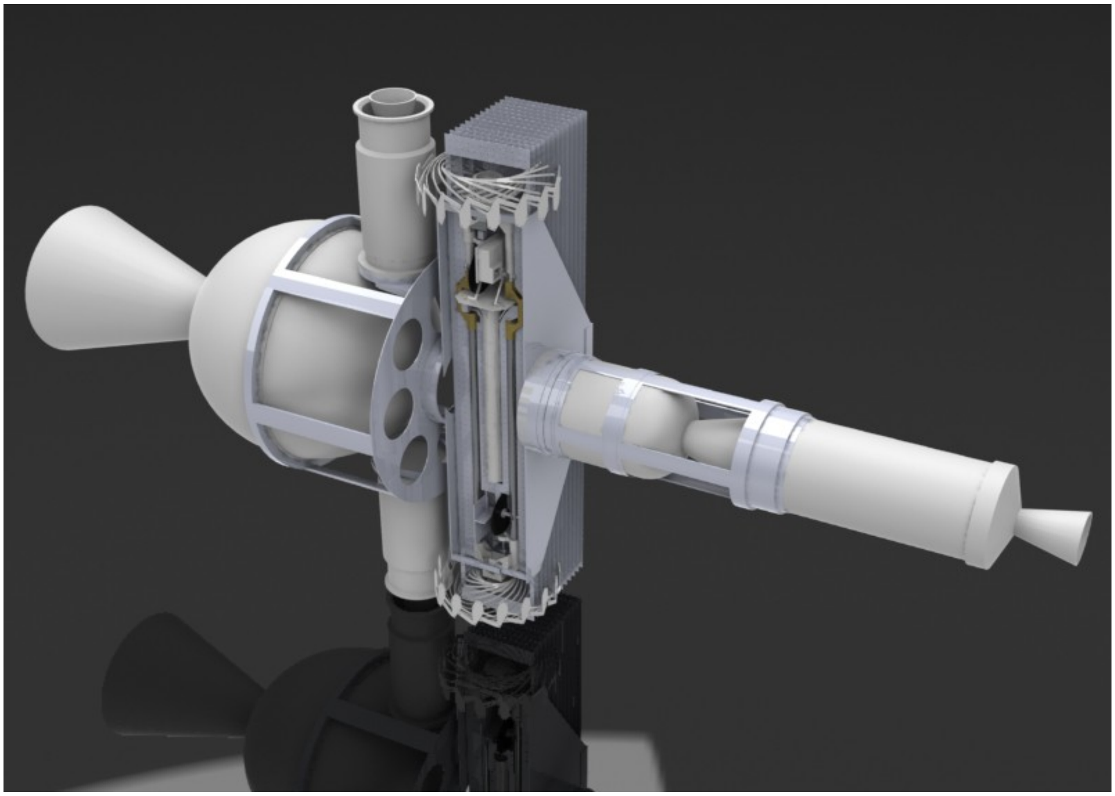
Team PlanB, Landing subsystem. Development and Verification plan.



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October 2013.

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Acknowledgments

The Team PlanB wishes to thank the support of all volunteers for their input in design of the mission, rover, and critical components; especially in a "plan B" scenario, because "plan A" is considered as "plan for Dreams" of the way to the moon. The Team Plan B would also like to acknowledge that technical information from Boris Chertok's books was extremely valuable in designing the blueprint for the mission.

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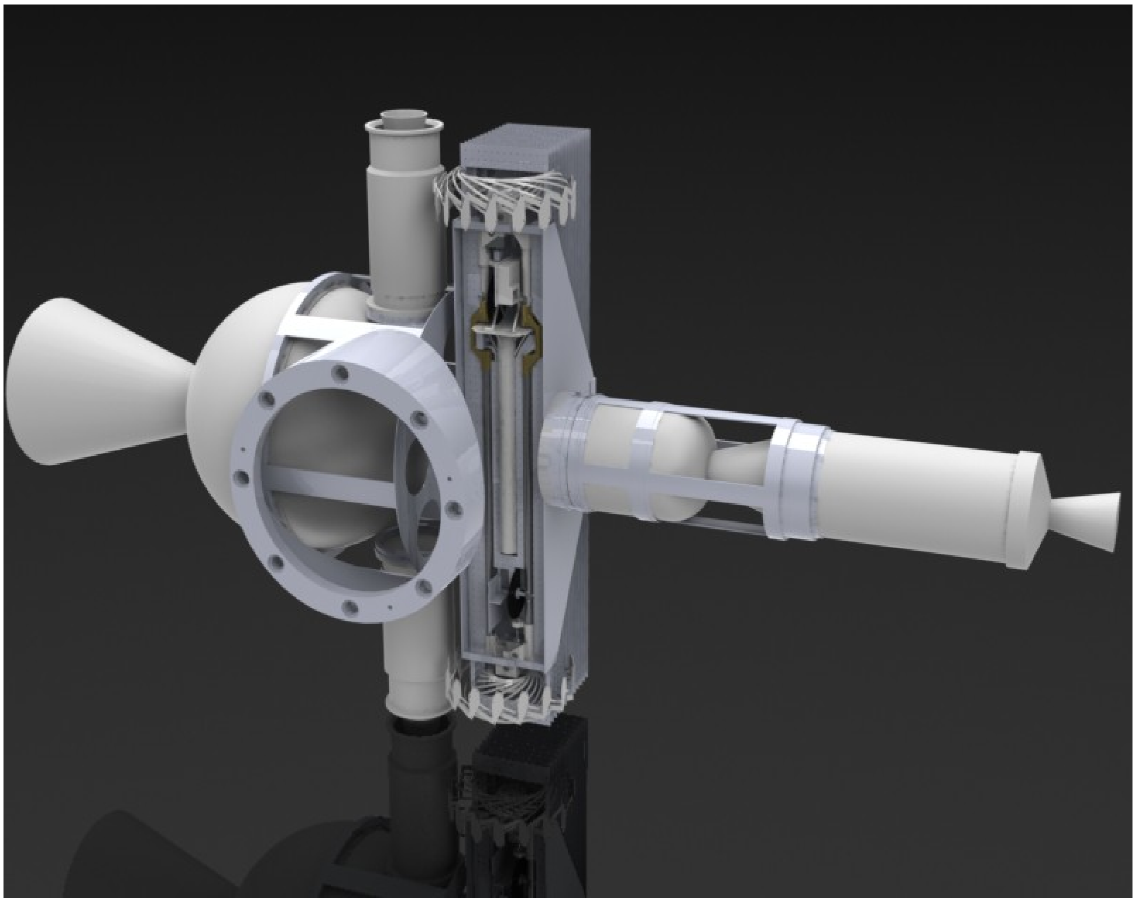
1.0Executive Summary

Craft with payload adapter:



2.0The landing system

The landing system includes hardware and software that support a soft-landing craft on the Moon. Within, there are explanations of any critical technical risks or challenges that the team has already overcome.

2.1 Attitude control en route to the Moon

Challenges in development of a gyro-platform: Available information about the design of gyro-platforms in 2011 was sketchy. They assumed to use gyro-sensors together with accelerometer and magnetometer in mobility platform and automated aerial vehicles. This

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approach allowed having low precision data from solid state gyro-sensors combined with correction data from the accelerometer and magnetometer. Consequently, the correction of "knowledge" about the desired direction will be precise enough to have a practical system implementation. This process of cross-correction of the data between accelerometer, gyro, and magnetometer in mathematical studies, is named the Kalman filter. It is a manner of cross-correction that exists with some predictability in a system's behavior. On the earth, or on the moon's surface, the accelerometer measures gravity and it points to the center of the celestial body. Despite that, the accelerometer sensor by itself has low resolution sequential measurements can give precise direction to the center of the earth. The same story with earth magnetic field, it has some stability (in case of stable position of the mobility platform) averaging, which allows finding the direction vector of the earth's magnetic field by magnetometer with desired precision. A Gyro-sensor from another point, measures the speed of the rotation. To get a direction, it needs to "integrate" speed measurements in each small moment of time. Any error in the sensor's data, along with any delays and missing data intervals, can add an error to the direction measured (result of "integral" calculation or "integrated values"). That error has a name -"zero drift". To correct "integrated values" of the direction, "zero drift" is measured using cross-data provided by the other two sensors. For the earth, this is a suitable approach, and is most likely "okay" on the moon (except for the fact that on the moon, the magnetic field is weak). The magnetometer approach does not work on orbit, it is useful only in the case when the earth's magnetic field is "known" and does not fluctuate much. To solve the challenge and to find practical solution with gyro sensor usable on orbit was done research to extract as much as possible from the gyro-sensor sensitivity. The sensor's operational modes were also investigated. Having the on-board temperature compensation, resulted in a big source of errors. Also, methods were discovered to improve sensitivity by 8 times. It proved practical to use two (and any even amount) sensors to compensate different polarity of independent sensors' axis. Quaternion mathematics was used to perform "integration" of a rotational speed. It does require a large amount of computer power, to perform mathematical calculations. To improve precision and simultaneously reduce the amount of calculation, the "Bradis tables" technique was used, with results of calculations retrieved from the tables. "Zero drift" calculations based on quaternion mathematics principles. But, to properly count for "zero drift" a technique was used to keep "zero drift" as a function, instead of a one quaternion value. "Zero drift" function reflects a periodic. It was unknown whether the dependency of that periodic was from (a) specific characteristics of the particular device, and/or from (b) particular type of solid state gyro-sensors in common. It was found that the "zero drift" function does not depend on a temperature stabilization technique (in that technique, time is allowed for temperature stabilization, before it

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calculates "zero drift"). The Arithmetic's operations were also optimized for the target micro-controller. The resulting gyro-platform module was capable of detecting the earth's rotation. Measured "zero drift" was around 1 degree in a 1 hour test run, and in another 6 hour test run, it was around 4 degrees. The Earth's magnetic field simulation was investigated to include trajectory simulation software to allow for working with the magnetometer.

