Wh total yields 790 Wh battery capacity needed for the spacecraft.

The spacecraft battery system does not need to be rechargeable; high energy density batteries can be utilized, providing about 500 Wh per kg. This is ~1.6 net kg battery mass, leaving another 1.4 kg for a vacuum/thermal environment compatible enclosure, converters and power distribution system.

10 Proof of Reaching the Moon

At 3 kbit/s, a 20 kbyte compressed low-resolution B/W picture as shown below can be sent back to Earth in a few minutes, along with spacecraft telemetry data.

Figure 11 - The Moon at a 4000 km distance, 90x90 degree field-of view, Ecliptic North up, 430x430 pixels, image data size 20 kbytes (the Moon will be about half full in actual pictures).

4,000 km distance is about a half hour before arrival, during which the Moon will move to the center of the frame. During this time span, the Spacecraft will transmit as many pictures as allowed by its power budget.

These pictures also serve a strictly technical purpose as input to the spacecraft navigation system. The distance to the Moon can be estimated from its size in the picture, until it finally fills the frame completely. The loss of signal at the expected time will serve as final indication of reaching the Moon.

11 LEO Ascent Trajectory Analysis

The trans-atmospheric ascent profile of the Moonspike launch vehicle from ground to LEO covers the largest delta-v segment of the proposed mission, hence it also represents the largest technical liability in terms of realizing vehicle performance levels.

The nominal mission profile requires the two stage launch vehicle to be capable of injecting the spacecraft into an orbit featuring altitude characteristics compatible with extended LEO duration, so as to provide margin for ascertaining correct phasing between TLI maneuver initiation point and the position of the Moon in its orbit plane.

As such, a parameter variation study has been performed to assess vehicle performance margins over the ascent trajectory profile and to verify a minimum 200kg payload LEO capability of proposed launch vehicle design independent of launch site location in the low and mid latitude ranges, thereby enabling a wider range of potential launch sites.

11.1 Analysis Prerequisites

The analysis includes all major factors pertaining to launch vehicle <-> environment interactions, including gravity modeling (EGM2008), atmospheric properties (ISA, NRLMSISE-00 at maximum solar activity), vehicle aerodynamics, engine performance and spacecraft ballistic coefficient. Combustion efficiency of 1st and 2nd stage engine systems are estimated at 94%. Structural elements of the launch vehicle design contain a 15% mass margin. Payload mass (spacecraft) is fixed at 200 kg (approximately 20% margin for the preliminary design).

So as to minimize fairing mass and vehicle structural stresses, the launch trajectories considered are further optimized toward limiting the maximum dynamic pressure to 45 kPa. Trajectory profiles enabling direct LEO injection have been favored in this analysis, thus removing the need for a second stage engine restart capability to achieve orbit circularization and the associated mass and system level complexity penalties inherently incurred by such a system (However, it should be noted that the trade on direct injection vs. two-burn injection remains open at this time). Using the direct injection method to produce a prograde circular orbit to ease requirements for launch timing and TLI execution is though a known GNC challenge, wherefore the mission plan seeks to accommodate extended LEO mission segment durations to enable optimal orbit phasing for a larger range of LEO eccentricities.

To demonstrate the margined capabilities of the proposed launch vehicle, two end-to-end performance envelope simulations for the ascent and LEO phases of the mission are presented in the subsequent sections.

11.2 Case 1: Low Latitude Launch

The first case considers a notional low latitude launch from the Broglio Space Center located at 2.9383°S, 40.2125°E. The launch azimuth is set to 111°, thus achieving reasonable range safety conditions for the first stage drop zone over the Indian Ocean while increasing the LEO inclination and thereby easing requirements on TLI and subsequent correction maneuvers. Also this azimuth provides decent land based tracking opportunities during the ascent phase. The simulated attitude controller for the launch vehicle first stage is based upon a simple pitch rate program initiated at T+10s which upon completion progresses into a gravity turn trajectory for the remainder of first stage burn. Similarly the attitude control of the second stage utilizes a combination of gravity turning and pitch rate programming to arrive at the orbit insertion point.

