Team PlanB, Lunar Mission Concept description

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Acknowledgments

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1.0 Executive Summary

It was proven in the past that direct flight mission from the earth to the moon was least expensive and simplest to perform.

Current state of lunar exploration allows taking seriously into account only one type of mission - when a craft flying to the moon will be a secondary payload on a launch vehicle. In this scenario it is hard to predict initial LEO parameters of the craft. Key part of reaching the moon is calculations when craft is waiting on low earth orbit. Calculations for finding times and directions for 5 impulses separated from each other in days: 2 corrections on a low earth orbit, 1 correction/main burn, 1 or 2 corrections on trans-lunar orbit and 1 brake impulse. This scenario does not consider planning maneuvers on lunar orbit. Skipping Seleno-central orbit eliminates design of additional control systems and software capable to perform such maneuvers probably at real time. For optimization of calculations of direct flight with unknown orbit parameters it is better to assume that values of all impulses are fixed, which gives another benefit – makes possible utilizing well known solid state engines perfectly used in the past in numerous space missions.

Choosing direct flight as a main mission scenario for the Team PlanB shifts mission design and implementation from hardware solutions to software solutions.

To achieve mission objectives (to win a competition, and not get broken) it requires to have as little weight as possible on lunar surface. Set target - maximum allowed weight for a rover is 4 kilograms, additional 2 kilograms of impact one side shield protection system makes all dry mass delivered softly to lunar surface = 6 kilograms total. Rover is a 2 wheels design with antenna and observation camera retractable stand. Power harvests from solar panels and accumulation in high capacity value capacitors allows achieve high pick power consumption during communication sessions and brief autonomous mobility movements.

Craft design is driven by existing off the shelf designs utilizing without any customization 6 solid state engines available for purchasing from Canadian companies. The mission craft structure is: rover + impact shield + engines + orientation system mounted together on a6061-T6 aluminum frame with pyrotechnic fasteners. Firing engines sequence is: (a) LEO correction, (b) LEO pre-main burn, (c) LEO main burn, (d) lunar trajectory correction 1, (e) lunar trajectory correction 2, (f) brake burn. After each burn fasteners separate burnt engine with the frame it is attached to from the main frame. Frame designed that axis of the next engine to burn goes through the center of mass of the rest of the craft. Craft attitude is controlled by gyro platform, sun sensor, infrared center of earth/moon detector, reaction wheels with 3 independent axes. Laser range finding system mounted on brake engine frame used to ignite with precision of 10 milliseconds at proper distance from lunar surface. Brake engine profile is critical to choose ignition time

Communication will be on 2.4GHz, hopping frequencies with gain of rover/craft antennas 16dBi (6dBi on test nano-satellite mission). Gain on ground stations 18 dBi (16dBi on test mission). Transmitters 10Wt (1 Wt on test mission). Receivers sensitivity 145dBm.

Orbit determination performed by: (a) GPS system with reporting raw GPS satellite data with post processing, (b) independent raw GPS/Galileo signal processing system, (c) based on time measurement system of communication systems. Ballistics calculations use proprietary software

3 steps set to achieve objectives: (1) test flight mission of a nano-satellite (launch contract signed for March 2012), (2) first attempt of flight mission (planned in April-May 2013), (3) final attempt to win competition flight mission (planned in fall 2014). Test nano-satellite mission includes all systems required for a "attempt-to-win" flight. None of these planned missions was accomplished yet. The reason – lack of awareness of all the launch vehicles "market" lucrativeness. We are not the only team who was confused by the Cold war "rocketeering". At times when Canadian Space Agency was forced to keep its 4bln satellite on shelf with storage price of 0.5bln per year what are the flight chances for a low-budget jokers from unknown Canadian Team involved in Google Lunar X PRIZE competition?

Main goal of participation in this competition is to promote idea of having a technological base/plant on Moon with ability to produce/make all required space exploration equipment on Moon utilizing 3-d printing technology.

Table 1.1 Mission and summary

Test Mission	Nano- satellite Flight. To demonstrate "ready" status for a main mission.	total mass
Launch	Launch contract with Interorbital system. Second launch of IOS Neptune launch vehicle, march 2012, postponed	1.34kg
Power	5 HCC, 1Wt harvesting solar cells	0.4kg

Mechanical	frame, anodized aluminum, carbon fiber	0.12kg
Communication	2.4 GHz, hopping frequency, nano- satellite's TX power 1Wt, Ground station's TX power 1Wt, RX sensitivity -145dBm, triple packet send over 3 different frequency channels, majoritarian 2-from-3 for packet's restoration, CRC protection, data synchronization on "connect" state, time measurement support for orbit's determination. Antenna helix, fixed, deployable.	0.12kg
Attitude	Active, 3 axes, 3 stepper motors	0.15kg
Backup communication	TX, RX	0.15kg
Imaging system	1 low resolution camera	0.05kg
HD Imaging	1 HD Camera in hermetic container	0.2kg
Command and data	main board computer with 4 CPU, 1 GPS receivers and 1 front end RF GPS receiver, memory storage with triple reservation.	0.15kg
Ground stations	Location Canada, USA, Ukraine. Design - lightweight moon rover and simplified version of the rover with just 2 freedom axes	

First attempt's flight	Planned for April 2013, postponed because of a delay in Nano-satellite mission.	262kg
Rover	2 wheel designs, powered by solar panels, with active antenna positioning, stand for low resolution 4 cameras. Includes communication system, power harvesting system, mobility system.	4kg
Impact shield	designed to adsorb impact of a fall up to 300m from lunar surface, protects only one side of the rover, to support orientation at impact, separates from craft's frame in time of burn the break engine, and special device holds orientation in time of fall after separation.	2kg
Break mpulse engine	weight of the engine, fastening system, frame connects to rover and impact shields	16.2 kg
Translunar correction fixed impulse engines	weight of the engine, frame, fastening bolts, connected to a frame of a brake impulse engine	6.5 kg
main impulse engine	weight of the engine, frame, fastening bolts, connected to a frame of the brake impulse engine	194.8kg
1 LEO orbit correction engine	weight of the engine, frame, fastening bolts, connected to a frame of a main impulse engine	3.2 kg
2 LEO orbit correction engine	weight of the engine, frame, fastening bolts, connected to a frame of the main impulse engine	3.2kg
attitude	3 hockey pucks with 3 stepper motors, connected to a	2kg

correction system	frame of the brake engine.	
adapter/MLS	different for different launch vehicle: Dnepr/Falcon9/Rockot	10kg

Second attempt's flight	Planned for fall 2014, postponed because delays in first attempt, that fight MUST be redesigned to fix mistakes made in original design.	252kg

2.0 Study Background and Assumptions

2.1 Introduction

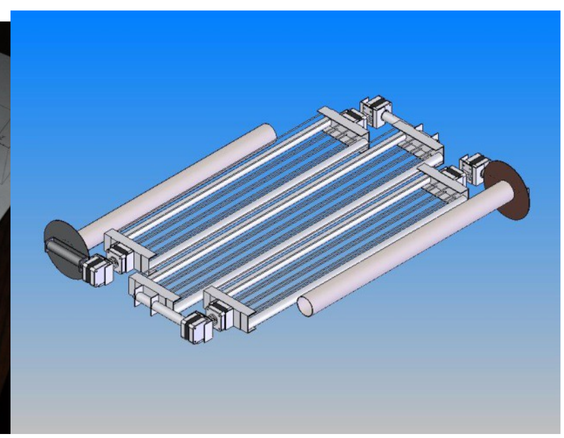
Project was started in November 2009. Our team kept low profile to avoid premature steps before analyzing technical and business possibilities in competition participation. We started with careful study of Boris Chertock books. Unfortunately for history of space exploration from small cohort of key rocket designers in the past century he probably was the only one who left behind "know how" memoirs. Other designers did not have such opportunities due to government restrictions, luck of "writing" skills", or mostly because engineers were too busy. Book was written in Russian and author mixed real memoirs with "know how" information data about space projects. It was the smart move made by old man; otherwise the book could be considered as "restricted" technical information, with all consequences of a word "restricted" meanings. Needs to mention that translation to English was done by NASA and some "between the lines" know-how information got lost.