The challenge in working with "rotation wheels"/"reaction wheels" was to make sure that stepper motors will be performed in a vacuum and under temperature conditions. To satisfy the requirements, the bearings of the stepper motor were replaced. Furthermore, upgrades were made to the wiring to allow for temperature resistant plastic. Device manufacturing was considered to be straight-forward without any challenges. The stepper motors used on the nano-satellite were chosen to include smaller sized motors without ball-bearings. To solve functionality challenges of the nano-satellite, it was assumed that the 3 stepper motors are to be sealed inside the carbon fiber box together with an HD camera and a backup communication modem.

In the nano-satellite test flight was a challenge to design different scenarios, including the conformance of all techniques for orientation and orbit determination into one combined test. The solution discovered, was that the antenna on the nano-satellite must be fixed and directional. Plus, short communication sessions with the ground station, has to be performed with active use of attitude control only. In the testing scenario, the nano-satellite test flight will attempt to control orientation with 2 "rotation wheels" instead of 3, and the testing mode will be set from mission control. That overall experiment with the 2 rotation wheel configuration will be no different from that of a 3 wheel style. Algorithms of attitude control are also designed to be adaptive. Each possible single step action, performed by all possible combinations with the stepper motors will be recorded by a gyro-platform. The possible combination for 2 stepper motors configuration is 25, and for the 3 "rotation wheels" it is 125. That list is the parameters for all attitude control algorithms and is not dependent on the size.

Another challenge for the nano-satellite test flight was also considered. It will not be possible to place the magnetometer far from the stepper motors. The chosen location for the magnetometer was on the tip of the antenna reflector, as far as possible from a magnetic field source created by the stepper motor. It will be used to calibrate the process to measure real earth magnetic field on rotation stand, with the nano-satellite mounted and with the antenna deployed.

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It was a challenge to design the helical antenna on the nano-satellite. The size limitation of a 1 CubeSat unit, does not allow for the space of a helical antenna. It was decided to use a deployable antenna for nano-satellite test flight.

2.2 Systems for tracking and orbit determination en route to the Moon

The challenge to receive a raw global navigation system was solved, by finding a GPS device capable to output time stamp, global satellite navigation system position and velocity in SIRF protocol's ID29/30 record. In the public domain, different software is available which is capable of calculating positions based on raw signals. These do include loggers with a sampling rate of 40MHz.

Calculating position is a challenge solved, by adding to the trajectory calculation software the possibility to simultaneously simulate global navigation satellites. In each moment of the simulation, the orbit distance from the satellite is to be determined and satellites from the global navigation systems. The orbit point where navigational data from the GPS satellite is received will be converted into a local time stamp and the position/velocity of a GPS satellite is used in a "raw" data simulation. To precisely find the position, internal calculation's time interval reduces to 33ns, which allows 10m accuracy in the simulation. 4-5 such "simulated raw" records (ID 30), can now feed into the orbit determination part of trajectory calculations. Initially, estimated orbit parameters in 3 punch cards format(TLE) or in velocity + position format, will be entered as a starting point of calculations. Trajectory simulation (now running in distributed orbit determination mode) starts at the beginning of measured time frame, and calculates first position/velocity with the attempt to match first ID30 record. The error in distance (time) is used in the iteration process of adjustments for initial (starting) velocity + position/ TLE values. Half division is used as a mathematical method, to find match in first ID30 recorded point. After the first match is performed, the second match for a second ID30 is measured with continuance to match all consecutive ID30 measurements. For both processes (generation of a raw global navigational data records ID30, and orbit determination) the same software was used. This chosen method is more computer power hungry, but it allows the research of analytical formulas to be skipped.

Another challenge was the conformation of the functionality of a GPS receiver device with raw data output, without flying on the orbit. A one-channel GPS hardware signal generator test device was selected, with the direct feed of data from the software simulation to the hardware signal generator. It is not critical to generate many GPS signals for orbit determination, because the raw data for different GPS satellites will be generated sequentially and time stamps can be artificially adjusted.

Another challenge was to find a good method of obtaining raw global navigation system in a different way, than using a standard manufactured receiver. Modern RF GPS/Galileo RF front end devices were one of the choices. In 2013, it was considered to use one of the available devices which satisfied temperature operation conditions and functionality to receive and digitize data from two systems. But the device went out of production, and the substitution device was less capable than original. For both "prime/standard" and "backup/raw" GPS/Galileo L1 signal receiver will be used the hardware GPS signal generator. Compatibility for calculations from both devices will be provided by converting data from second ('backup") device to ID30 record. Mission control orbit’s determination software will deal only with ID30 records.

The same challenge with the orbit determination by raw GPR/Galileo signal was presented when determining orbit by measurements of RF signal travel time from the ground station to craft/nano-satellite and back. That feature is included in the communication subsystem. It uses "loop" packets, initiated by mission control, which sends to a ground station, through to the satellite and back to ground station to arrive at mission control to close the “loop”. “Loop” packets are stored in a database for separate processing by the distributed orbit determination system. First, to investigate implementing a simulation of all processes in the trajectory calculation software, distances from ground station locations to the satellite was calculated. That distance requires defining time, and then it needs to simulate a delay for the processing of data on the orbit and time of the transfer “loop” signal back from the defined second point on the orbit. From the second point on the orbit, the second distance was calculated and that distance defines the second measured time. The summary of the first and second time loads into the first "loop" record. This record is stored in the DB and a delay in data processing onboard loads into the second "loop" record. After this, both "loop" records are ready to be processed by the orbit determination distributed software. By running on separate computer software, "loop" records will be retrieved from the DB and will perform an orbit "match" by the same iteration algorithms as for the raw GPS data determination. Initial (TLE / velocity + position) values are attempted to match data from two sequential "loop" records. Sequential matches for different "loop" records can be used to match orbit with desired precision. Distributed calculations implemented in trajectory

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calculation software allows for use of different computers to try different orbits matches to reduce overall time.

All orbit determination / trajectory calculation software relies on a ground station position. The problem was to decide which methods are the best to determine real geostationary coordinates of a ground station. The decision was made to use raw signals for the ground station's position determination. Mission control sends requests to ground station (hardware is equal to a nano-satellite/craft recording sequence of ID30 records). The distributed orbit determination software can also perform independent tasks to determine the stationary position of the ground station (not the orbit). Originally, methods were considered using the original GPS position determination from the standard GPS device. This method will be available and can be used to cross verify first method.

Determining the trans-lunar trajectory after the main impulse burn is a challenge that was solved by the decision to record constantly, when the craft will be in range of the GPS/Galileo satellites. ID30 records captured in mission control will be the last source of GPS methods of orbit determination on trans-lunar orbit.

Determining orbit after the fourth correction burn on trans-lunar trajectory (sending craft on collision course with moon) was decided to be solved in two different ways. The first way is to use a communication subsystem to determine distances. The second method is to actively use calculated directions to the sun and the direction to the center of the nearest celestial body. The time stamp of the detected sun direction with directions to earth/moon centers, together with quaternions which describe rotation to those directions will be retrieved from a craft by mission control. To solve the challenge of the second manner it was decided to implement additional calculation functionality in the trajectory calculations software. With the path to the moon simulation, directions from satellite to the sun, earth, and moon were calculated. That record can then be inserted into the DB to simulate real observations on the trans-lunar orbit. Inserted records will be a source of data for the distributed orbit determination software running independently.

2.3 Guidance Navigation and Control (GNC) for the lunar descent (including sensors)

Challenges in the development of a laser range finder was solved, with design hardware and software capable of measuring distances 5-25km. Precision for the ignition command of a brake fixed impulse engine was assumed at 0.001s. First, digital to analog and analog to

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digital convertors were chosen that are capable to work under required temperature operation conditions with a sampling rate at least of 33ns. That sampling rate allows for measuring travel time of the reflected signal from the lunar surface with 10m accuracy. Double paths for laser beam travel allows to measure distances with a precision of 5m. A special sequence sends out to a laser TX module to be emitted over an optical system. This sequence is converted to an analog voltage level signal by DAC. The reflected signal accepts by matching the wavelength receiver, and converting by 8bit analog to a digital convertor value which is stored into a Magnetoresistive RAM. The write operation is synchronized by providing 33MHz impulses to 3 digital counters. That allows addressing MRAM for a light travel time up to 50 km (5Kbytes). A comparison routine scans 5K, to find a match of the same pattern which is sent over the laser beam. To be successful, the match requires performance of 160,000 processor's instructions, with a processor speed 40mln operations / sec. It will be allowed to have 250 cycles of measurements/sec. After the pattern is found and confirmed during sequential measurements, the next second of the flight will be measured with speed precision of 10m/s. That value will be compared to a calculation by mission control and the difference of the speed measured and calculated, will initiate recalculation of the ignition moment. One second before ignition, at a distance of around 8-15 km (2K data to be processed), it will switch to a short scan mode. In this case, the attempt to match measured data transmitted, will be reduced to the "expected" area of comparison (expected 500 byte) and that will increase the amount of measurement cycles to 1000 per/sec. The expectancy delay in the ignition of the engine will be added to predict a last measurement cycle before ignition. After the last measured cycle was calculated, a fraction of mks is to be used to send the ignition command. The command will be sent to ignite the engine and to separate the laser range finder with the attitude control system. That separated chunk ("face-off package") will crash to the lunar surface, but a landing subsystem will perform the brake impulse burn.