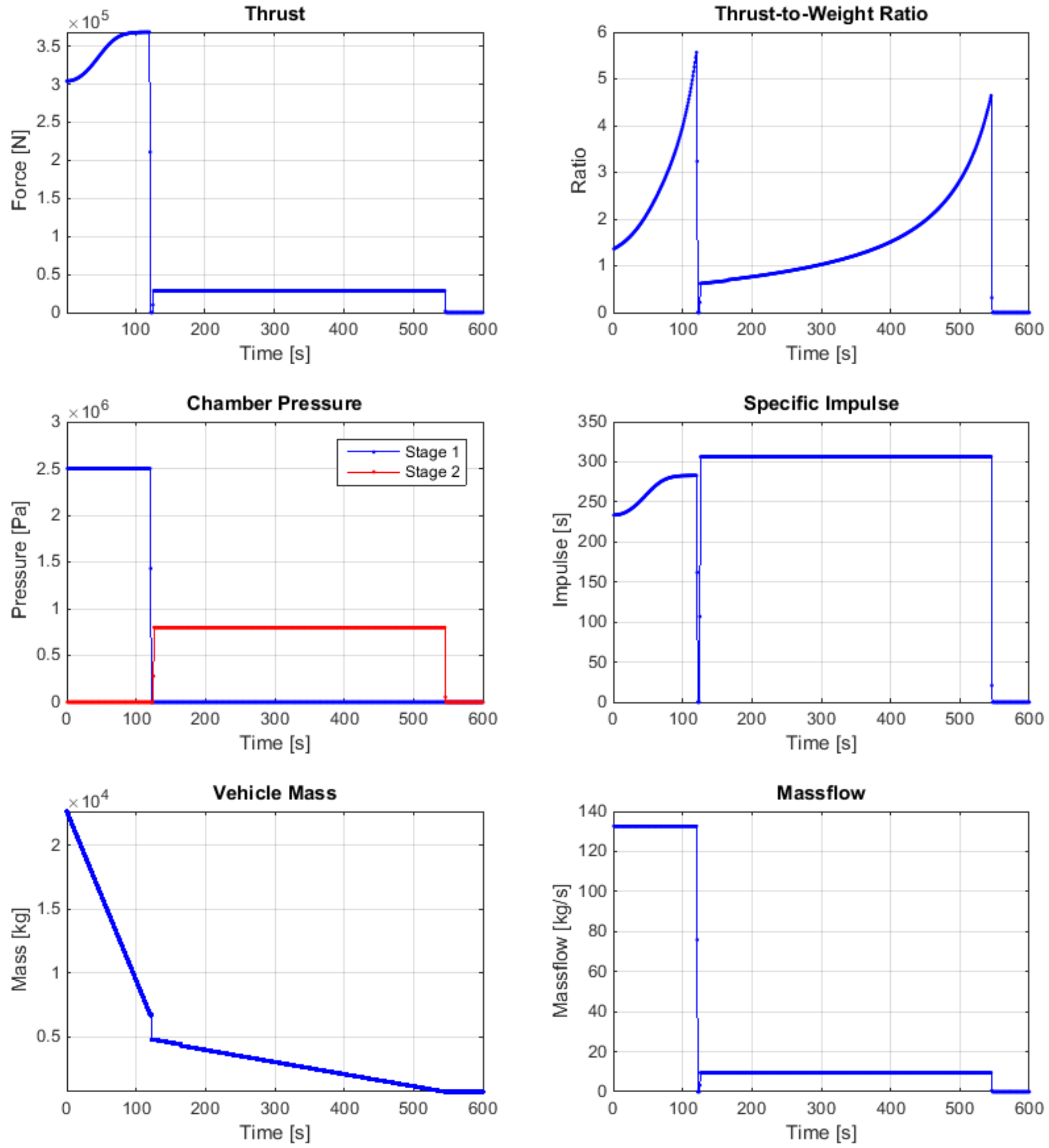


Figure 12 - Evolution of primary launch vehicle parameters for powered phase of low latitude launch.

Figure 12 depicts the evolution of key launch vehicle parameters over the powered phase. Initially the thrust level of the first stage main engine is reduced due to suboptimal expansion, which is also reflected in the specific impulse ascertained for part of the first stage burn. TTWR initiates at the design value of 1.3 increasing to 5.7 over the course of the first stage burn. The second stage burn initiates at T+125 seconds with a TTWR of only 0.62 increasing to 0.7 when fairing separation occurs at T+165 seconds, which effectively serves to limit maximum acceleration levels incurred during the latter part of the second stage burn. The notional ascent trajectory has been designed so as to avoid the need for engine throttling, which in turn results in constant chamber pressures and mass flows throughout the burn.