After carefully analyzing technical and business aspects we realized that main claim for the competition: "to be on time for a creation of new business-oriented space industry" is not achievable. Without political support any space projects must be considered as an amateur's attempt. Such "political" or "government" support was not allowed by the competition rules. Goals like "to be a starting point" for the new "space" business was visible probably as a decoy. Team decided to participate because we are amateurs by nature and because shifting attention from "business" aspects to "technical" focus gives us a chance in the competition as a small team with strong skills in software development and problems solving. Possibilities were analyzed to join other teams but that approach was unfortunately abandoned, because two teams with approach close to ours quit competition at the time of our entry.

In 2010 was made experiments and studied the mobility platform and the flight itself. Two prototypes were made of the rover with an attempt to achieve minimum weight. Four-wheel design was skipped right away because of exceedingly high weight of critical parts - motors and gear boxes. First version of trajectory calculations software was created and scope of a development of full ballistics calculations was estimated. For a second, crawling version of rover a prototype of a one "flip-flop" leg was made. Once prototype was finished we realized even without powering stepper motors that the weight is far away from the estimated original target of 6 kg dry mass on the lunar surface, design requires gear boxes with 1:10 ration that brings extra weight and will have problems in dusty conditions. Three-wheel design was also not a promising approach but there were not many choices and in December 31, 2010 we entered competition with 3-wheel crippled design under name of a Canadian Team PlanB. It actually was PlanC at that time, but name "B" was chosen as more attractive. Imaging systems were analyzed and "buy-and-test" approach was chosen for selection. Available HD camera tests were simple - camera has to pass physical 1.5 meter drop test and thermal survival -75C +125C test. Vacuum chamber tests were not necessary for HD camera. A technique to seal camera inside of epoxy cocoon with memory storage and power connectors coming from outside of the sealed area was developed.

pic 2.1.1. wheel from 3 wheels design

pic 2.1.1 crawling rover



In 2011 we developed ballistic calculations software capable of calculating gravitation potential of the earth on low earth orbit. That was done basically because in private talks with NASA representatives was obtained information that nobody will help our team in orbit determination because of "regulation" reasons. Another push for proprietary software was:

commercially available orbit’s determination software for the same "restriction" reasons did not disclose their methods of orbit calculations. Despite the fact that their numbers were exact match of a test case of old FORTRAN program from openly available "Space track report #3" “no comments” policy from an industry leader removed all hesitations towards trajectory calculation software development. It was decided that is very risky to depend on vital element "orbit determination" for planned mission and participation in the competition overall. Developed software calculates impulses performed by fixed impulse engines (profile for the engine can be entered in a convenient way) and has ability to optimize vector of impulse and ignition time semi-automatically. Auto modes allows to set “target” landing point on lunar surface and adjust vector and time for engines individually. Later versions of software were extended to be able to work in "distributed" calculation mode.

In 2011 2-wheel rover configuration was chosen. Long helical antenna with reflector creates a problem in such configuration. Placing wheels far from each other creates difficult in movements on terrain with approximation of 5 holes with depth 1 meter, and 20 holes with depth 0.3meter in 100 square meters area. Instead of providing just physical stability by mechanical components was decided to add software stabilization based on gyro and accelerometer sensor readings. Instead of gear-box carbon fiber springs in wheels was chosen, that was risky move but resources to make a custom gear was limited at that time. Concept "without software rover is not capable to move" was adopted.

Calculations of signal’s levels transmitted directly from lunar surface to earth’s located ground stations were performed. The calculation for two helical antennas (one on rover and another on ground station, with gain=16dBi) shows that with peak 10 Wt power transmitter (PAE=40% require 25Wt consumption at peak) it requires -145 dBm sensitivity to support communication session. To transfer all 16 min of the HD video will require 20 hours with 0.5kWt*h, and power harvested 6Wt (from solar cells) can make it in 3 days. Nordic BT microchip was chosen as core module. BT protocol’s enhancements were investigated for ability to be switched off. Hopping over 3 frequency channels not only reduces requirements for ground station prototype's operations license (and operation in nano-satellite test flight mission) but also allows fixing broken packets in “noisy” conditions. Stripping standard BT's "CRC" and "address" fields off in packet allows achieving ability to restore original packets in “noisy” environment (around 50 Wi-Fi/Blue-tooth devises working simultaneously) with 10% damaged data. Communication originally was implemented as AT type modem, but idea was abandoned considering requirements of short time session window. Because of that we developed new implementation with fast automatic delivery of data without frequent acknowledge packets. Another enhancement in communication system is ability to measurements signal travel time for orbit determination.

Additionally for the craft orbit determination development of proprietary software to process raw GPS/Galileo signal was chosen. Two systems, one with full GPS receiver capable to output raw data, and another with front end RF receiver with internal extracting Geo-positioning satellites data were developed.

2.2 Purpose and goals.

Following goals were chosen for the mission:

a) Win competition (main)

b) Study, information collection and analysis of ways to deliver lunar equipment to the lunar surface.

c) Gain experience in practical physical mobility on lunar surface.

d) Conduct experiments how to use 3D printing equipment in dusty lunar condition.

e) Collect data for implementation of metallurgy on the lunar surface.

Last four goals were based on the "shadow" goal in our project - to establish technological base on the Moon. That requires solving numerous questions about metallurgy and 3D printing.

In our mission we spent big chunk of time in attempt to find answers to questions raised from the "shadow" goal. Best metallurgy and 3-d printing experiment will be - conduct it on lunar surface with tools and material delivered from earth. This limits weight of material and tools close to zero weight.

2.3 Assumptions and Approach

2.3.2 Past Design Starting Points

Prior to October 2013 was investigated possibility of different trajectories. This was mainly done using our "in-house" developed software. Software is not only in a form of a deliverable executable, but rather a tool with source code changeable for rapid investigations of questionable scenarios. Executable form trajectory simulation software was used for distributed calculation. In that case designated web server delivers to each "calculator-cell" small task. After the finish process reports results to the server and asks for the next task. Task is to break different trajectories calculation to small tasks for investigation and delivery of each task to separate "cell" lying on the server. Mission control system is based on this capability. One designated server allows creating, entering, and sending group of commands to craft/rover/ground station. Access to the server can be done from any point in the world via standard internet browser. Extension and additions to the functionality of a mission control allows dynamically change processing of the mission data. Designated computer performs calculations of the craft orbits and additional GPS satellites. Mission control server delivers results of the distributed calculations to ground stations.

Ground stations software was integrated with the mission control server. All communication traffic is recorded in DB as original data. MySQL was chosen as one of reliable standards. HTTPS industry standard can be used to secure communication from mission control to a ground station communication hub. Such approach has side effects - slow data delivery and low security. To overcome these limitations we developed in-house total-encryption software. Data from satellite to database entry point will be encrypted by individual random sequence.

Without reliable gyro platform capable to sense rotation precision higher than 0.5 degree rotation per hour in 1g environments it does not make sense to attempt any space flight. We developed in-house gyro-platform capable to detect earth rotation with gyro-platform sensors based on solid state gyros. A low power consumption micro-controller used for data processing gyro platform. All system consumes 0.12Wt. System uses well known mathematics but was optimized to reduce power consumption. Instead of industry standard Kalman filters was developed similar but simpler approach with a porpoise to save instruction cycles in a goal to reduce power consumption.

Lots of experiments have been conducted to choose proper methods to manufacture mechanical parts. Carbon fiber was chosen as main structural material. After investigations and research we chosen method of waiving/knotting carbon fiber inserts. Selected epoxy used for carbon fiber, was found conditions under which outgassing characteristic in vacuum is acceptable for a space flight. To guarantee outgassing we established sequence of steps that required for epoxy to cure. As result of mold process investigation was chosen different molds creation process for carbon fiber manufacturing of different rover and craft parts. Intensive use of 3D printing for all molds allowed collection of crucial information for "shadow" mission study.

Communication system was studied on all project stages, from design, prototyping, manufacturing RF devices to software design and implementation. IC components were selected after trying at least two different ICs with temperature operation conditions from -40C to +125C. The process of design and manufacturing of main computer board was studied. We adopted rule: 2 week-2 prototypes, designed PCB ordered on Tuesday with delivery on Friday, soldering SMD on Saturday, fixes in design on Sunday and check of fixes with second order of PCB on Monday. This approach allows getting any electronic device ready for the flight certification in 2 weeks.

Ready status was declared for different missions. For test mission - 3 month ready status. For main mission 6 month ready status. Time for technological operations of all manufacturing stages was measured. These time measurements give us estimates for the full system production time. Such approach allows focusing on software development to keep hardware be ready in known time frame. Basically it is the same principle as it was applied by abstractionist's artists to compare with old painting schools. In the time of Rembrandt to create one picture painter need to have full house of helpers, 1 year of work and that one picture was one implementation for one idea created by the "production house". Abstractionists on other hand tested "idea" by simple sketch-up like technique. If the "idea" was worth to implement it, then longer time can be spend, and full house of can be used to make "implementation" by Old's Rembrandt style.