Another challenge for the range finder was to use a "test" system where all of these steps would be debugged. A practical way is with the use of sonar equipment to simulate the all of discovered steps. A measuring device was placed in the tank over the surface of water. With the waves simulated on the wrong axis rotation of the craft, different distances were simulated by pumping water in and out from the tank. The averaging/noise reduction pattern search algorithms were debugged on a simulation device.

2.4 Avionics

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A challenge was solved in developing in-house proprietary RF communication equipment. Design and manufacturing steps were analyzed with the time for manufacturing measured. Communication will be over unlicensed frequencies bands at 2.4 GHz (no limitation on antennas design, different limitations on max power in different countries). Operational steps and technologies were measured to be communication subsystem flight certified. Issues were solved in matching the impedance of transmitter, receiver and amplifiers. In 2012, a selection was done out of the available power amplifiers and LNA. Two versions of the communication system were developed. The second version used an "AT" modem implementation at one layer of a data transfer protocol. The challenge with the complexity of a supporting error correction on the top layer (up from modem protocol) was solved by abandoning the use of an "AT" type modem. In development, there is now a third version of the software. The source code was ported to all possible micro-controllers. This supports having an extended hardware implementation for the ground station with a capability to log all that is received from the craft/nano-satellite communication packets to a mission control database. Challenges in the integration with ground station and mission control were solved. These included (a) transfer of encrypted data to a mission control server, (b) design, development, and verification standalone HTTP server as a ground station communication hub, (c) synchronization time between Mission control server, ground station communication hub, communication micro-controller and (d) logistics for range tests for 1.7km, 4km, 25km with 0dBm transmitting power.

A solution was found for the deployment of antenna to have a gain of no less than 16 dBm. This was solved by prototyping 4 antennas: (a) helix with 16dBm (b) deployable for nano-satellite 6dBm (c) low profile reduced polarized helix >18 dBm (d) low profile deployable reduced polarized helix > 12dBm. Certificates regarding the techniques for manufacturing flight nano-satellite and rover antenna were verified. Manufacturing techniques for the ground station antennas were also verified. The challenge around the ground station manufacturing antenna was in the low precision of 3D printed parts. Long antennas were not stiff enough. As a result, it required a special 3D design to support the structure. A polarized version of the helix has to be precise with 0.1mm, otherwise all advantages of a polarization (transmitted energy will be polarized and gain will be improved) are not solved yet.

The delivery of data over noisy environments, or over long distances with low transmitted power, is a constant area for solutions and improvements. Doubling and tripling the communication channels, is a usual choice. But with any increase of the transmitting channels, the data restoration numbers become challenging due to possible de-

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synchronizations. Delivered packets can be perfect, but timing of the events can ruin the tripled sensors reading with full flight lost. Sequential tripling was implemented, with each packet transmitted over the first, second, and third communication channels, one after each other. Preamble was increased and the address field was equal preamble. Each packet arrived to ground station, with mandatory storage in the mission control DB for a future possible restoration. Restoration of packets with broken CRC was done by complex measures including shifting data and majority voting 2-from-3. Precision of the channel's switching was done by stabilizing crystal with temperature operational condition of -40+125C, and software algorithms with adjusting switching time. The frequency drift had to be compensated because of different temperature operational conditions on the craft/nano-satellite and on the ground station. Constant adjustment processes were implemented during the communication session and the initial session's adjustment process, when the ground station tried different channels with craft/nano-satellite listening on one frequency.

Challenges in transferring data from/to different layers of communication protocols were found to be a critical process. On the communication device, exchange algorithms were implemented for data synchronization stored externally for micro-controller FLASH memory.

The challenge with precisely measuring distances from the ground station to craft/nano-satellite was solved by selecting stable crystal for the micro-controller, and routine to load "loop" message with measured data.

The challenge to deliver an "extreme" power burst from the ground station to the craft on trans-lunar trajectory was solved, by design swappable technology for ground station antenna. Instead of 1 antenna on the ground station, for the main mission four can be mounted. The simplification of a winding 3D printed antenna, allows updating amplifiers for the helix in 1 hour and for polarized helix for 4 hours (on a ground station).

The challenge to have backup communication on LEO was solved by incorporating an ORBITCOM communication modem into a communication system. Previously, the modem was successfully used for satellite to satellite communications.

The challenge of selecting a solar sensor (IR sensor) was overcome, by obtaining different sensors in 2011 and conducting testing with the connection to the micro-controller over two switches and two sensors. The switches were allowed to choose which IR sensor was used as a voltage reference and which one is measured against another. Maximum value read

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after the ADC conversion in both configurations allowed for the detection of the moment when IR radiation would be equal for both sensors.

Having separate methods for directional determination to the center of the nearest celestial body was solved by developing an imaging system for a low resolution camera in 2006. It included hardware that incorporated an IR capable low resolution imaging sensor and software algorithms which processes the image with compression simultaneously. The challenge continues to remain, because of the production status of the imaging sensor and an absence for substitute sensors with suitable parameters for operational conditions and sensitivity.

The challenge to protect data over RF and IP was solved with proprietary encryption software, which includes strength better than existing industry standards.

2.5 Propulsion

Calculating required impulses for all fixed engines was solved by embedding semi-automatic optimization steps into trajectory calculation software in 2011. The challenge to simplify the process of matching fixed impulses engines was solved by "weight matching" (propellant). In trajectory and impulse calculations, the profile and parameters of an amateurs fixed impulse engine was relied upon. This engine was developed as a hobby, by Canadian group of enthusiasts. Parameters of that engine was 2 times less in impulse performed, as compared to a similar propellant weight fixed impulse engine produced by industry leader ATK. It is assumed, that if hobbyists and enthusiasts would be able to manufacture something which can be used to reach the moon in contrast to obtaining an industry grade engine, it will make the flight less restrictive in finding landing site and time of landing.

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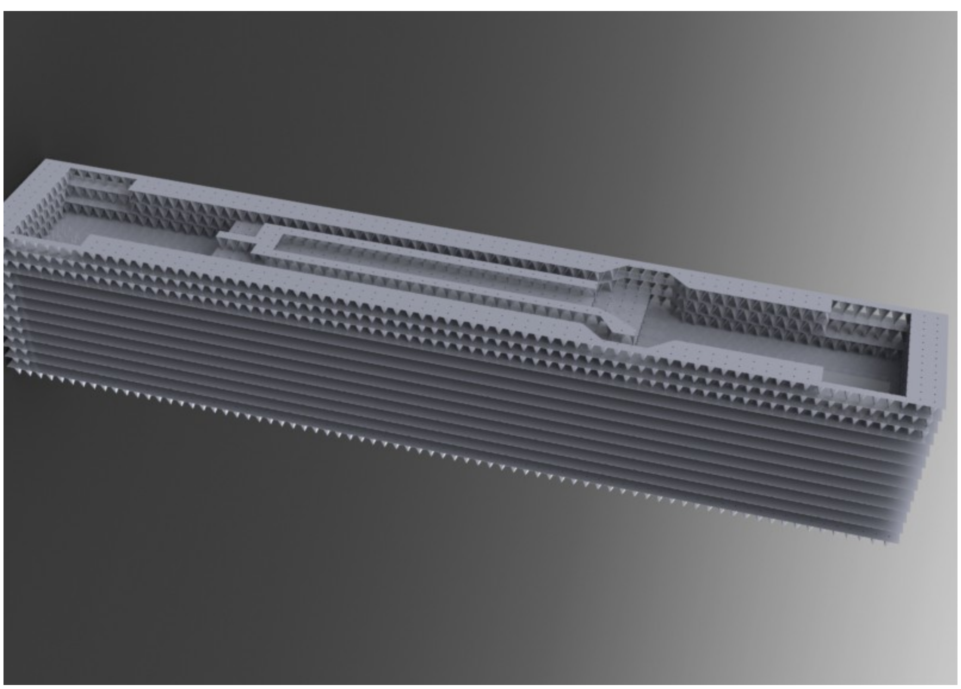
2.6 Touchdown devices

The challenge in designing an impact shield was solved by investigating technological processes of manufacturing carbon fiber layers with different property epoxy, to be applied into the same manufacturing part. Adding different grade graphene to the same epoxy makes the epoxy less or more structurally strong. The impact shield's layer is required to have a different property to convert energy of the impact into thermal energy.

Challenges in manufacturing a spring for the compensation of rotation after the separation of a brake engine, was solved by selecting carbon fiber with a special epoxy (-75+300C temperature operational conditions). This also included methods involving knotted carbon fiber inserts. The property of the manufactured part depends on the knotting technique and reinforcement threads inserted between the knots. The stiffness of the spring cannot be adjusted, but the speed of rotation of a craft before the brake burn can be precise. The temperature of a spring needs to be a measured to achieve final adjustment.

Pic 2.6.1 Impact shield assembly.