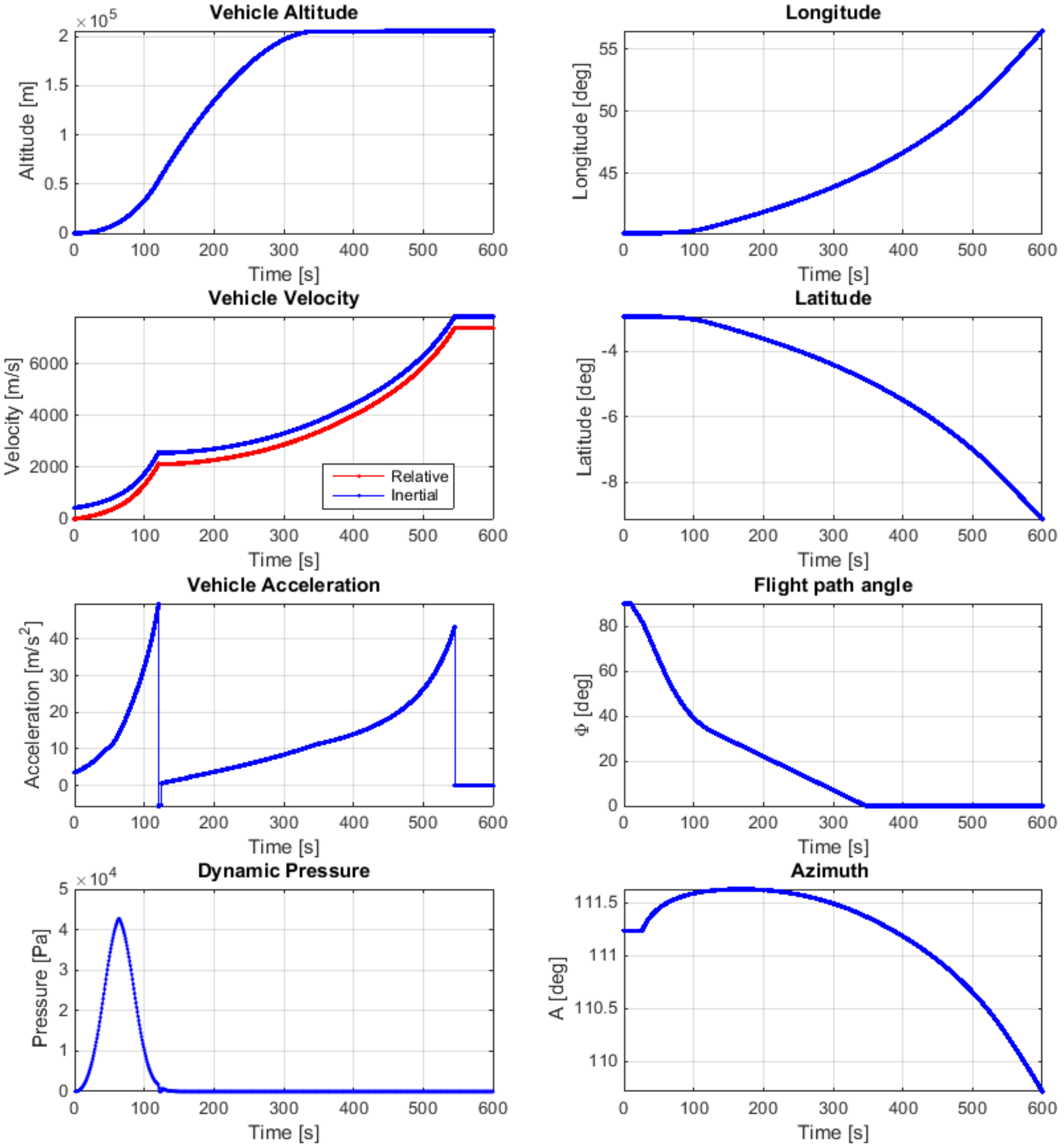


Figure 13 - Evolution of primary trajectory characteristics for powered phase of low latitude launch.

As illustrated above orbit insertion is completed after T+545 seconds at an altitude of 205 km. The orbital parameters ascertained yield a perigee altitude of 205 km, an apogee of 359km and an inclination of 20.5°. Dynamic pressure during ascent peaks at 42.7 kPa in accordance with the optimization criteria. Following stage separation, the spent first stage of the launch vehicle performs a ballistic reentry. Impact will occur approximately 500 km downrange some 6-8 minutes after launch depending on the specific reentry attitude profile and condition of the stage.