Two weeks in both cases reserved for vibration and outgassing tests/certifications. Vibration tests are the most critical. Resonance frequencies of manufactured probe/nano-satellite can be calculated at a design time by using simulation software, but "real" frequencies can be verified only by real test. Vibration table for frequencies up to 100Hz is designed to test 4kg rover. A 100-500 Hz frequency audio system with 1 Wt was designed to be working with vibration detectors. Fixing undesired resonance frequencies (inside of 12-500Hz range) can be done by applying carbon fiber patches or removal of small masses of carbon fiber (based on simulated results) and tests can be repeated on vibration table and acoustic station.

2.4 Growth, Contingency and Margin Policy

Assuming mission's type is a "second" payload type. That makes it hard to follow reasonable and not reasonable recommendations and practices. Boris Chertock suggested use of 25% growth for full system. Some other recommendations set that number to 30%. Contingency can be in hands of chef designer and can be spend in various ways, rules, engineering culture folklore. It can be done at stages of design and/or manufacturing or on occasion, or even at time of a space flight. All this vital and unavoidable in case of big groups of designers, working independently on separated parts. Design done by small team looks seems to be free from weight grow problems, but it is not true. Only main goal can be arbitrary in decision. Limitations like type of trajectories or type engine type (fixed impulse) makes problem just worsens.

Rules chosen in design case:

- Rover has fixed mass 4kg - no more than 4 kg;

- Impact shield has mass 2kg.

- 15% will be added to the mass of each fixed impulse solid state motors, as the weight of the frame.

- Placing gears to wheels of the rover require additional "rotation" wheel(s) in attitude control system.

- imagine the picture of the rover each morning and common sense are prime source for contingency and margin policy.

With all it summed up a weight for trajectory calculation has to confirm reaching the moon with total mass 250kg + 10% as a secondary payload mission. Craft under any circumstances are bigger that "small satellite" type of 190kg. It is assumed to be in category less than 300kg.

2.5 Mission(s) highlights. Preamble.

Two flights are needed to reach the moon. First flight may simply fail, something got forgotten, something was not taken into consideration, and something went wrong with company that delivers us to the orbit. Radio connection may fail, or there a strong x-rate spike happens that leads to malfunction of processors memory flash and it was impossible to re-program the processors.

Because of that in any case we need to plan two missions.

In order to fit in the competition schedule we need to plan for an extra year (till the end of the competition) that is December 31, 2025.

One extra year means that end of 2014 is the deadline of the second mission.

In order to make a second mission attempt we need to learn how:

1. be able to orientate satellite with precision of 0.01 degree in relation to closest (distant) celestial body.

2. be able to calculate orbit at any mission stage.

3. Verify all secondary ways of orientation and connection.

4. Verify energy collection and storage system.

5. Confirm system of energy distribution to the consuming components is functional and controllable.

Attempt to fly to the Moon on something small like nano-satellite is absurd. It is very hard to plan finishing of second school year without completion of the first year. We can create any type of design, make and test rover, test communication equipment in vacuum and thermal boxes, draw and simulate behavior of satellite, nano-satellite and rover, apply any software simulators. We can show and tell, make videos, attend conferences, popularize travel to the Moon and stars, look for investors and explain them what good we are and how brave our team is. We can collect public donations do fundraisers by printing t-shirts or space money, write business plans and defend them, have recognitions and win prizes for Mission Control Description. But still, until nano satellite performs every point from 1 to 5 (above) and does it without looking back at uncle with big radar or at aunt confirming the hit area with humankind artifacts, until then we can consider all those as Simulation of High Level Activity. Moreover all representatives of all space industries of all countries and nations will not be able to help by definition – space amateurs for them are some kind of tiny green people which may exist but not being considered seriously. Thank you to the agencies for publishing open information. No genial, simple, avant-garde or any other ideas cannot get us closer and complete flight to Moon until we learn ourselves to perform the above mentioned 5 points.

In order to verify real rover abilities would be good to make earth connection station as a variant. This station will work not in vacuum but on earth, but this approach will allow save on auxiliary equipment. It will be getting power from a wall outlet as opposite to solar powered batteries. Maneuvering and analysis of data from gyro platform will be similar to lunar. At the same time task of pointing antenna to flying satellite is more difficult than pointing it from Moon to Earth.

This test mission was planned for March 2012. All amount of sending to the orbit was paid. Since then design was done with main goal – minimize time required for making of nano-satellite to three months. This includes complete testing and everyday improvements of main mission and nano-satellite design. In February 2013 we worked out all details of such approach. Three months before the launch our team is guaranteed to know how to make nano-satellite and six month before the main mission our team can make probe and rover for the flight to moon. Three and six months were picked as a security measure in case adverse condition of launch market or export limitations (from Canada to third countries) manufacturing of all electronics will be on the territory of the country of launch.

There are possible alternative variants of test mission. There is still open possibility to fly for 180000 for one nano-satellite with new players in the space carriers market. Will it allow reach moon in main mission – questionable, probably even negative. So far, apart from big words, new space business has not shown any interest in partnership with small amateur groups.

2.5 Main Mission Description

The primary mission will be realized by a planned direct flight to the Moon. A launch vehicle will carry the payload into Earth orbit. After separation from the launch vehicle, an automatic system check will take place, resulting in an “READY” message being sent over a backup communications link to mission control, acknowledged the craft is ready to accept commands from mission control

At the same time, the craft starts an orientation sequence. The sequence includes a search for the Sun using a solar sensor. The direction vector to the Sun from the gyro platform data will be stored for a future use. If the solar sensor cannot find the direction of the Sun, then orientation to Sun will be determined by the maximum current of one of the solar panels. The craft turns its solar panel to the Sun, to maximize the power harvest.

In all maneuvers, the craft uses an “adaptive attitude” system. The attitude system calculates 125/25 possible movements, performed by stepper motors, and records the resulting quaternions, representing special rotations, corresponding for each possible 125/25 moves. These values later allows a precise control of the direction of the craft, and to optimize require movements to reach orientation during flight.

The process to stop unknown rotations after separation, or after any "sleep" mode, will be done by a calculating the period of function, with measured voltage values, from the solar panel, or from the solar sensor. To stop rotation, which is equal to maximization of a measured period of the function, the first attempted movements will be chosen the biggest angle of rotation from 125 quaternions in the stored table.

After separation, and after "sleep" mode, on command from mission control and planned over backup communication, recorded GPS/GALELEO raw data will be will be transferred back to the mission control. Continuous recording of the raw signal is not necessary, and we only need to get 5-8 records from the navigation satellite of the past 30-60 minutes. That data will be analyzed in mission control, and orbit will be determined by trajectory simulation software running in distributed calculations mode. If the GPS system's raw data will be not available, the a raw stream of digitized data on L1 frequency will be recorded, with a later extraction of a raw navigational satellites coordinates and velocity data from recording on board.

After determining the orbit parameters, mission control sends a command for craft to find direction to the Earth. The craft then starts rotating itself with constant speed, and using infrared sensors, detects edge of Earth. Each crossing of the edge will be a signal to mark and store for processing the direction, provided by gyro-platform. The direction to the center of the Earth is determined via calculation. The adaptive attitude system has a special mechanism for tracking rotation movement without involving data from the gyro-platform. Rotation initiated from any still position can be reversed, and all movement commands can be repeated precisely in backward order by attitude controller. To confirm the period of the orbit, to preform maneuvers, craft perform rotation to conform Earth edge detection. It is not intended to keep constant track of the detection to the center of Earth.

When direction to the center of the Earth and direction to the Sun determined, mission control can a send sequence of orientation maneuvers to orient craft for communication session on the main communication frequency of 2.4GHz. The commands includes vectors in an on board system of coordinates and time marks for each vector. The attitude control linearly extrapolates vectors in the time between two time marks. If the linear extrapolation will be not possible, then mission control can reduce time between time marks. The session with ground station/mission control confirms the functionality of communication system. In these sessions, not only will the transferred telemetry data be sent, but also data from the imagining system.

From the communication session via main 2.4Ghz system, additional data will be collected, measurements of the time for an RF signal to travel from ground station to craft and back from craft to ground station. Two consequent "loop" measurement records can be used to determine orbit of the craft.