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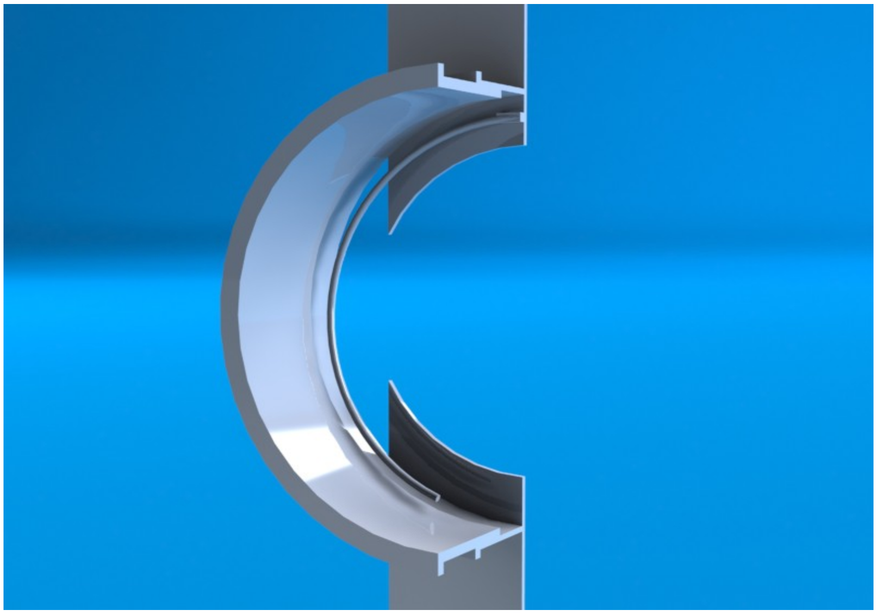
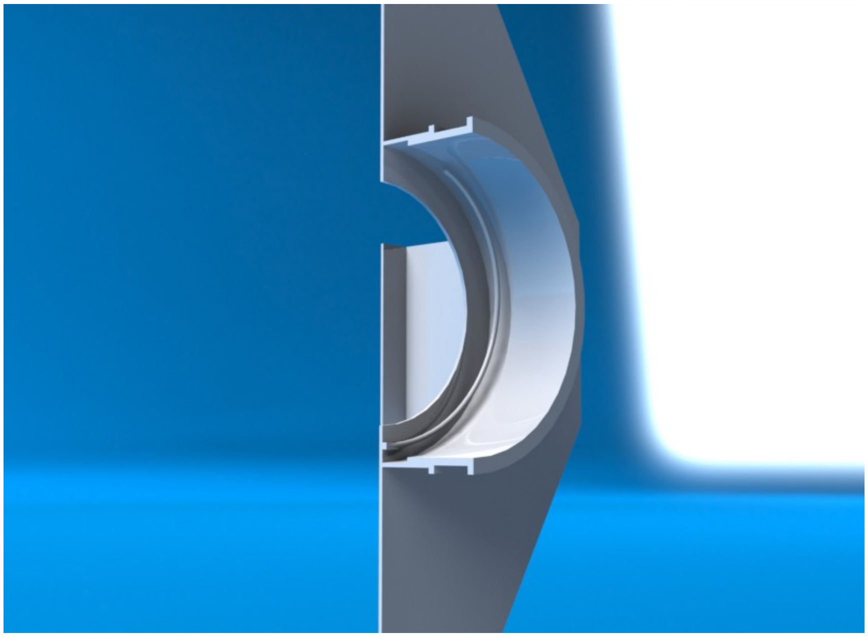
Pic 2.6.2 separation mechanism with spring to reduce rotation (left part).



Pic 6.2.3 separation mechanism with spring (right part)



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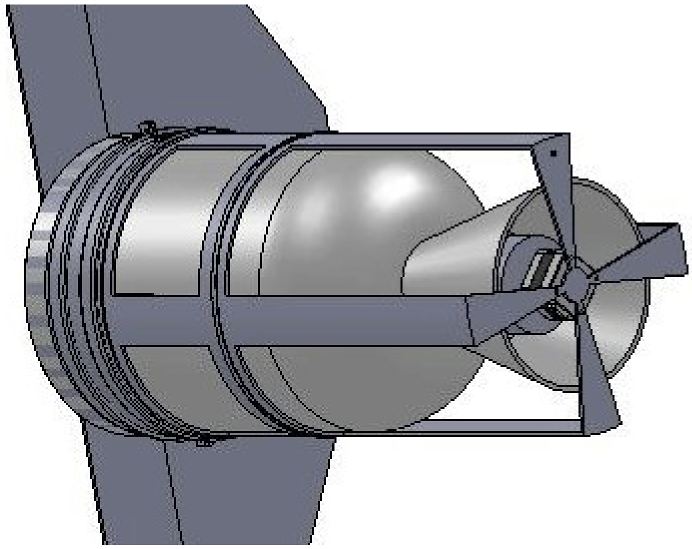
pic 6.2.4 Face-off device (attitude control / hokey puck ejected with laser range system)



2.7 Thermal control of subsystem a craft

Challenges for temperature measuring technique were overcome by software design, implementation, and debug. Software work with DS1822 sensor with accuracy of temperature measurement +-2C in -55C +150C interval. Sensor has special interface to communicate with multiple device over same power and data line. Numerous micro-controllers can do all measurements. Special measure was applied to all software development to optimize number instructions executed by micro-controllers. Each instruction executed by micro-controller mean additional power require by power plant. Challenge to work with different architecture of micro-controllers was overcome by producing source code portable to different platform and compilers.

Challenge for thermal control study for HD camera box and low resolution camera's box was done by experimenting on extrudes system of 3D printers. Different technique was tested and adapted to design of software (for supporting active camera's thermal control). For "on moon" mode of a thermal control was chosen additional temperature sensor located on the tip of a camera box. That additional sensor solves challenge for thermal control with use of lunar regolith as a source for heat or cooling.

Challenge active thermal control in "on orbit" mode of operation was solved by embedding calculations and storing derivatives of a temperature's measurements, done in different orientation position of the craft. For such software implementations was considered testing scenarios with heat source and cooling applied to prototype of a nano-satellite.

Challenge in passive thermal control for a test mission will be resolved by applying 6 Peltier elements to each side of nano-sattelite. Connecting pairs from opposite sides allow to transfer heat from heated to a heat radiated site.

2.8 Onboard autonomy

Challenges in onboard autonomy was solved by apply as much as possible software instead of hardware solutions. Software is main and strongest area of experience in our team.

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2.8 Challenges in interfaces to other subsystems.

Challenges are constant in software development. Software was considered as a main tool that can help to solve hardware challenges and to eliminate challenges in mission itself. A lot was solved and still are waiting to be solved on the way to the moon. First, was decided to do not follow any standard methodology in software development, as all methodology designed to increase costs of development instead of concentrate on the goal. Second, was abandoned any standard protocols and API interfaces for a communications, instead was chosen serial and I2C protocols, and "state machine" for instructions/commands processing. For real time processing the date was considered technique with interrupt hardware support in micro-controllers, pipes support for data, and call backs functions provide interface. Challenge for transferring data from imagining subsystem to a mission control was overcome by rapid designing, open implementations, and debugging procedures with automatic testing. It is hard to say which software belongs to a communication or to a imagining subsystem, source code for all micro-controller with different architecture shared. That allow in matter of week to change type, computing power, and power consumption of all electronics in imagining subsystem. Solutions in a software interface for imagining subsystem, benefits communication subsystem.

Sample of successfully solved challenge is a gyro-sensor and accelerometer used in imagining subsystem interface. Gyro-platform for a craft or nano-satellite used two compensated gyro-sensor and accelerometer. The same software but compiled to work with one gyro-sensor and one accelerometer used as add-on to microprocessor controlling movements of a camera's stand, and antenna mount (it will be also correct to say HD camera box mechanism.

From another challenge in software development of a gyro-platform was a hard task by itself, even all formulas for quaternion mathematics, and Kalman's filters was well known, but real, fast, implementation, with less as possible instructions required for date’s processing was required enormous amount of verification in calculations.

On high level imaging subsystem keeps its data (picture/video) in FLASH memory (micro SD memory card with operational conditions -40+125C). Challenge to protect that data from damage by high energy particles, was solved by keeping that data in three separate FLASH devices, with physical location as far as possible from each other. Write operation are performed simultaneously to all 3 devices and read based on majority voting 2-from-3.

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Challenges also were to develop tools for testing of a sealed box with HD camera with helical antenna in in-house vacuum chamber.

Developed technique for incorporation of from-shelf HD camera to be adapted to work on rover/nano satellite. That challenge solved, time for incorporation measured to be account in total time calculation for manufacturing rover/nano-satellite.

Technique for laying protection layers of urethane for low res camera tested, it was not considered as a challenge but rather development task.

Protocol for retrieving data and making low resolution picture by low resolution camera was designed; implemented and verified, interface for mission control was tested.

Challenges was solved for rapid prototyping and manufacturing of all electronics in-house. That was done to make guarantee that manufactured electronics will pass outgassing tests, and will work in vacuum and temperature operation conditions.