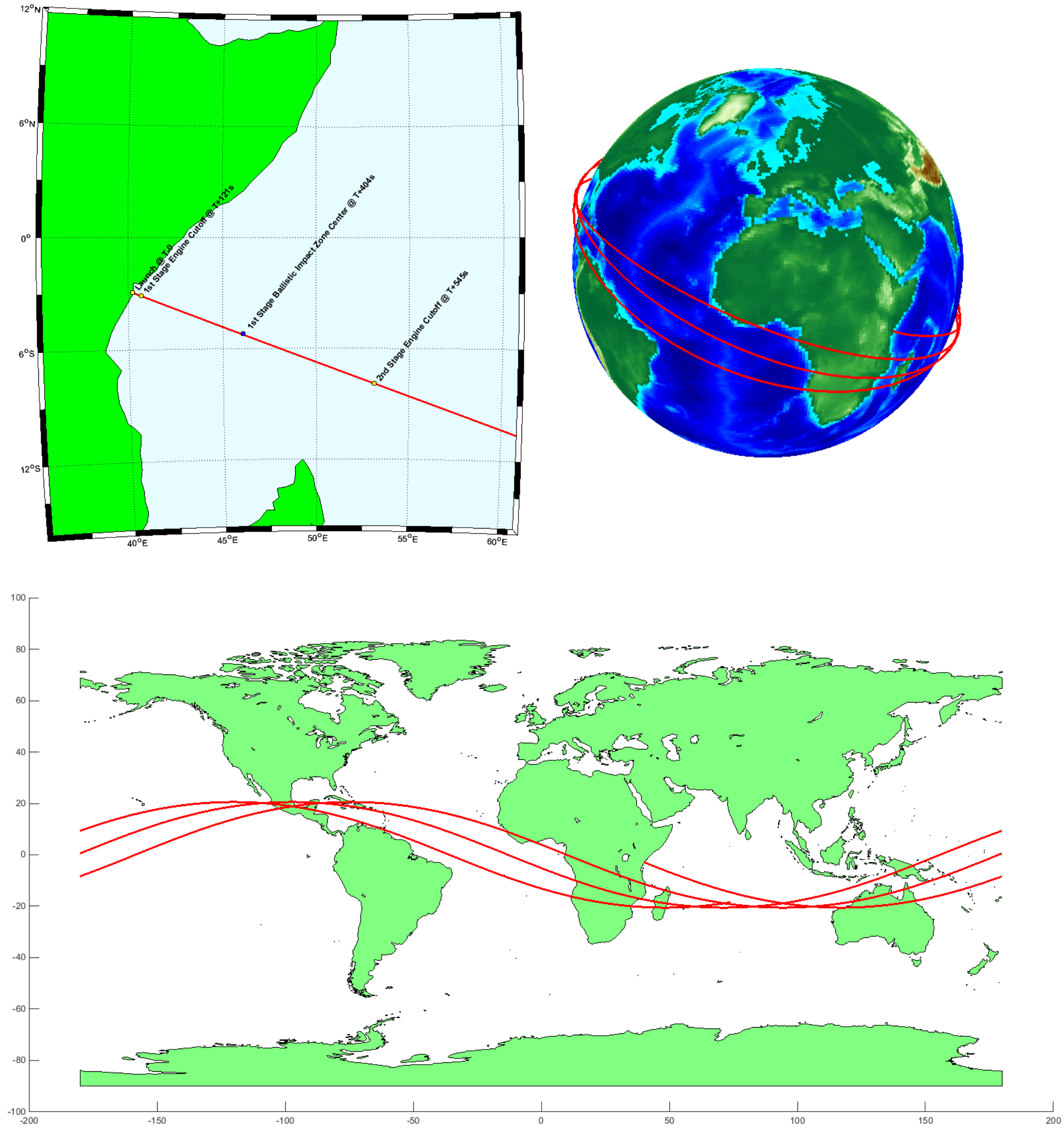


Figure 14 - Ascent and early LEO trajectory projections for low latitude launch.

By extending the simulation to cover 3.5 days so as to encompass the worst case duration of the nominal mission LEO segment at 50 hours, it is observed that a stable orbit with minimal altitude decay is maintained.

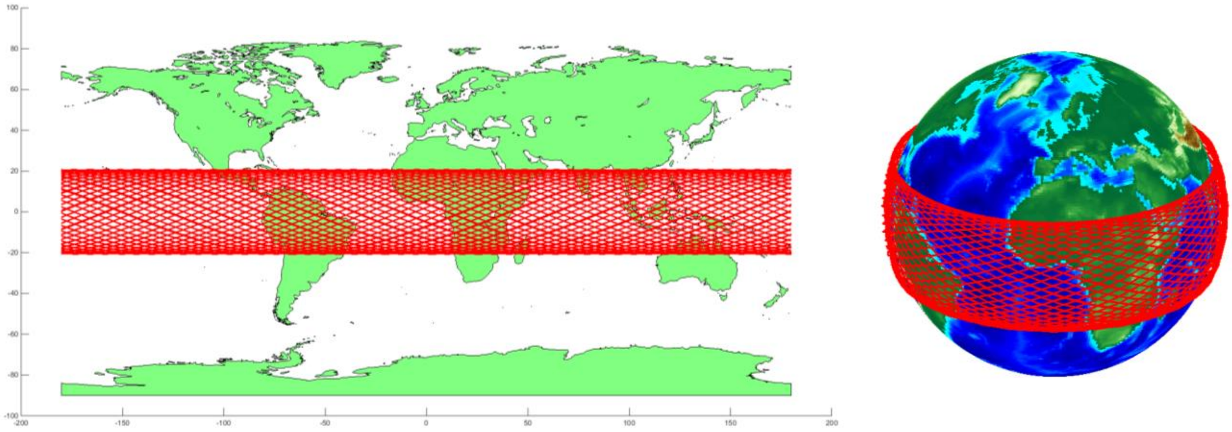


Figure 15 - Extended LEO trajectory projections, 3.5 days or 55 orbits.

Over the course of 55 orbits the orbit apogee is observed to drop 26 km from 359 km to 333 km due to atmospheric interaction (at Solar maximum conditions), which is well within mission requirements. If TLI does not occur, the decay rates observed are consistent with an atmospheric reentry of the spacecraft some 20 days after launch, thus rendering the Moonspike mission concept fully compatible existing international space debris mitigation guidelines.

Time (s)	**Mission step**	**Altitude (km)**	**Velocity (m/s)**
T+0	Lift off	0	0
T+64	Max Q	11	400
T+122	Stage 1 separation	57	2,120
T+125	Stage 2 ignition	62	2,110
T+165	Fairing jettison	102	2,180
T+545	Stage 2 cut-off	205	7,830
T+700	Spacecraft separation	206	7,790