From that moment, all efforts will be concentrated on finding proper trajectory from LEO to the lunar surface. Some preliminary data about a possible orbit will be available. This data will be confirmed or adjusted based on measured data. Communication sessions will confirm functionality of the craft, and orbit parameters. If for some reason LEO will be lower than expected, then the effect of atmospheric drag in trajectory calculations needs to be adjusted.

Before first orbit correction payload adapter/ MLS adapter separated from a craft.

The first orbital correction firing will be the first task on the way to the Moon. The purpose of this firing is to allow the craft to stay on LEO orbit, before beginning the optimal trajectory to the Moon. Earth based calculations will calculate the direction and time of the impulse. The sequence of rotations and movements to perform this impulse will be the same as during the communication session. This orbital correction will be done 10 minutes after confirmation of the direction to the Sun, and the conformation of the direction to the center of the Earth. Before the firing, a command to point the craft's correction engine into the calculated direction will be executed. Rotation of the craft around the axis crossing the engine nozzle will be performed. This rotation is required to stabilize the craft during engine firing. In this operation, and similar, the pre-recorded 125/25 possible movement plays an essential part. After the impulse is done, pyro-bolts separate the part of the frame with the used shell of the engine. A decision to eject or keep this additional weight on board depends on the desired trajectory. The additional mass of the empty engines can be used to make the next impulse less in value. In any case, the stored quaternions of possible 125/25 rotations have to be re-evaluated because the difference in craft mass. During, the gyro platforms accelerometer will record values of accelerations to integrate measurements of the impulse.

The second LEO correction is planned after a longer duration in time after first one. In the time interval between these two corrections, the craft in orbit can perform numerous tests like confirmation of orientation, taking pictures of the Moon, and areas of the Earth. Here, the functionality of HD cameras can be verified, video can be transmitted. The duration of these sessions on LEO will be short: probably from 30 sec to 3 minutes maximum. HD video will be send over by portions, and 15 minutes of recording will require 2-3 days to transfer, using ground stations around the world. The second impulse makes an orbit correction before main burn impulse. Sequence of the movements will be the same as on the first correction: verification of the direction to the Sun, verification of the direction to the center of the Earth, orienting engine to the vector of firing, rotation around firing direction to stabilize craft in time of firing, and ignition at a specific moment of time. If the second correction brings craft closer to the Earth for a main burn, then air drug can slow down craft and second and third orbit will be impossible. After the second correction, there is a risk of deorbiting after one or two orbit.

The main burn will occur approximately 1/2 orbit (30-40 min) after the second. Sequence will be: ejection of the shell of the used engine, stopping rotation (5 min), finding the direction to the Sun (5 min), finding the direction to the Earth (5 min), orienting craft's main engine to the firing direction (5min), rotation craft around firing direction (5min). This is 25 minutes total in total which should be enough for 1/2 of the orbit. It will be desirable to have plenty of time been the second and main burns, but his depends on a chosen trajectory of the flight to the Moon.

After main impulse burn, the craft will be on trans lunar trajectory, there will be a short time window to collect as much information as possible for the trans lunar orbit determination. The GPS main, and GPS GALILEO front end RF system needs to record as much data as possible. GPS satellites flying at distances of 5 Earth diameters and their transmitted signal is beamed toward the direction of Earth. Ideal will be last, before leaving beam, recorded data. That max, far from the Earth, recordings will give max accuracy of "to-the-Moon" orbit determination. After that, on the way to the Moon it will only be possible to measure RF signal travel time. The error in orbital position at this part will be much bigger than on LEO. From that moment, until next orbit correction(s), an expected 2-3 days will pass. Another method of orbital determination is to measure 3 directions, each in different time - to the Sun, to the center of the Earth (which has good visibility on trans lunar trajectory), and to the center of the Moon (which will have bad visibility from a craft). Based on these measurements, and times, it will be possible to get 3 point on the orbit. Repeating the process can give another 3 coordinates. On the way to the Moon communication sessions become longer, and ground station shifts around the world become more permanent. Windows of communication sessions when 2 ground stations simultaneously can see the craft are important in determining the orbit. Such events are planned with the pairs of Donetsk and Kazakhstan, as well as Hawaii and Cook Island. These pairs have near perpendicular connecting lines. This will allow measuring the RF signal time to better estimate the trans lunar orbit.

The word "orbit" means something flying around another body without collision, and it assumes parameters like period, inclination, and 5 others values to be converted into to useful information numbers. Because in our simulation software, the orbit can be calculated from position and velocity vectors, we take this vector as a prime source of all decisions. For determination the orbit, we take in account value of impulses recorded by gyro platform accelerometer. Error in impulse's calculations will be less than error in position measurement by RF signal travel time. Trying in trajectory calculation / orbit's determination to match values for a position, we assumes the velocity is less volatile. Distributed calculations can match the results, done by a series of measurements. The gyro-platform accelerometer records 3 axes of accelerations values.

The impulse # 4 on the way to the Moon is essential to choose a landing point. Coordinates of the landing can be different from planned because of imperfections of main impulse burn. The main impulse by itself does not bring craft to the Moon. Correction is mandatory; otherwise the craft will miss the Moon. Preliminary study showed that changing direction of the firing with a fixed firing time, gives the same flexibility on the landing point, but precise targeting is impossible because of the nature of fixed impulse engines. The calculated error was 600m on latitude and 300 on longitude. In the case of flight with help of a gravitation of sun and moon landing time will be at lunar night, best precision (less error) at targeting the landing point depend on lunar time. If time close to sunrise, then error will be bigger, if time close to "lunar midnight" then precision is better. The ideal will be landing at Sunrise points (terminator line on the moon) in a flight mission with help of gravitation of the Sun and the Moon

On trans lunar trajectory the orientation of a craft become the challenge: only the direction to the Sun can be correctly determined. Two maneuvers are performed every half an hour: a picture of the Moon and a picture of the Earth, then the data is transferred to the Earth. The picture of the Earth (the Sun at this moment will perfectly eliminate the Earth) will accompanied with the old vector of the direction to the Earth (drifted value). At mission control adjustment value vector can be calculated and that correction will be delivered as a earth correction. Gyro-platform to perform such operation needs to be running with 'zero-drift" precision of 0.5 degree per 30 min minimum.

Before correction of the trajectory based on orientation data from solar sensor the direction of the engine's firing can be set. This command will be followed by rotation of the craft, and ignition in preset time. The integrated value of impulse will be recorded. The shell of the engine will be ejected. The acceleration vector data of the impulse will be transferred in a communication session to the Earth, in 20 minutes after craft stabilizes itself. The impulse value can be the last "reliable data” for the trajectory Possibility of a additional correction on trans lunar trajectory can be considered instead first or second low earth orbit correction (it require different mount of the fixed impulse engine of the craft's frame). That fourth correction is less accurate. Available information to determine orbit will be direction vectors to of the Sun, Earth, and Moon, and time when vectors measurements was done. We hope that time measurements performed by the communication system can be useful during next 24 hours. The forth correction engine has to be ejected without use.

Next are the last moments before the landing on the Moon. The Moon will be close to the craft and it is possible to use the same sequence as on Earth orbit: to rotate the craft by detecting the edge of the Moon with 2 infrared sensors, and calculate the direction to the Sun by solar sensor. Next task is to orient the craft that the break solid state engine placed to the desired firing direction (direction vector is different from an vector direction of trajectory and lunar surface. The craft is rotated and engine ignition will now be based on data provided by a laser range finder. In a study provided in 2011, it was around 24km from the Earth surface and 4 sec before impact to the Moon . Precision of the ignition (in study) was 10ms, and error was around 300m on top of lunar surface. Burn of the last engine takes 20 seconds, and that time, when engine burns, it also ignite the pyro-bolts to prepare for the separation of the engine shell from the landing craft (6 kg in mass) consisting of the rover and impact shield. Rotation of the rover and impact shield will be compensated by carbon fiber spring, released on separation.

Calculations done in 2011 showed that from random chosen low earth orbit landing point on the moon can have error in longitude of 600 m, latitude of 300 m. In study the expected landing time was 6-24 hours before Sun rise.

At ignition of the brake engine, the HD camera starts to record video, three previous (pre-landing) recording (length 1 min each) will be done separately to satisfy a better observation of the approach to the Moon. Recording will stop after 5 minutes, and this will cover the entire process of landing, 4 minutes before impact and 1 minute after. To record the performance of impact shield, it will be illuminated by an LED after the burn has finished. Low visibility conditions in the case of landing before Sun rise will not be ideal, but 5 minutes of HD video at Sun rise probably will be an interesting and scientific extension to 15 minutes required by rules of competition.