Challenge raised by a long schedule asked by launch providers and delays was overcome by abandoning "frozen design" ideology. All nano-satellite designers after private conversation confirmed that delays will be from 1 to 2 years from conformed date of launch. In that period of time technology in manufacturing, electronics, software will be obsolete. Instead of "sketch-up" design was used as a prime method. Equipment, hardware, electronics components, software technique should be investigated, tests to incorporate new hardware/software into all system (mission) was performed, time for implementation and manufacturing hardware/software was estimated, but manufacturing of full hardware can be postponed till "where is your cubesat?" question asked. One of the samples of that type of challenge was orbit determination subsystem. As one of the tools to detect direction to the earth, in 2010, was considered imaging subsystem. Picture taken by low resolution camera can be analyzed, edge can be detected, and direction to center of the earth can be calculated. Special image sensor with good temperature operation conditions was chosen to process gray scale picture. Black and white sensor was sensitive in 720nm and capable to detect starts with apparent magnitude 1 or 5. Those capabilities allowed detecting earth edge on night side of the orbit. Software to convert image to jpeg was ported to microcontroller (jpeg was a deliverable format from imaging subsystem). Technique to detect earth age by projecting unparalleled lines was tested on gray scale picture on PC, formulas implemented to pinpoint position to the center of circular body. All was done to find that imaging sensor was discontinued from manufacturing; substitution was not even

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close to parameters of obsoleted sensor. As a result imagining subsystem’s requirements become "lighter" from functionality. In a middle of 2013 different sensor (now color with temperature operation conditions -40+105C) was appeared on market. That opens back opportunity to make imagining subsystem little bit smarter, than to take just a picture and to deliver it.

3.0 Tests and demonstrations to retire existing risks.

There are group of measures which will be required to retire risks. That includes different studies of a mechanical parts, software, hardware, tests/measures will be performed regularly, and/or on demand.

Test which performed daily/weekly:

3.1.a) daily software tests to verify that added functionality is working, old functionality did not broken, and removed functionality kilted for records. For that tests development process organized to split any software changes/implementation into a 1 day frame. Each added / changed line of code hast to be "stepped" in debugger. Compiled code analyzed on instruction's level. Test scenarios with external measurements / indicators created. Units are tested by implementing test's cases on a PC with serial communication connection to a testing unit.

3.1.b) Mechanical’s part and electronics hardware exposes weekly to a vacuum <1 Torr and 125C degree for 30 minutes. After cooling to 25C degree it exposed to -5C for 8 hours. Monthly added test - after cooling (from vacuum test) hardware exposed to -75C for 3 hours.

3.1.c) In manufacturing carbon fiber parts technological procedure conformed to follow curing process requirements.

Special test performed on demand-

3.2.a) range tests for communication 2km,5km,25km

3.2.b) vibration tests 0-15 Hz for assembled mechanics. Duration of the test is 20 min.

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3.2.c) drop/shock tests for assembled mechanics Duration of the test is 20 min.

Tests for assembled components. Requires additional verification procedures. Planning.

For assembled rover and nano-satellite planned weekly performed vibration tests. Frequencies 0-200Hz. Also are planned acoustic study stand for detecting resonance frequencies 200-5000Hz. That acoustic and vibration test not only for an assurance of mechanical integrity but mostly to confirm that short marriage of a craft and launch vehicle will be not destructive for both sides. Those tests usually performed on third party facilities. And some time launch vehicle owners do those tests by themselves. To pass acoustic and vibration tests needs to have daily available in-house tools which can help to prepare assembled craft or nano-satellite for tests. Basically acoustic test is a loud expose to a sound (139dB) frequencies applied to a testing assembly. It is impressive by it noise. Instead of this we consider to have in-house a stand, where regular sound, with variable frequency, will be applied to the object and sensor can detect vibration, which will be indication of the resonance frequency. In-house vibration table for a frequencies 0-200Hz is also essential part to detect problem in assembly and to verify mechanical integrity. For assembly it is important to perform shock tests. Shock usually come as a result of divorce process, after short-time relations between launch vehicle and mounted on that vehicle craft. Process is unpredictable in our case, to consider that main mission will be a flight as secondary payload. Payload adapters/ Mechanical Lock System/ Mini-satellite Separation System/ Pyro locks (which keep craft still inside payload compartment) will depend on shape and attitude of its "comrades" from cargo bay. Marriage of a craft with launch vehicle was done usually by various pyro-devices, and separation was done by explosion of pyro-component and braking bolts. Each fully or partly assembled object better be testes right away for such stress. We consider having pneumatic shock testing tool to perform such test on demand.

All this will require 3 additional stands to have in next year to retire risks in assembly and development of a craft.

3.3.a) Tests on vibration table. 0-200Hz. Performed in-house. Duration the test is up to 20 min.

3.3.b) Acoustic study stand. 200-5000Hz. Performed in-house. Duration of the study expected to be 1 day.

3.3.c) Shock pneumatic tests. Performed in-house. Duration of the test 1 minute.

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3.3.d) Nano-satellite deployment box will be manufactures to perform vibrations tests on nano-satellite.

There are also tests which we are planning for retirement of technical risks on near future. Those tests can be considered as "demonstration tests", because they will be indicators in a process of readiness of hardware for a main and test mission and also can attract our funs as “milestone” event.

3.4.1 Communication range test (25km) Rehearsal demonstration. One ground station will be located in Stanley Park, Vancouver (or at another place can be considered) and Nano-satellite assembled in Simon Fraser University park (requirements for both location is 25 km of visual site). Nano- satellite will be suspended on a wire to allow free rotation in horizontal plane. To the left and to the right from suspended nano-satellite will be objects representing (a) earth edge visible from LEO and (b) moon. In demonstration (controllable from mission control server) nano-satellite will determined its position and velocity by analyzing row signal from global navigation system. Nano-satellite will report that data to a mission control via backup communication system (via satellite communication). Mission control will calculates the orientation direction for antennas on nano-satellite and ground station and send orientation’s commands via backup communication to nano-satellite, and to ground station via IP connection. All calculation will be with assumption that nano-satellite orbit is equal to the circle and with period of 24 hours. On nano-satellite will be calculated direction to a center of earth by accelerometer and the direction to North Pole by magnetometer. Both antennas on a nano-satellite and ground station should turn to point to each other. Visible movement will be indication of correct performance of such event. Next will be the attempt to establish communication session over noisy environment over Grate Vancouver. Non-noisy path can be chosen as alternative for the 25km test. Planned transmitting power on nano-satellite will be 0dBm(1mWt). Planned transmitting power on ground station will be 0dBm (1mWt). Different transmitting power can be chosen in the test up to 30dBm(1Wt), without exceeding max allowed in Canada 36dBm(4Wt) for 2.4Ghz Frequency-Hopping Spread Spectrum systems. After communication session establishment from the mission control will be send commands to orient nano-satellite to the a exposed object representing edge of the earth (picture) and object represented the moon (another picture). After two orientations turns two pictures will be taken, and nano-satellite will orient itself for a next communication session. Time of session will be provided at previous session. Then second communication session will be established and mission control will be able to retrieve low resolution and high resolution pictures. On third communication session it will be request to record 1 min HD 720p video. Session # 4 will be for a retrieving HD video to a mission control. Then ground station will be assembled to have "rover" configuration and placed on inclination table with 1/6 of the earth gravity and with ability to point antenna to earth located Stanley Park. Session #5 will be provided to confirm functionality of a ground station in rover configuration. Last will be command which will be send to a ground station (rover) to retrieve low resolution camera's box and take a pictures. In that case picture will be delivered to a mission control without RF communication. "Rehearsal" test duration time is 2 hours.

3.4.2 Rehearsal up-side-down demonstration test. All electronics for the nano-satellite is equal to electronics on rover. Deferens (if will be) is in software. As a result rehearsal demonstration test 3.4.1 any time can be switched to "up-side-down" when nano-satellite performs functionality of a ground station, and ground station can be tested as remotely controlled rover. In this case mobility test of the rover can be performed with mockup of a lunar surface. For best regolith simulation will be used rye flower, for rocks and craters can be done quick made mockup from available material, dusted with rye. In this test all “broken” mechanical / electronics hardware can be tested, functionality for reduced functionality of a rover and imaging subsystem can be verified.

3.5. Communication range test (min 100km) demonstration - the same as in rehearsal demonstration 4.4.1 test but two points in BC(or BC + Washington) mountain range will be chosen to have 100km tests. Transmitting power in the test will be 30dBm (1Wt), with attempt to reduce power to 0dBm on both transmitters. Duration of the test the same as in 4.4.1 test.

3.6.1. Vacuum (<1Torr) chamber test/demonstration (with nano-satellite inside) and all systems are working, This test similar to test #1 except transmitter power will be 0dBm and orientation of a nano-satellite will be conformed visually, all picture and video (inside surfaces of the vacuum chamber) will be taken. From a mission control, starting from second session, parameters of the orbit will be faked to a real satellite orbit flying over Vancouver in time of test. That fake orbit’s settings will force antenna of a ground station to orient it into a direction of a real flying satellite. Second HD camera, mounted on an antenna stand (different observation angle than designed for a rover), by commands from mission control will be able to record parch of sky with flying over real satellite. For that test needs to choose proper time at the evening or at the morning for a visible (recorded) conformation of the orientation (attitude), and mobility subsystem. Vancouver weather needs to be taking to account at this test. Duration test’s time approximated to be 2 hours.

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3.6.2. Test the same as 4.6.1, but in vacuum (<1Torr) chamber will be a rover (ground station) powered by its power plant and nano-satellite will be switched to work in a ground station mode (all electronics hardware on rover and nano-satellite are equal). Rover will be inserted to a vacuum chamber without wheels, and gears. Energy storage of a power plant (high volume capacitors) will be charged to a maximum level. Will be checked performance of a power plant with a task performed by imaging subsystem.

3.7. Test of a thermal subsystem of a nino-satellite - the same test as 4.3.1. Instead of vacuum chamber will be nano-satellite suspended on wire supporting free rotation. Nano-satellite will be placed between heat element (+80C) and cooling container (-75C). Outcome of a test expected to be the same test as in 4.3.1. In this tests thermal control subsystem should calculates best directions for cooling and heating and perform autonomous orientation maneuvers to heat and to cool overheated imaging subsystem parts.