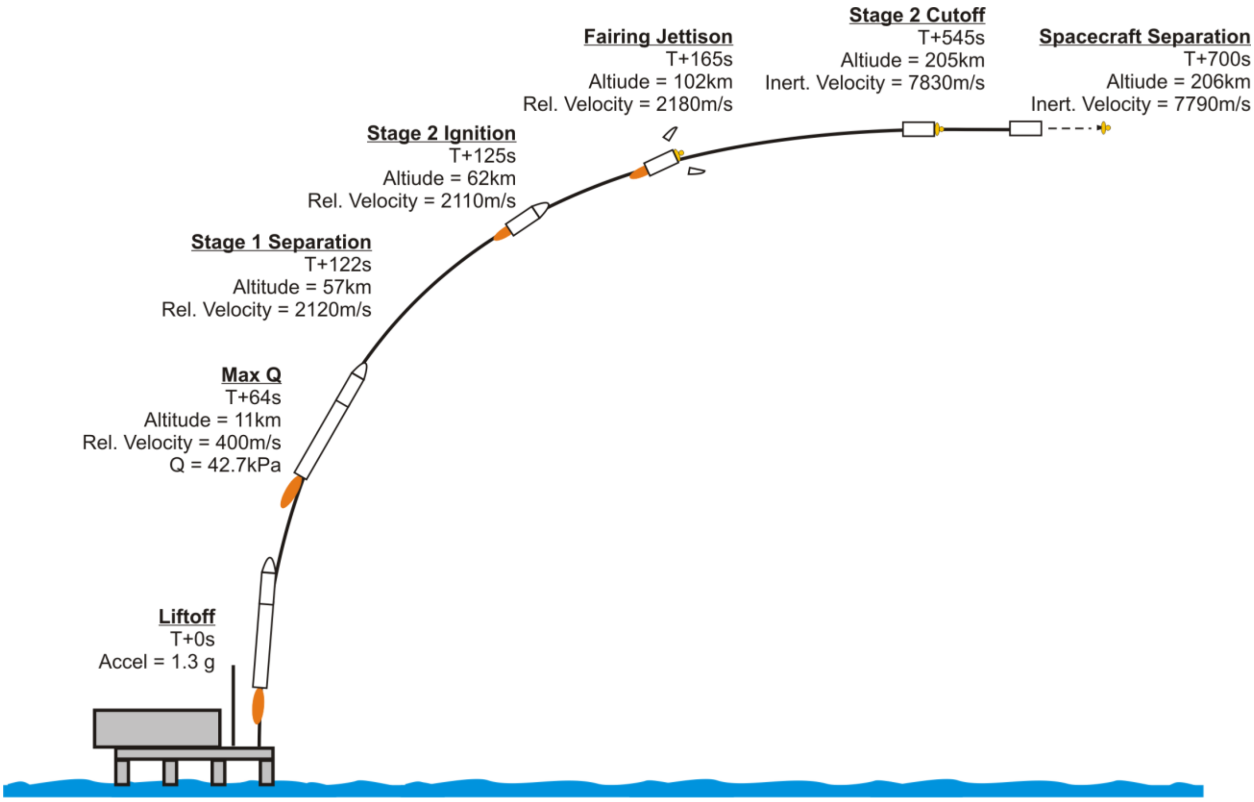


Figure 16 - Low latitude launch ascent profile for notional simulation scenario.

11.3 Case 2: Medium Latitude Launch

The second case considers a notional mid latitude launch from the Mthatha coastal region in South Africa located at -31.5953°S, 29.6075°E. So as to minimize the angular separation between the spacecraft and Lunar orbit planes while optimizing inertial velocity gain from the Earths rotation, a direct prograde launch with an azimuth of 90° is executed in this simulation. Akin to the low latitude case, this azimuth also achieves reasonable range safety conditions for the first stage drop zone.

To compensate the increased latitude, the simulated attitude controller for the launch vehicle first stage employs a slightly more aggressive pitch program to initiate the gravity turn ascent trajectory. Similarly, the second stage attitude control profile is tuned to ensure a flight path angle of 0 degrees and an absolute velocity surplus at the time of burnout, thus placing the orbit perigee at the point of injection.