From the moment of landing, the mission will be moved to a mobility phase.

2.5.1 Trajectory Baseline for the mission..

Main mission probably will a secondary payload type of mission, and it will be less control on parameter of the initial LEO orbit, if no control at all. It will be desirable to stay on low earth orbit and wait for a moment, when position of the moon and sun will help to get additional impulse for a craft on the way to the moon.

In 2011 was practiced the exercise (study) to find all firing directions and time for a mockup main mission. As a starting point (random chosen waiting LEO in that exercise) was assumed ISS orbit. Two impulses was optimized to get as best as possible (help from Sun and Moon) main impulse. Optimization capability allowed setting target and finding optimized direction/time values to reach distance little behind (10%) the lunar orbit. That approach outcome the possibility to rendezvous with moon in 6 possible points - each with around one earth day from each other. Last point was 1 days before sunrise, but speed on rendezvous was much higher than on first possible moment of landing. Third impulse (orbit correction) was used to guide craft to desired point on the moon. Brake impulse burn was calculated and allowed to reach target point on the moon with precision of 600 meters from target point. Exercise showed that chosen approach is correct. Waiting on random chosen

LEO for a period of 1 month will give 2-3 possible "good" trajectories to reach the moon. Second trans lunar correction of the orbit was skipped - software performed perfectly.

Instead of investigating the physics of each celestial body and creating formulas was proved try-and-maximize-the-result method for that optimization. It was useful after finding "draft" solutions, to re-optimize impulses direction/time to keep the target point on the lunar surface the same. That approach can give time window and tolerance in orientation vectors for each impulse. In exercise was found that precision in orientation must be 0.01 degree minimum (matched Boris Chertok requirements for orientation for Luna-9 = 1/120 degree.

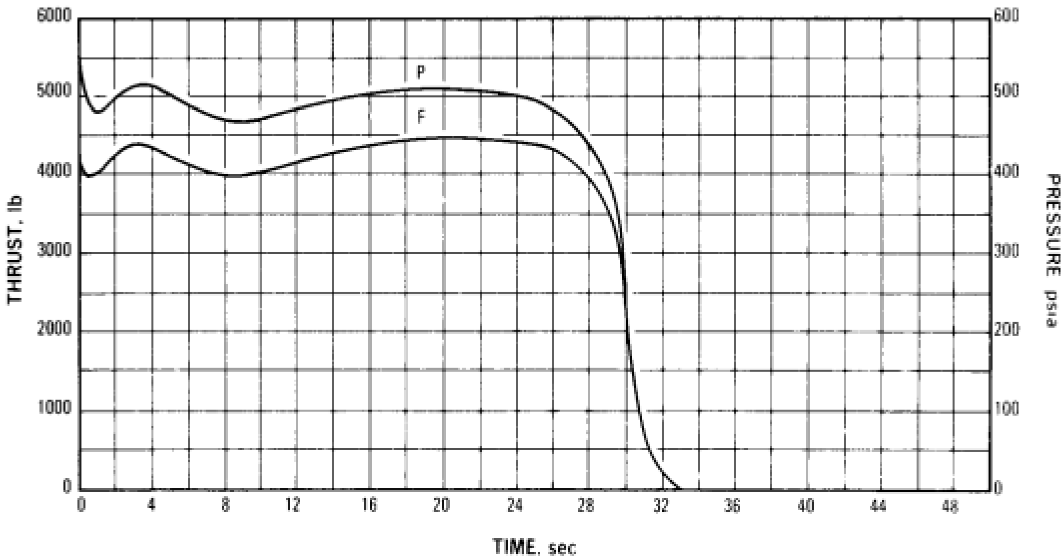
2.5.2 Mission Analysis.

2.5.2 Propulsion analysis.

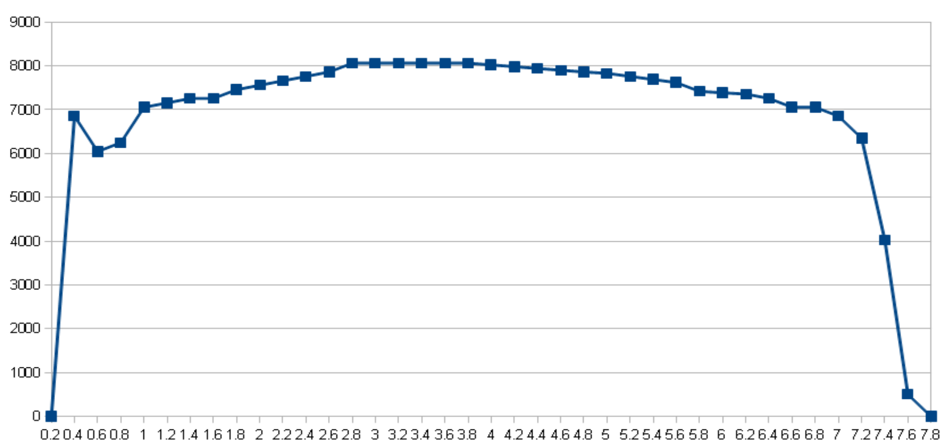
For analyses was chosen data available from public domain - two fixed engine produced by [http://www.atk.com/wp-content/uploads/2013/02/ATK-Motor-Catalog-2012.pdf ATK] company, and amateurs fixed impulse engine manufactured by Canadian space enthusiasts club [i.e. http://www.canadianrocketry.org/motor_files/37148-O4900-BS-P.pdf] .

Engine TE-M-604-2 - Data on similar STAR 24 rocket motor. It was qualified in 1973 and flown as the apogee kick motor (AKM) for the Skynet II satellite. The motor assembly uses a titanium case and carbon-phenolic exit cone. Total impulse, 126,000 lbf-sec. Total mass

pic 2.5.2.1 Engine TE-M-604-2

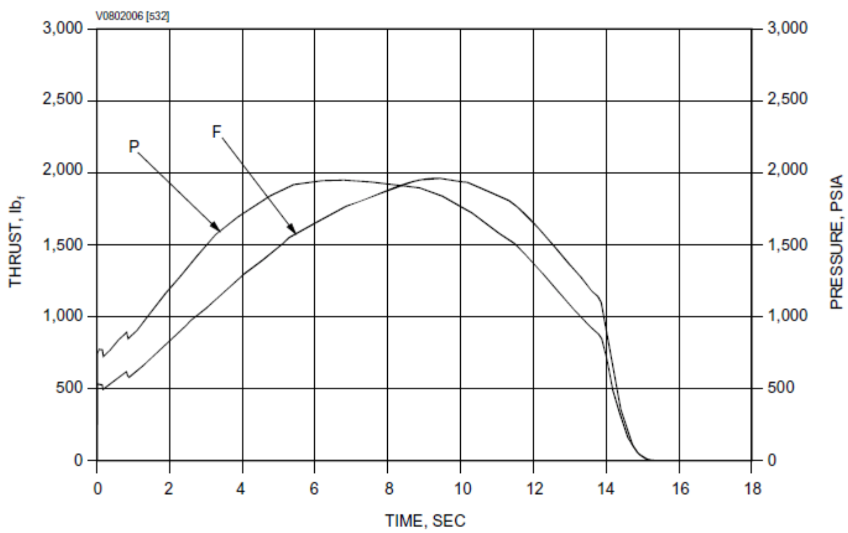


pic 2.5.2.2. Performance of the engine used in exercise in 2011 for finding optimized trajectory to the moon. Main impulse engine.