3.8. Test with fully equipped rover with impact shield will be dropped from 70 m (around 24 floors of the modern buildings). Before test, control communication session will be established with mission control. HD camera of imagining subsystem will be switched to record the landing video. Rover will be rotated on a suspension support before drop. After impact rover has to be established communication session with mission control to delivery HD video of landing test. Adoptive filters will be placed on solar sensor to simulate earth position by sun. On landing point will be placed box with simulated lunar regolith. For lunar dust will be used rye flower. Test will take 2 hours and time equal 1 min HD video transmission.

3.9. Certification outgassing vacuum test of nano-satellite. It will be flight to Ontario, to test facility. Testing nano-satellite will be placed into vacuum chamber, exposed to vacuum 1x 10-4 Torr, applied heat to nano-satellite till it reached 70C, wait for 3 hours with recording pressure value inside chamber. Duration time is 4 hours. Filming will depend on a permission of the owner of the facility. Results expected - the pressure records will be passed to a launch vehicle provider.

3.10. Vibration and acoustic test, for certification of the nano-satellite to flight (for test mission) will be performed on in-house vibration and acoustic test stand. Resonance frequencies for nano-satellite will be documented and passed to a launch vehicle provider.

3.11. Vibration and acoustic study of a craft's mock-up. All parts of a craft will be manufactured. Special mockup of fixed impulse engines will be manufactured. To have the

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same exact weight of a fixed impulse engine, into mockup will be filled wax, nozzles will be 3D printed from titanium. The same test as 4.10.

3.12. Flight-to-ground test. The same as test 4.4.1. Nano-satellite will be on the orbit. When it will fly over ground station, ground station(rover), based on information from mission control has to track nano-satellite on a sky, and communication session will be indication of a successful mission. In subsequent sessions pictures will be taken and video will be downloaded. It is planned to use 2 ground stations one in Vancouver another Langley/Donetsk.

3.13. Certification outgassing vacuum tests for a rover and impact shield. The same test as a test 4. 9.

3.14. Certification vibration tests for fully assembled craft. Probably will be done at facility provided by launch vehicle provider.

3.15. Test for laser range finder system, has to conform functionality of a laser range finder on 1km, 2km, 4km,10km, and 25 km range. Flat cliff like surface has to be selected to perform such test in BC mountains. Test to be performed under diriment lighting conditions at day and at night time. BC provincial campground will be booked, shashlik and slipping bags will be provided

3.16 Tests for all pyro-devices with separation frame and mockup fixed engines will be performed and the process of separation will be analyzed. This will include individual tests of each frame connection mechanism and all of the mockup craft. For the nano-satellite, such a test is not necessary because it is not allowed to have any pyro devices on board. Instead of this, the nano-satellite's deployable antenna is released, by burning nichrome wire in the lock mechanism

3.17. Brake engine parameters tests. Manufacturer will be asked to verify parameters of the brake engine and the test will be performed on the manufacturer facility.

3.18. Gyro-platform calibration. Measurements on the performance of the gyro platform will be recorded and decisions will be made to either accommodate existing precision, or to use second gyro-platform to improve twice the precision.

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4. Development, verification and integration steps planned for the landing subsystem.

4.1 As alternative for already chosen low resolution cameras can be selected different image sensor. In that case it will be software development to accommodate interface for the sensor, and porting of existing code for jpeg compression into micro-controller architecture. That can simplify hardware for low resolution camera box. One of benefit - imagining subsystem in this case can provide information for attitude control / orientation subsystem of the craft. Together with process of taking picture can be calculated direction the center of celestial body appeared on a picture. Another benefit is a control of a quality of the picture. Despite huge amount of patents on video/image compression, modern compression depends on 200 year's old Fourier transformation. That mathematics allow for "nature" (with source from nature) function, (and pictures is one of such function) to skip small details of the image by skipping high frequency spectrum of transformed function. Controlling level of "skipping" dynamically, by accounting the performance of a communication and power plant subsystems can give advantage in mobility and functionality of a rover. It is a preferable development, but it depended on other tasks in development. Delay in test mission can make such development happened, and "on time" launch on nano-satellite in spring of 2014 can postpone this.

4.2 Manufacturing of all carbon fiber parts for testing mission of nano-satellite flight needs to be done.

4.3 Incorporation of a final version of electronics' hardware for low/high resolution and video capabilities has to be done with all subsystems of nano-satellite. Hardware will be designed in-house, PCB manufacturing by order, soldering will be done in-house. Full circle between incremental hardware versions is estimated to be one week.

4.4. Hardware vibration tests, and acoustic tests on assembled nano-satellite needs to be performed - that will include in-house tests and study, and certification tests on separate testing facility.

4.5 Thermal and /or vacuum tests weekly planned for all flights hardware.

4.7. Another development desirable to finish - for a time been in a queue for a launch of nano-satellite, some advances appeared in hardware of energy harvesting. Incorporation of

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old Peltier’s elements into a thermal control subsystem of a nano-satellite can have visible benefits. Six Peltier’s elements on each sites of a cube can create simple passive thermal control with transfer of a heat from one side to another with skipping transfer of a heat of area inside Cubesat. Additional micro-controller with 1 nano-watt can allow not only to actively control temperature inside HD camera box on nano-satellite but also to harvest initial energy for powering power plant micro-controller. Such approach can benefit main mission's imaging subsystem thermal control.

4.8 Development for synchronization of the data in flash memory of a communication subsystem. That will reduce complexity of a video and pictures transferred from imagining subsystem.

4.9. Constant undated and tailoring of the trajectory calculations software will be performing various necessary, distributed calculation tasks. This will include the visualization, global navigation system satellite's simulation etc. Extending the system with calculating the magnetic field of the earth and pulsar location, are other areas of improvements. The calculation of atmospheric drug and solar pressure is another long planned implementation, with monitoring the correct time for incorporation into the calculations.

4.10. Design and development of a directional determination system with the aid of pulsars is preferred, but it depends on the nano-satellite flight schedule and the delay in launch of the nano-satellite can cause this to happen.

5.0 Critical risks will retire throughout the development and verification activities

5.1 "Rotation wheels" (or "reaction wheels") can be broken because of a mechanical failure exposing them to vacuum and temperature conditions. There is risk with stepper motor and permanent magnets – both risks will be confirmed by 3.1.b). In the case of conformation, another stepper motor can be chosen.

5.3 Ejection of a "face off device" can fail at time of ignition, if the break engine and brake engine nozzle become clogged with the hockey puck.

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5.4 A risk that sensors memory storage, micro-controllers, electronics can be fail, because of temperature operation conditions, or because of high energy charged particles. Design mentioned in [] N.N will be applied.

5.6 There is a risk that configuration with the retrieved and fixed position of low resolution cameras and antenna, will be different to the performance of attitude control. That risk can be mitigated by applying adaptive software, not based on calculations and simulations of the designed craft, but based on performance of a craft in real flight.

5.7 Trajectory calculations are based on in-house developed system. There is a risk that the system can produce wrong values of the orbits. This risk can be reduced by cross-checking different and independent calculations and simulations. Final verification of this risk can be done on the pre-launch procedure when "big" space agencies will be willing to verify the risk.

5.8 First subsystem for determining low earth orbit is based on a GPS receiver with the capability to output raw navigational satellite data in record IDs 29 and 30. There is a risk that such receiver will fail to produce ID records 29 and 30. In case of such event, the second subsystem based on raw data will be used. This risk can finally be verified by the flight of nano-satellite on the test mission.

5.9 There is a risk that distributed calculations of low earth orbit will fail to determine orbit in time. This can be verified by the test flight of the nano-satellite. In this case, time can be measured from the moment of real delivery of the data via the backup subsystem and the time when orbit will be determined. The risk can be estimated by simulation software producing all required "fake" records stored in the mission control DB.

5.12 There is a risk that the first and second subsystem for orbit determination, after main engine burn, will be not able to collect data for orbit determination. In this case, embedded in the communication subsystem, distance measurements can be used to replace filed data collection. The risk can be verified by the test flight of the nano-satellite mission, with testing when orbit will be recalculated based solely on distances measured.

5.13 There is a risk that uncertainty and errors in the orbit determination occur after the correction impulse on the route to the moon. Nothing can be done to verify this risk except the flight to the moon by itself.

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5.15 There is a risk that the solar sensor will fail to detect the sun direction and the system for detecting the nearest center of celestial body, because of errors in the navigational software. Software debugging and verification procedures can be used to lower the risk, but the risk cannot be eliminated.

5.18 There is a risk of wrongly calculated angle of intersection of the path of the craft and the moon and the direction of the firing of a break engine will be incorrectly calculated. That risk can be verified and examined by the test flight of the nano-satellite.

5.19 There is a risk of determining the direction to the sun before firing the brake engine because of time limits. That risk cannot be eliminated, but it can be verified on a mission flight.

5.20 There is a risk that rotation will not reach the desired 5 rotations per 1 sec before firing the brake engine. That risk can be verified on the low earth orbit, when the craft will be on a low earth orbit. Simulation and calculations can estimate the possibility of such an event and that will be done during software development.

5.21 There is a risk that shock from the separation of the attitude control "face-off- hockey-puck" will change rotation vector of the craft. This risk can be verified on mission flight.