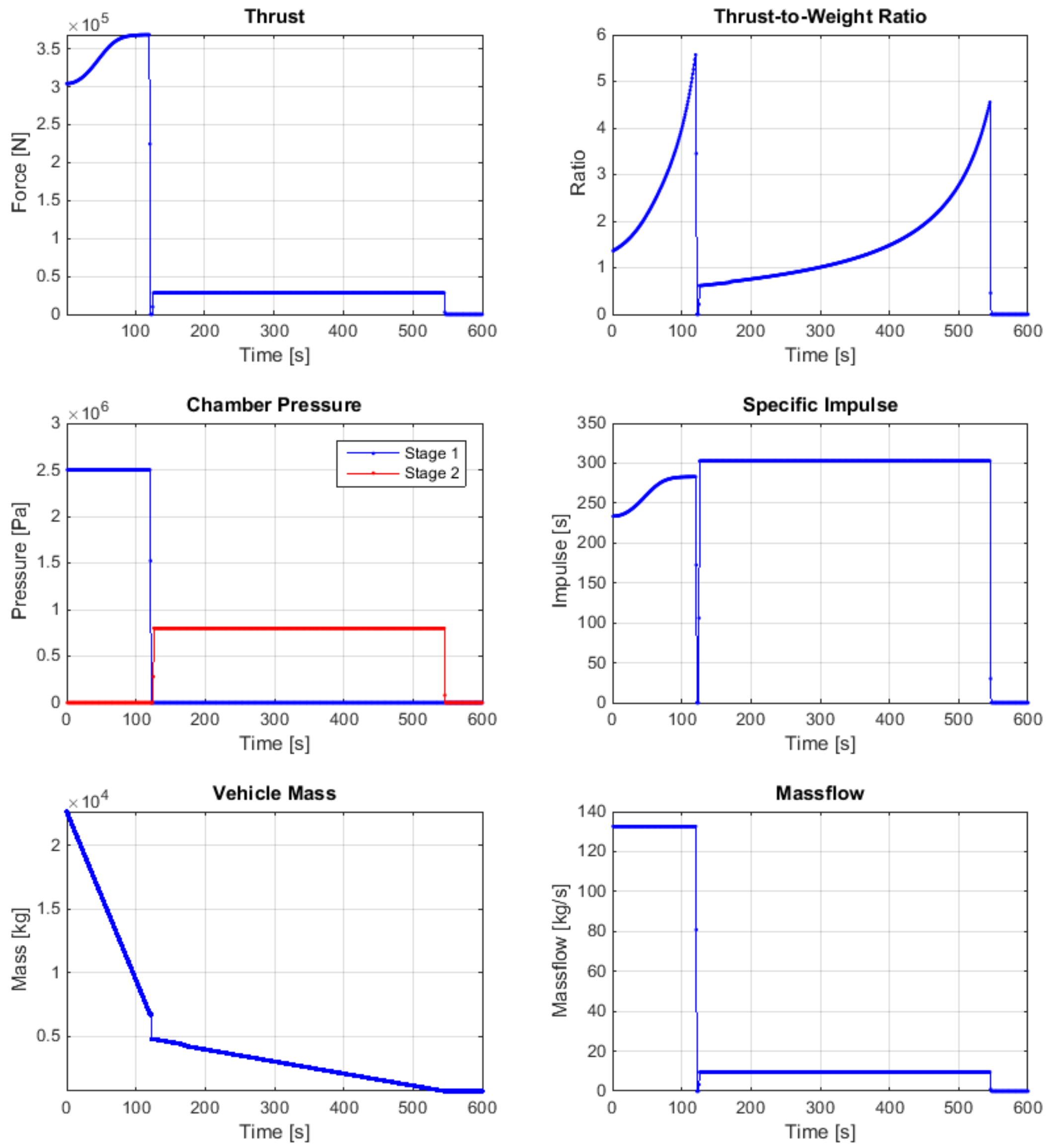


Figure 17 - Evolution of primary launch vehicle parameters for powered phase of mid latitude launch.

As illustrated by Figure 17, launch vehicle functional performance incurs very little change over the low to mid latitude operating range as no engine throttling has been employed in either case. The evolution of trajectory primary characteristics as depicted in Figure 18 show a slight increase in dynamic pressure to 43.3 kPa as compared to the low latitude case due to the changes in the pitch program of the vehicle first stage.

Moreover, the ascent profile maintains a maximum g-loading of approximately 5g. The ascertained orbit characteristics feature a perigee of 174.5km, an apogee of 193.6km with the inclination set at 31.7°, consistent with the reduced inertial velocity component derived from the increased launch latitude.

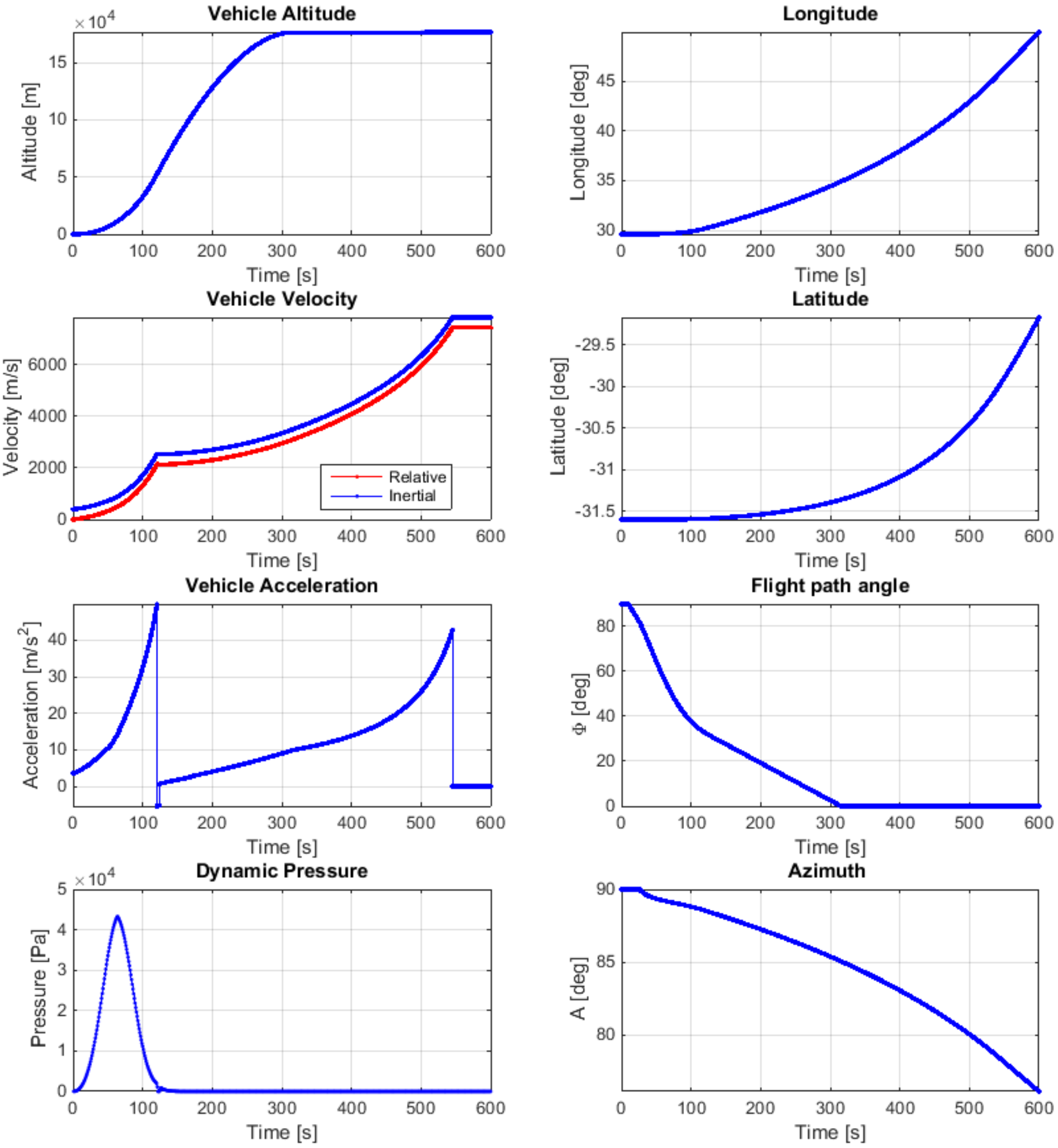


Figure 18 - Evolution of primary trajectory characteristics for powered phase of mid latitude launch.