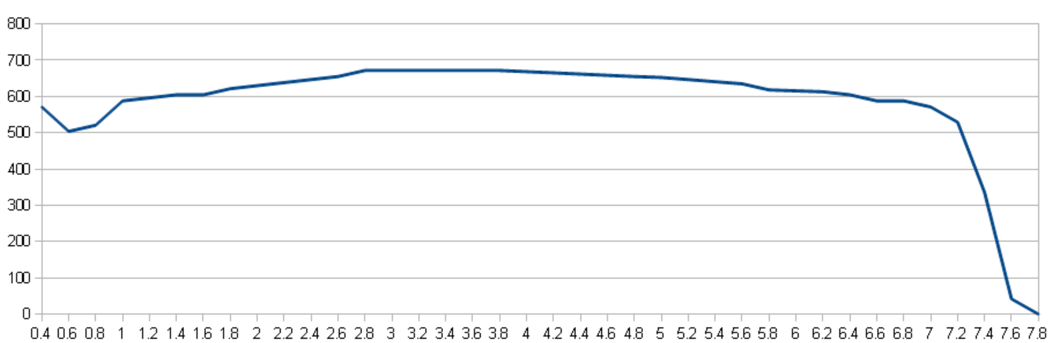


Engine TE-M-956-2 E-M-956-2 for brake impulse. Available data was instead of STAR 12, but for STAR-12GV rocket motor (around twice the performance of a START-12) . It was served as the third stage of the U.S. Navy/MDA Terrier Lightweight Exoatmospheric Projectile (LEAP) experiments. The motor first flew in March 1995. The stage has TVC capability, head-end flight destruct ordnance, and utilizes a graphite-epoxy composite case. Total impilse was 20669 lbf-sec and total wight 42kg.

pic 2.5.2.3 engine START-12GV performance



Pic 2.5.2.4 Corresponded hypothetical engine used in 2011 study with total impulse 4441 lbf-sec:



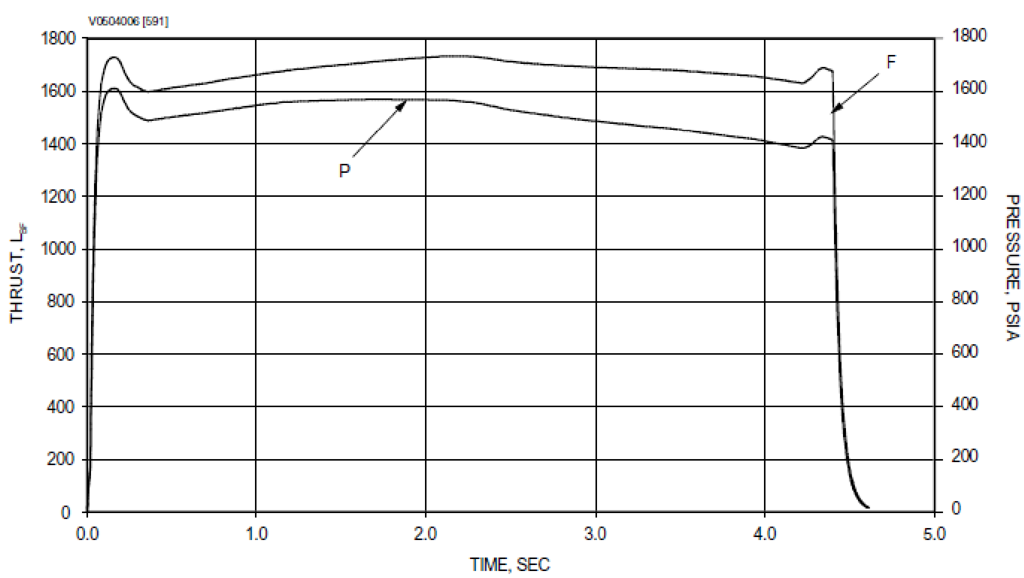
In analysis in 2011 was chosen hypothetical engines with parameters twice less than maximum value of trust and 3 times less burning time. Total impulse was 53292 lbf-sec. That was 2 times less performance of a "real" engine. ATK engine from catalog [] was chosen by closest propellant weight, instead of closest total impulse. It was done to try the worst case scenario. May be ATK engines with required parameters will be not available, and it will be only choice but to buy something twice less good as from industry leader.

If to choose instead of match by "propellant wight" do match by "total impulse" parameter, then in catalog present another engine TE-M-251 (66,600 lbf-sec, 123kg total) engine or TE-M-521-5 (71800 lbf-sec and 126kg) engine. Last one is popular according catalog and was more tested.

Again, as in a case of main impulse, engine was chosen to match weight of a propellant. According total impulse (in 2011 exercise) it will require just half of a total impulse for a break engine for optimal trajectory with sun and moon helping to achieve additional gravitational bust from main celestial bodies.

Ideal engine for break impulse in that case will be to probably TE-M-1076-1 (STAR8 engine) with total impulse 7,430 lbf-sec and weight 17 kg (famous The STAR 8 was developed and qualified as the rocket assisted deceleration motor for the Mars Exploration Rover for JPL's Mars Pathfinder program).

pic 2.5.2.4 STAR8 engine trust:



2.5.3 Communication analysis.

For analysis was chosen well known formulas was chosen well known formulas

*Power Prx = Ae * S*

*Power Ptx = S * 4 * 3.1415 * R * R / Gt*

Frequency f= C0 / A

*Electric field Strength E = Z0 * H*

*Power Flux Density S = E * E/ Z0 = Z0* H * H*

*Distance A0 = G * lambda * lambda / (4 * 3.1415)*

Distance R = Distance Rx - Tx

Where:

C0 = 3e8 m/sec

*Z = 120 *3.1415 = 377 Ohm*

f - Frequency

Gt - antenna Gain of a transmitter

Gr - antenna gain receiver

R - Distance

Ptx - transmitted power

Prx = receiver power

E - Electric field strength

H - Magnetic flux strength

Z0 - characteristic impedance of free space

C0 - speed of light in vacuum

3.1415 - constant

lambda = wavelength

Ae = effective area.