5.22 There is a risk that the laser sensor will fail to determine the distance from the lunar surface and ignition command will fail to be in the 10 mks time frame. That risk can be verified in 3.14 tests.

5.24 There is a risk that the separated (from the shell of the brake engine) rover with impact shield will continue to rotate, because of changed characteristics of the loaded spring. At the time of the mission, a mock-up of the craft will be at the mission control center. Before landing, temperature readings from a craft will be applied to a spring and separation on the mockup will be performed. The measured momentum on the mockup will be used to set rotational speed of a craft to adjust rotation to the measured data.

5.25 There is a risk that the "parachute" device/engine will not move out from the rover and the impact shield from falling to the same spot as the brake engine. This risk can be verified in the flight mission.

5.26 There is a risk that terrain will be not as predicted on the landing site. As a result, calculations on time to ignite and the direction of firing the brake engine will be calculated

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wrongly. To reduce risk, mapping of the lunar surface needs to be applied to the trajectory calculation software.

5.27 There is a risk that weather conditions on a ground station will not be allowed to communicate with a craft at a critical moment of time. This can be reduced by transferring all required maneuvering data on each communication session and the next communication session will be updated from mission control.

5.28 There is a risk that noisy environments could ruin the communication session. The location of the ground station will be chosen in non-urban areas (it can be done easily in the Vancouver area). Risks can be verified by range tests of the communication system over noisy environments. The nano-satellite test flight will be with the ground station locations.

5.31 There is a risk that all calculations for the communication subsystem are wrong and communication sessions will not be possible with lunar distances. This risk can be verified with independent methods of calculation, but final verification can be done only in mission flight.

5.32 There is a risk that there will not be enough power to support communication. This will be verified during testing.

5.33 There is a risk that tripling packets restoration will not be enough to support the communication session. This risk can be verified by range tests and the nano-satellite test flight.

5.34 There is a risk that the communication session will have more than the expected errors in packets to transfer data, with enough speed for delivering 15 minutes of HD video. Protocols of communication can be modified in the flight of the nano-satellite and craft. Simple and alternative methods to speed up communication can be chosen. In this case, grouping of the packets can be increased 2, 4 and 8 times to support a fast rate without compromising the tripling of data delivery. Testing with the nano-satellite will verify such method(s).

5.35 There is a risk that FLASH memory used in the communication session will be damaged by high energy particles. This risk can be reduced by tripling the FLASH memory storage and by allowing two versions of the software for all micro-controllers on board. At any time, versions of software, with instruction code located in undamaged FLASH memory, can be re-programmed inside the microcontroller.

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5.37 There is a risk, that different temperature conditions on the craft/nano-satellite and ground station can change channel frequencies. That risk is reduced by auto-adjusting channels based on transmitted data.

5.38 There is a risk of failure of the algorithms and software bugs in on-board avionics. This risk can be reduced by the ability to reprogram all micro-controllers in the flight time of the nano-satellite and craft.

5.39 There is a risk that in the communication session, the ground station communication could be interrupted from mission control. This can be reduced by applying a technique with synchronization of the data inside the communication system FLASH storage.

5.40 There is a risk of failure of the solar sensor's direction determination and the direction to the center of the nearest celestial body because of shortage of a time. The risk can be verified by the test flight of the nano-satellite. With the measurements required for the system to perform the direction's detection, additional time will be added to communication sessions.

5.41 In regards to the imaging sensor capability and the detection of the center of the nearest celestial body, there is a risk processing images from the image sensor and the failure of algorithms for such a determination. If such a system is implemented on the nano-satellite mission, tests of this system will be performed in time, for taking low resolution pictures requested by mission control and alternative methods of determination. Corrections can be updated in-flight of the nano-satellite test mission.

5.42 There is a risk that the rover's helix Communication antenna will have a mechanical resonance frequency in diapason of a 10-200Hz. This can create a problem in certification tests for the full craft. That risk can be verified in tests N.N and correction to the antenna design can be implemented. Mounting for the launch vehicle adapter, can be used to change resonance frequency of the antenna in full mockup tests.

5.43 There is a risk of failure of the ignition of any fixed impulse engines. That risk can be verified in flight only. Manufacturer of the engine will provide data for estimation of that risk. Manufacturer will be asked to embed this into the engine second ignition device.

5.44 There is a risk of wrong time fixed impulse engines ignition. In the case of bad timing for igniting the fixed impulse engines on LEO and on the trans-lunar orbit, parameters of the

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orbit can be recalculated. The second orbit correction/ main impulse burn can then be adjusted to reach the moon.

5.45 There is a risk of varied performance with the fixed impulse engines in a vacuum and temperature conditions compared to expectations. Four accelerometers on the rover will record the process of the engine's burn and on the next communication session, aggregated data will be transferred to mission control for analyses of the orbit determination and engine performances. Difference in performance will be analyzed with the manufacturer to make adjustments for the next engine's characteristics.

5.46 There is a risk of failure of the separation of any frame /engine shell, after impulse is performed. Tests N.N will be performed to reduce such risk. Pyro-device performance cannot be controlled, as it is a parameter obtained from manufacturer.

5.48 There is a risk of higher than expected velocity at touchdown. Test N.N will provide guidance to reduce such risk.

5.49 With the risk that a power plant will fail to accumulate enough energy, daily tasks mentioned in 4.1.a) can reduce a risk, but it is impossible to eliminate it completely. It is expected to have twice the reduced capacity of a power plant, after the craft will be en route to the moon.

5.50. There is a risk that active and passive thermal control for both HD and low resolution cameras can fail. Retirement of this risk is a dream, but the test described in 4.7 can be used for verification. The solution described in 5.7 can be used to reduce risk.

5.51. A risk exists that the camera's li-ion battery can be damaged by out of operation temperature conditions. This risk cannot be eliminated, but only reduced. The test described in 4.7 can assess functionality with the damaged battery and the power provided by power plant.

5.52 There is a risk of failure in the software algorithms which keep stable temperature conditions. The way to reduce this risk is described in 4.1.a)

5.53. A risk of failure, partial or total, of a micro SD FLASH storage because of external radiation events exists. This risk was lowered by [] 3.4, and [] 4.4 and cannot be totally eliminated.

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5.54 Risks of the algorithms' bugs/errors (in implementation of data exchange between units, or between units allocated remotely from each other), can be lowered by 4.1.a), but cannot be eliminated.

Appendix A. List all partner organizations expected to make substantial technical contributions to the team's development and verification activities in the Accomplishment Round.

Jatasonic Technologies Inc 2075 Brigantine Drive, Suite 2, Coquitlam, BC, V3K 7B8, Canada

Adobri Soltions Ltd. #1407 – 950 Cambie st. Vancouver BC, V6B 5X5, Canada

Appendix B. Team vision of a launch contract

In user guides of "Falcon 9", "Dnepr", "Rokot" launch vehicle is stated that minimum time from signing launch contract till moment of the launch is 18 months, all guides are strait forward in the matter of schedule.

In "Space Launch System. Dnepr. User's Guide". On page 76, in figure 18-1 it is described Launch Campaign Schedule with quarterly time line. Big chunk of time planned to be spend is for "Interface Control Document", "Release of documentation for additional hardware", and "Fabrication of necessary hardware", with total of 1 year. That means if launch provider partner SDB-YUZNOE in Ukraine will be willing (as it described in page 49) to provide for benefits of both sides already designed payload adapters/ locks, than 1 year can be reduced to a "Fabrication of necessary hardware" time frame. This is conformed on a page 71 about "LSA(Launch Service Agreement) for launch of a small spacecraft may be concluded 10 months prior to the planned launch date". In this case in "Design and Technical Documentation to be submitted by spacecraft authority" on a page 73 pp.3 "Detailed drawing of the spacecraft adapter interface with LV" will be predefined. "Project feasibility study" is another time reserve which can be used for beneficiary of both launch provider and its customer. On page 7-6 of another document "Rokot User guide" it is specified "Risk management" on second paragraph of 7.2.1.9 it is mentioned "Political Risk" with mentioning a partner company "Astrium" (51% EUROCKOT) providing financial backing all require funding. List of partners in Kosmotras (from Dnepr User's Guide) is impressive too,

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counting 15 members. All this opens the possibility to reduce time frame for launch vehicle integration. Falcon9 user's guide officially is less flexible on schedule, and our heritage not allows to fully estimate that process. From one point, quote obtained in 2011 about 1kg CubeSat showed that "time acceleration" is follow the same rules, in proportion 1-3. From another point, requirements for technical writing includes 19 books for a Rokot, 29 technical documents for a Dnepr, and 7 documents for Falcon9, which can makes Falcon9 more flexible for a selected customers.

Prices for Launch Service Agreement did not showed any significant changes in past 4 years. Established long queue of demands drive prices only high, and restriction on entry to a club, do not help market to conclude fair prices. Best described by a Space Shuttle paradox, prices depend on infrastructure around launch “event” itself. 1:10 ration between fair and “regulated” prices probably will not change in foreseen future.

To summarize, or basically to have a Launch Service Agreement needs to be "useful" for a launch provider somehow, ether by future possible business, or by amount of money. "Competition" frame, with its non-repetitiveness, is not the best attractive characteristic for launch provider, which makes launch "manifest" the "money first" choice. By following "the spirit" of Google XPRIZE useful be idea for discussion: "to lift-off all space equipment production from the earth to the Lunar surface", that (we still believe) can attract attention of today owners of a launch vehicles. It will be logical extension of today space industry status quo, and it is a matter of a time when it will be implemented, just next day after the competition, or in next loop of space rush.