As for the low latitude case, the ascent and early LEO trajectory projections shown in Figure 19 indicate the first stage drop zone being located approximately 500km downrange of the launch site some 6-8 minutes after launch depending on the specific reentry attitude profile and condition of the stage. The orbit obtained is stable in the short term, yet the low perigee altitude leads to increased atmospheric drag and low lifetime. To explore whether the nominal mission lifetime requirement of 50 hours in LEO can be met an extended simulation has been performed spanning launch+60 hours.

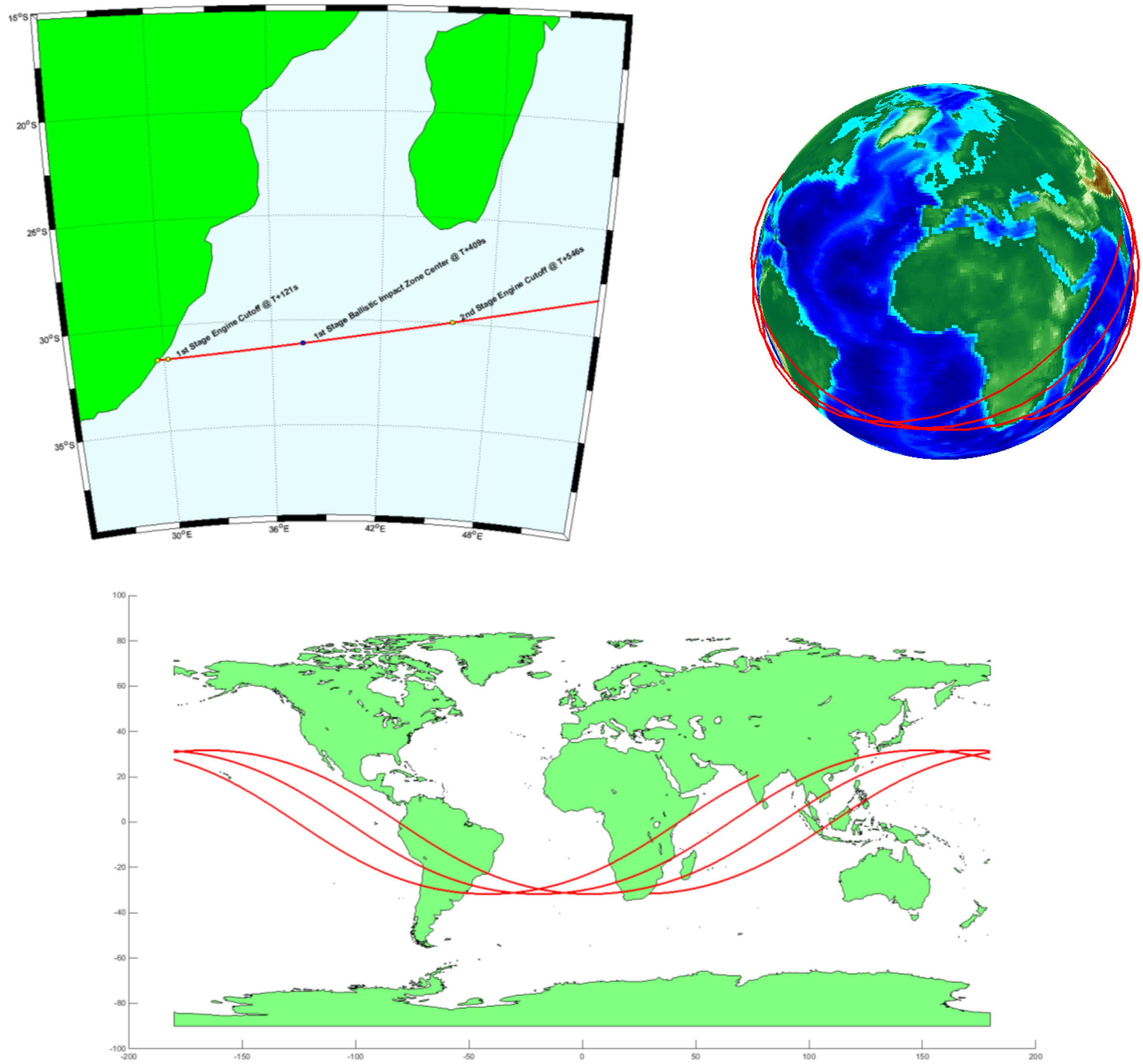


Figure 19 - Ascent and early LEO trajectory projections for mid latitude launch.