That formulas output on frequency 2.45Ghz

With transmitter antenna gain 16 dBi

With Receiver antenna gain 16 dBi

Distance 400,000 km

And transmitting power of 10Wt

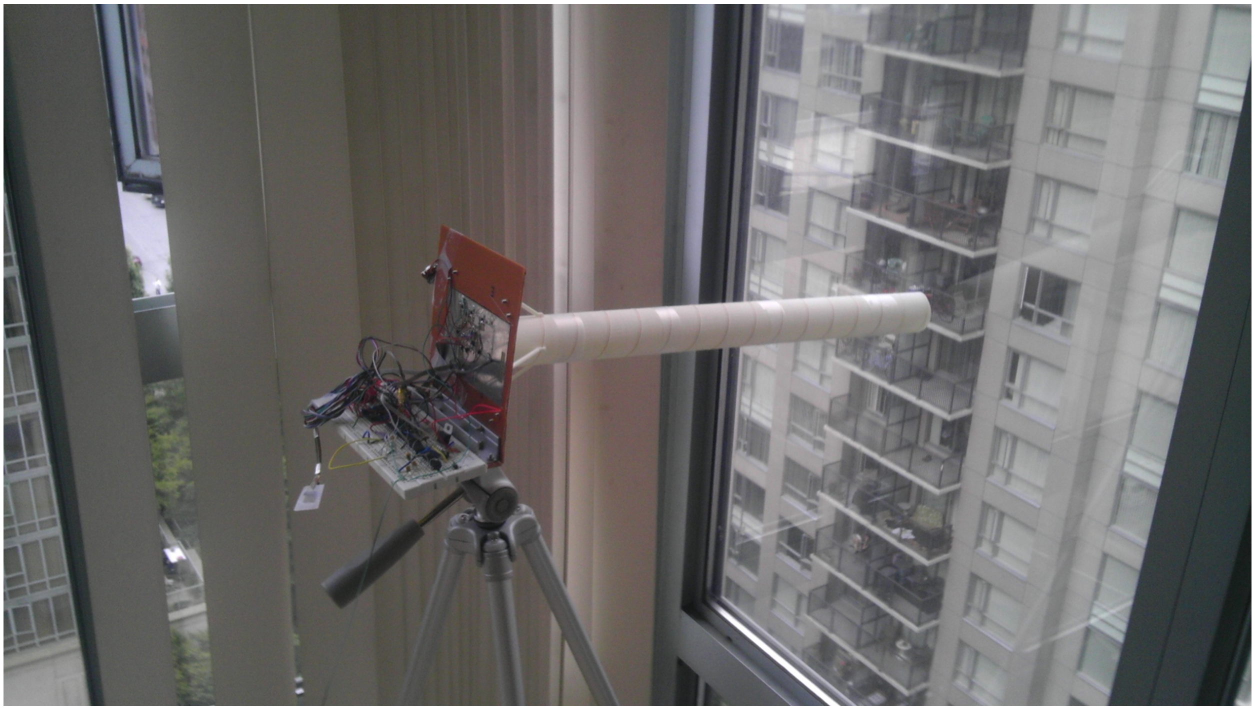
Electric field strength 273 V/m

Magnetic field strength 724 A/m

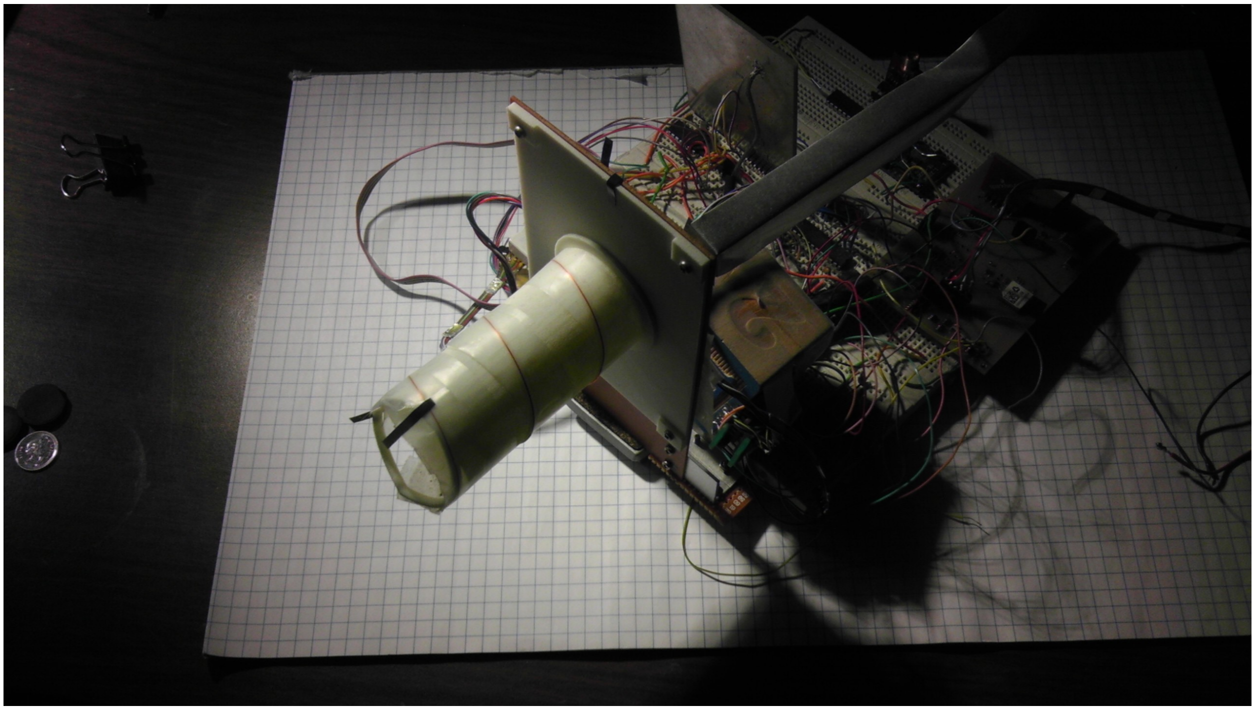
*Power flux density 198 W/m*m*

With BT receiver capable to pickup of -83 dBm signal that require additional 5 low noise amplifiers to achieve required gain on receiver (140-83 + 0.9*n)/12.5 = n. With n = 5 it showed good outcome. Additional gain achieved by antenna, using instead of regular helix, "polarized-reduced-size" winding of antenna, or mounting instead of 1 receiver's antenna on a ground station 4 of antennas will add 3-6 dBm gain on receiver sides. Use of "polarized" winding of antenna require additional rotation of antenna during communication session (antenna on a lunar surface will be in unpredictable angle position), But that can save weight on a rover.

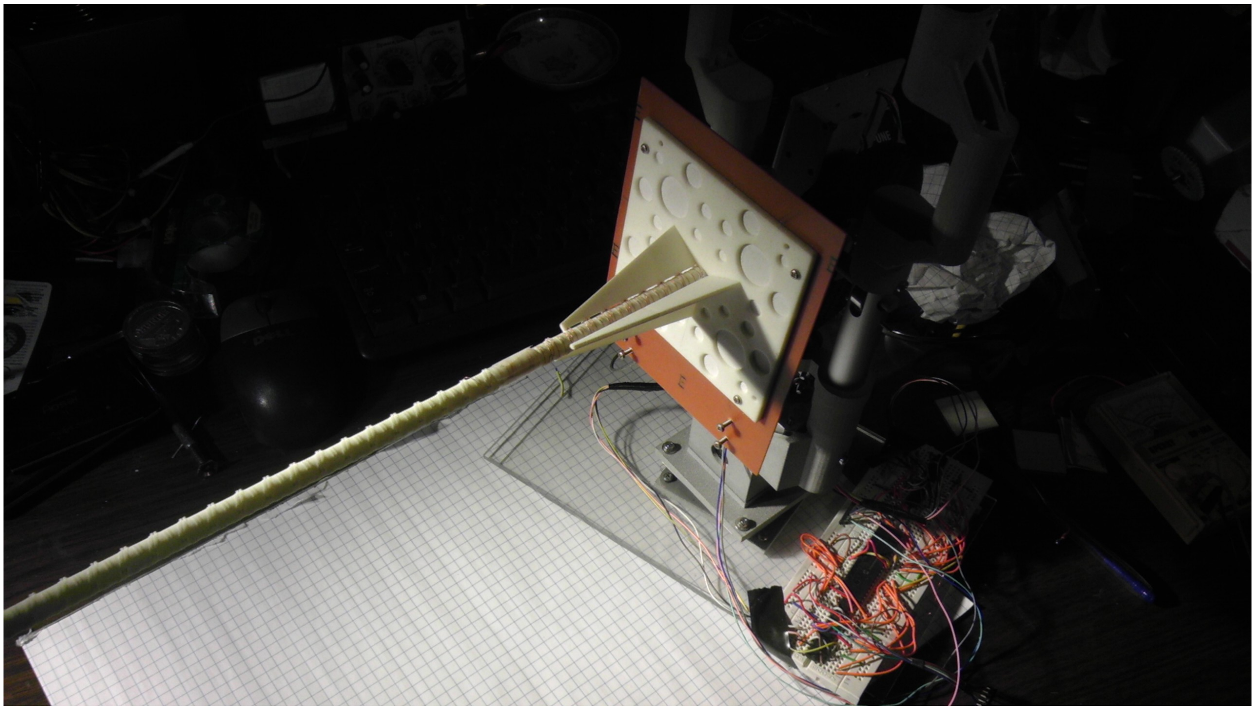
Pic 2.5.3.1 helical crafts/rover /ground station 2.4GHz communication antenna



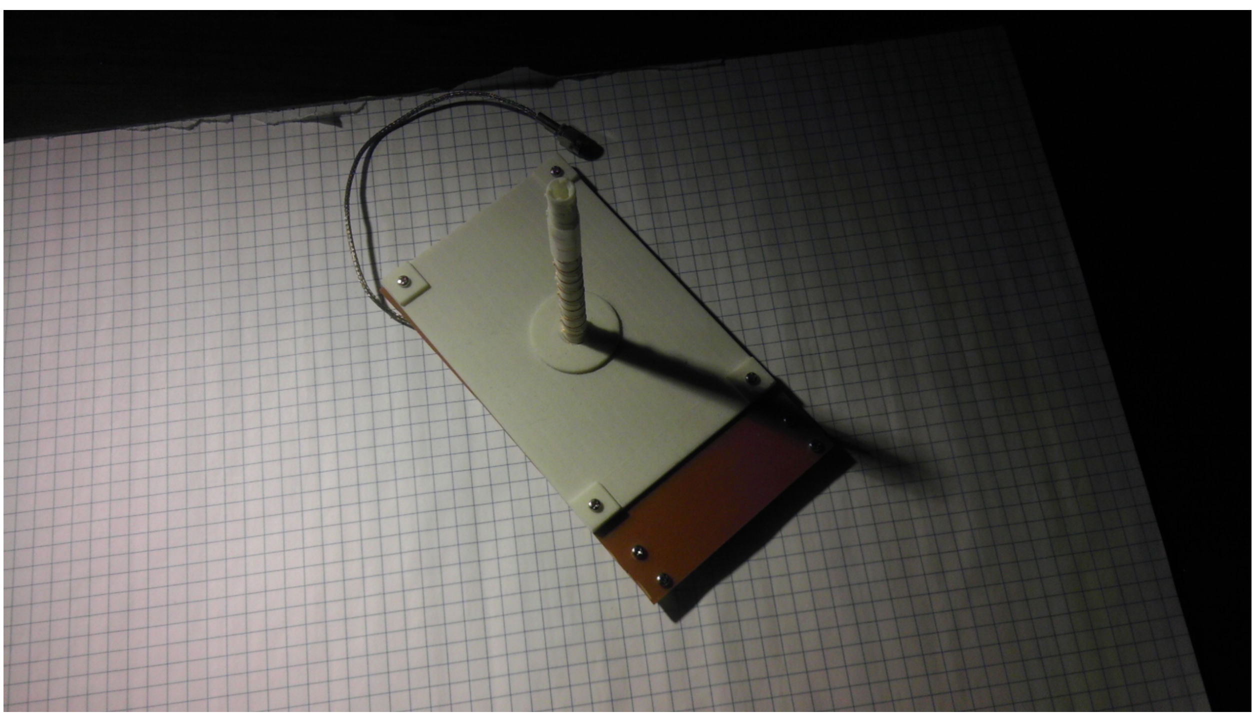
Pin 2.5.3.2 helical nano-satellite 2.4GHz communication antenna