That is our team’s vision of a launch contract - it is possible to touch the moon before end 2015.

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Appendix C. Time required for systems development

manufacturing part	estimation	man-days
rover:		
wheels	2*16 *3d + 2d	98
frame's stepper motors holders	2 *4d + 2*2d	8
Connectors tubes	4 *2d	8
camera's stand	2d + 2d + 5d +2d	11
antenna stand	2d + 2d + 5d	9
Low res camera box(leg)	2* (3d +2d)	10
HD camera box	2* (3d +2d)	10
containers for temp stabilization	2d	2
container for capacitors	5d	5
helical antenna	4d	4
gears for cameras/antenna	2 * (2d + 2d + 3d)	14
power plant	5d	5
assembly	3d	3
Electronics 4 boards	4*5d	20
Nano-satellite		
frame	5d	5
stepper motors harness	5d	5
electronics 4 boards	4*5d	20
capacitor’s harness	5d	5
switch	2d	2
antenna deployment mechanics	5d	5

backup communication	5d	5
power plant	5d	5
ground station assembly	3d	3
craft		
impact shield	10*2d	20
craft frames	7*3d	21
mockup engines	6*3d	18
mockup payload adapter	4d	4
Tests, demonstration tests	14*2d	28
total +10%		385

For 385 man days, with 4 people working it is 6 month. In August 2014 it is possible to start procedures to incorporate craft into a payload compartment of a launch vehicle. Incorporation will include manufacturing payload adapter. Max time frame for this operation On Jetasonic facility 1 month.

Appendix D. Key review and schedules.

Planned to conduct review of tests results sharp after each test done. Schedule published in Appendix E. Anybody from judges and public can attend, 3 days before test schedule can be adjusted. Most important reviews are

test N and summary	planned date & location	Objectives
3.4.1 Communication range test (25km) Rehearsal demonstration.	14 may 2014. Vancouver/ Burnaby/ Langley. BC, Canada.	To see full system is functioning from mission control to rover/nano-satellite.
3.5. Communication range test (min 100km) demonstration	2 July 2014 BC, Canada.	To see full system is functioning on distances comparable to LEO.
3.6.1. Vacuum (<1Torr) chamber test/demonstration (with nano-satellite	2 August, 2014 Vancouver/	To check full system functionality in

inside)	Burnaby. BC, Canada.	vacuum conditions.
3.7. Test of a thermal subsystem of a nino-satellite	17 August, 2014 Vancouver. BC, Canada.	To test performance of a thermal control subsystem.
3.8. Test with fully equipped rover with impact shield will be dropped from 70 m	1 September, 2014 BC, Canada.	To check LS and full system on landing impact.
3.9. Certification outgassing vacuum test of nano-satellite.	Launch date -1 month Ontario, Ontario,	To obtain certification for a flight on launch provider vehicle.
3.12. Flight-to-ground test.	Launch date + 1week.LEO +Vancouver/ Burnaby/ Langley. BC, Canada.	To get experience in controlling craft on orbit, determination of the orbit, full system functionality in real space flight.
3.11. Vibration and acoustic study of a craft's mock-up.	1 October 2014 Vancouver/ Burnaby/ Langly. BC, Canada.	To attract attention of a public [to a scpoe of testing procedures in space industry.]

Appendix E. Summary of each of the tests, including the following information:

test N and summary	planned date	Location	item to be tested	Objectives
3.4.1 Communication range test (25km) Rehearsal demonstration.	14 may 2014.	Vancouver/ Burnaby/ Langly. BC, Canada.	full system	to check full functionality of a nano-satellite and rover(as a ground

				station)
3.4.2 Rehearsal up-side-down demonstration test	26 may 2014	2014 Vancouver/ Burnaby/ Langley. BC, Canada.	full system/broken system	to check full functionality of a nano-satellite and rover(as a ground station) in SNAFU situations.
3.5. Communication range test (min 100km) demonstration	2 July 2014	BC, Canada. (probably WA)	full system	to check full functionality of a nano-satellite and rover(as a ground station), with RF communication on distances comparable with LEO
3.6.1. Vacuum (<1Torr) chamber test/demonstration (with nano-satellite inside)	2 August, 2014	Vancouver/ Burnaby. BC, Canada.	full system	to check full functionality of a nano-satellite and rover(as a ground station)
3.6.2. Vacuum (<1Torr) chamber test with rover (ground station)	15 August, 2014	Vancouver/ Burnaby/ Langley. BC, Canada.	full system	to check full functionality of a nano-satellite and rover(as a ground station)
3.7. Test of a thermal subsystem of a nino-satellite	17 August, 2014	Vancouver. BC, Canada.	thermal subsystem	to check to check full functionality of a nano-satellite and rover(as a ground station), to check thermal subsystem
3.8. Test with fully equipped rover with impact shield will be dropped from 70 m	1 September, 2014	BC, Canada.	full system	to check full functionality of a nano-satellite and rover(as a ground station)
3.9. Certification outgassing vacuum test of nano-satellite.	Launch date -1 month	Ontario, Ontario, Canada	nano-satellite	To certify nano-satellite
3.10. Vibration and acoustic test, for certification of the nano-satellite to flight	Launch date -2 weeks	Vancouver/ Burnaby/ Langley. BC, Canada.	nano-satellite.	full system to certify nano-satellite for flight
3.11. Vibration and acoustic study of a craft's	1 October 2014	Vancouver/ Burnaby/	full system	to check full functionality of a

mock-up.		Langly. BC, Canada.		craft and rover (in assembly)
3.12. Flight-to-ground test.	Launch date + 1week	LEO +Vancouver/ Burnaby/ Langley. BC, Canada.	full system	to check full functionality of a nano-satellite and rover(as a ground station)
3.13. Certification outgassing vacuum tests for a rover and impact shield.	Launch date of nan0-satellite + 2 month	Ontario. Canada	Mobility system+ landing system	to certify parts of a craft for a space flight
3.14. Certification vibration tests for fully assembled craft.	Launch date of nano-satellite + 6 month	At launch provider facility.	full system	to certify craft for lunar flight

Appendix F Verification Matrix for the subsystem

technical requirements	3 1 a	3 1 b	3 1 c	3 2 a	3 2 b	3 2 c	3 2 d	3 3 a	3 3 b	3 3 c	3 3 d	3 4 1	3 4 2	3 5	3 6 1	3 6 2	3 7	3 8	3 9	3 1 0	3 1 1	3 1 2	3 1 3	3 1 4	3 1 5	3 1 6	3 1 7	3 1 8
operation temperature for all mechanism used in imagining system -70 + 125C.		x															x	x	x			x	x					
Dust conditions for mechanisms - particles size 10% < 5mkm < 10% < 20mkm < 10%, < 50mkm < 30% < 0.5mm < 40%													x									x						
Temperature operation conditions for electronics used in imagining sTemperature operation conditions for low resolution imagining sensors -20+85C.		x															x	x	x			x	x					
Temperature operation conditions for HD camera -20+85C./> Temperature operation conditions for HD camera -20+85C.		x															x	x	x			x	x					

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Performance by mechanisms - stand for box with low resolution camera has dual use - as a sent to lift camera for high observation point, and as a leg for movements of all rover.													x						x			x						
Resonance frequency for mechanics must be not equal of a frequencies of another part of the rover and craft's frame, or crafts engines. Low resonance frequency of a mechanism of a imagining subsystem must be high than 30 Hz.					x																							
High resolution camera needs to delivery 5megapixel picture and HD video 720p												x										x						
technical requirements	31a	31b	31c	32a	32b	32c	32d	33a	33b	33c	33d	341	342	35	361	362	37	38	39	310	311	312	313	314	315	316	317	318
common storage FLASH memory 1Gb																						x						
in a range of -20C+85C inside box with low resolution cameras		x															x	x	x			x	x					
in a range -20C +85C inside HD camera's box		x															x	x	x			x	x					
Interface with communication subsystem needs to be capable to delivery (via ground station) to mission control pictures and video in noisy environment around a ground station. Delivery of a video from micro SD FLASH storage needs to be traced as low as possible to broken/lost packets. Restoration of broken data can be done days after communication session.												x	x	x	x	x		x				x						
Orientation precision 0.1 degree for a craft, and 0.1 for a nano-satellite																						x						x
On LEO it is 100 m in position, and 1 m /s in velocity.																						x						
On trans lunar trajectory 100m in position and 1 m/s in velocity.																												
Laser range measurements on descend 10 m in position, with 10																									x			

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measurements per second.																												
Communication subsystem - TX power in peak 10Wt, sensitivity -145 dBm.																						x						
GPS raw data orbit determination with precision of 100m in position and 1 m/s in velocity.																						x						
At least 1% accuracy for impulses of all engines except brake engine. At least 1% accuracy in profile of the performed trust of the brake engine.																											x	
To support soft touchdown of the rover with max acceleration of 362g and expected acceleration of 40g.																		x										
technical requirements	31a	31b	31c	32a	32b	32c	32d	33a	33b	33c	33d	341	342	35	361	362	37	38	39	310	311	312	313	314	315	316	317	318
To reduce rotation of a impact shield and rover from 5/sec to 1 per min after separation from burned brake engine.																		x										
To orient rover and impact shield by impact shield side before impact with lunar surface.																		x										
To support thermal operational conditions automatically.															x	x	x											
To support communication session orientations automatically.																						x						

full stop. And now we can talk. About – How to reach the moon without attracting attention of the medical personal.

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