Time (s)	**Mission step**	**Altitude (km)**	**Velocity (m/s)**
T+0	Lift off	0	0
T+64	Max Q	11	396
T+121	Stage 1 separation	53	2,170
T+125	Stage 2 ignition	57	2,110
T+169	Fairing jettison	102	2,180
T+546	Stage 2 cut-off	175	7,840
T+700	Spacecraft separation	175	7,810

As Figure 20 shows the spacecraft is observed to reenter after 50 hours, which although consistent with mission requirements, leaves little design margin and imposes more stringent performance requirements upon the spacecraft main engine system. As such, a launch site latitude of +/-30° should be perceived as the outer limits of the current launch vehicle performance envelope for the nominal mission profile.

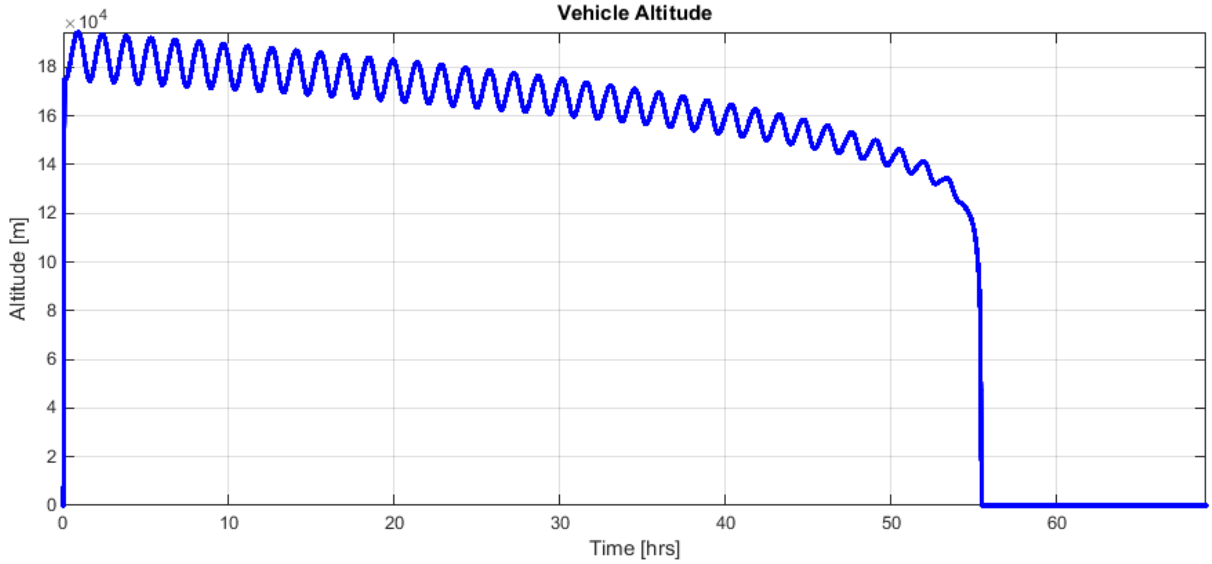


Figure 20 - Orbit altitude evolution for mid latitude launch.

11.4 Envelope Simulation Results and Future Work

The envelope simulations have demonstrated that the baseline launch vehicle configuration can indeed place the spacecraft payload into a low Earth orbit from where the Moon can be reached. Moreover, it has been shown that achieving an orbit with the desired characteristics is possible for launch sites in the +/-30° latitude range, even when a suboptimal attitude control strategy is employed during ascent.

Proceeding towards a PDR, the design margins shall be explored and detailed to a larger extent, specifically to establish firm margins for propellant utilization symmetry and worst case steering losses as the vehicle design matures. Likewise impact zone analysis will be extended to cover first stage aerodynamic properties, and combined with an analysis of FTS commandability windows during the ascent phase to ensure adequate coverage criteria can be met throughout. These analyses will further be used to reduce the list of potential launch sites.

12 Concluding Remarks

This document has presented a preliminary design to meet the mission goal of reaching the Moon carrying a very small payload. The basic design has been created using mass estimates of feasible design options, combined with propellants, engine types and performance showing that the delta-v requirement has been met.

A total of Δv of 13.2 km/s can be summarized using the Δv of all three stages, and furthermore the design and performance of the rocket has shown positive LEO insertion through detailed simulation, which has taken gravity and aerodynamic forces into the calculation.

We have shown that the current design is capable of launching into LEO from latitudes as high as 30° which opens up many future options for launch coordinates.

Design drivers have been presented to show our approach towards the goal. Especially making use of computer aided simulation is key to fast initial testing, and carbon composite tanks for all stages, both fuel and cryogenic oxidizer components, are a significant driver for low total vehicle mass and more importantly low structural coefficient.

High I_{sp} engines and turbo pump designs are a very significant design challenge, yet we believe that the pistonless turbo pumps are an enabling technology, offering the necessary performance without the full complexity of exhaust-driven turbo shafts.

This feasibility study must be seen only as the beginning of a preliminary design process, and all solutions presented are subject to change and improvement. While it would be premature to argue that the presented solution is optimal, it serves as an indication that a feasible solution exists.

13 Report Feedback

Moonspike welcome feedback on technical matters from this report, to technical@moonspike.com. Feedback and questions will be answered and posted on the Moonspike website (http://moonspike.com/resources/report-feedback/), in categories, with your name. We may combine general questions, from different people, into one answer. Please allow for some time to handle, combine and answer questions as we receive them.
14 Contact info

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