Pic 2.5.3.3 helical polarized-reduced-size crafts/rover /ground station 2.4GHz communication antenna



Pic 2.5.3.4 helical polarized-reduced-size nano-satellite 2.4GHz communication antenna



2.5.4 Mobility analysis.

Results of a required possible mobility movements including jumps with a help of a low resolution camera stand can be summarized in table

Pic 2.5.4.1 no gear stepper motor N17H118

	Mobility of the rover on		moon				9.8	1.622			
	N17H118			N17H018		mass kg	acseleration	F=			
Nm=	0.36			0.21		2	1.622	3.244		1.081333	
Dia=	0.27	Radius=	0.135							0.11034	
distance=	0.108	Deformation %=	80	delat deformation=		0.027			load per spring (kg)=		0.108
slope angl=	10	cos(angl)=	0.984808								
gear			1		1						
F=		3.28269251	3.232821	1.914904	1.885812						
F delata=		0.03869251	-0.01118	-1.3291	-1.35819						
Presision of measured travel:			Jumps performed by rover:								
acs sens=	0.001			F=m*a							
Dela T=	0.01			a=F/m							
error m/s^2=	0.00001		1.616411	a=	0.942906						
1 sec acs err=	0.001		0.2	t=	0.2						
s=V0*t+a*t*t/2			0.323282	V0=	0.188581						
T=	60		0.82	t jump=	0.47						
V0=	0		-0.28023		-0.09052						
s=	1.8		0.41	t/2 jump=	0.235						
		h=	-0.00378	h=	-0.00047						

Pic 2.5.4.2 Mobility of the rover 1:4 gear in N17H018 stepper motor

	Mobility of the rover on		moon				9.8	1.622			
	N17H118			N17H018		mass kg	acseleration	F=			
Nm=	0.36			0.21		2	1.622	3.244		1.081333	
Dia=	0.27	Radius=	0.135							0.11034	
distance=	0.108	Deformation %=	80	delat deformation=		0.027			load per spring (kg)=		0.108
slope angl=	10	cos(angl)=	0.984808								
gear			1		4						
F=		3.28269251	3.232821	1.914904	7.543249						
F delata=		0.03869251	-0.01118	-1.3291	4.299249						
Presision of measured travel:			Jumps performed by rover:								
acs sens=	0.001			F=m*a							
Dela T=	0.01			a=F/m							
error m/s^2=	0.00001		1.616411	a=	3.771625						
1 sec acs err=	0.001		0.2	t=	0.2						
s=V0*t+a*t*t/2			0.323282	V0=	0.754325						
T=	60		0.82	t jump=	0.47						
V0=	0		-0.28023		0.175383						
s=	1.8		0.41	t/2 jump=	0.235						
		h=	-0.00378	h=	0.132479						

2.5.5 Trajectory analysis.

From 2011 study. Initial LEO used was from first goggled Kepler's elements of International Space Station :

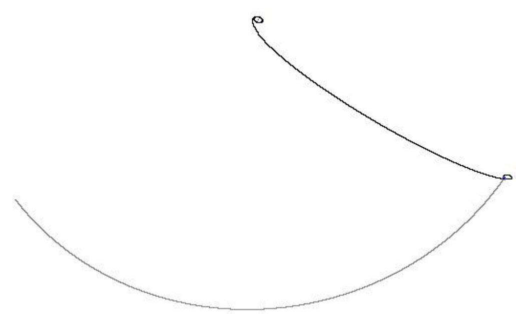
name="ProbKeplerLine2" value="1 25544U 98067A 11062.66968330 .00024457 00000-0 18183-3 0 9357" 2" value="1 25544U 98067A 11062.66968330 .00024457 00000-0 18183-3 0 9357"

name="ProbKeplerLine3" value="2 25544 051.6480 342.0829 0005279 039.9625 033.1957 10.72669582704292"

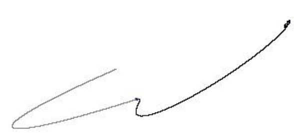
Two first impulses was 5 days before main impulse - LEO was little bit low than initial ISS - and if you ask me the reason why = I can say only - that was found in optimization runs during 1 week.

Then it was main burn, and a correction - trajectory actually looks like a corkscrew – probe travel 1.1 distances from the Earth to the Moon. Then fly little back to the Earth, makes a turn pulled by Moons gravity, and landed on the Moon. Precision landing can be controlled by a small impulse done somewhere from 1/3 to ¼ earth-moon distance to the Moon (in time that distance was 1.5 days). Error in 1 degree of firing vector (3 corrections on a way to the moon) still allows reaching the moon but error was in 500 km.

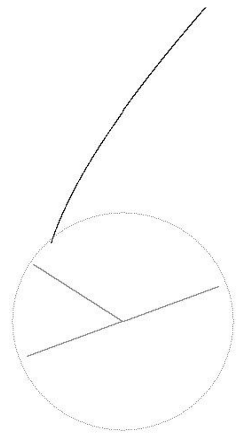
Pic. 2.5.5.1 Landing: (view from North pole) big ellipse is a trajectory the moon, small dot in the middle it the earth:



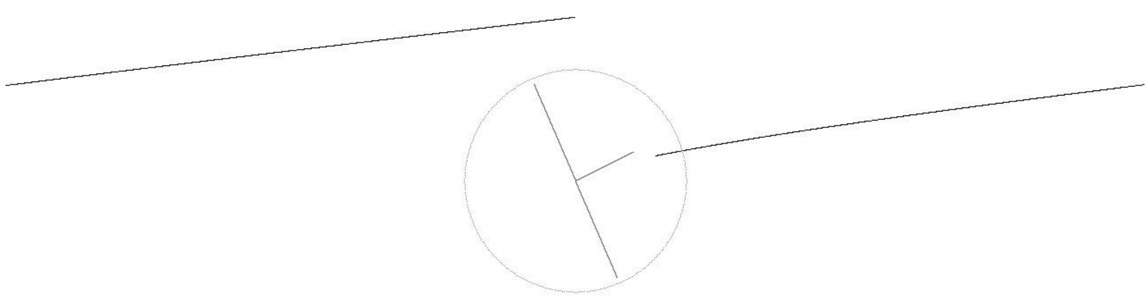
Pic. 2.5.5.2 Landing: (view to equinox direction):c. landing: (view to equinox direction):



Pic.2.5.5.3 Landing: Moon view X, Y (from North Pole): short gray line is a direction to the earth; big gray line shows day-night (sunset-sunrise) line. In exercise was used target point S2W15 degree.



Pic 2.5.5.4 Landing: Moon view Y, Z (to equinox direction): left line is a bug in visualization algorithm. Again short gray line connects 2 point - center of the moon and closest to earth point on the moon (S0W0), big gray line is day-night line in the moment of landing.



Basically big gray line on pic 2.5.5.3 & pic 2.5.5.4 mean that landing happened at midnight of lunar night. Pictures showed first possible landing trajectory. Next 5 was around 1 day after each other. And each was close to sunrise, but each time speed of approaching the moon was 100m/s bigger. Also time window for third correction become smaller and precision of firing become less tolerant (error in orientation was 0.001 degree) that made last trajectory unpractical.

After that study trajectory calculation software was also modified to support test mission of nano-satellite.

For trajectory simulation on low earth orbit was implemented 3 different groups of functions in source code. At first (for Earth's simulation as a dot mass) Kepler's elements of a satellite (eccentricity, inclination, longitude of ascending node, argument of perihelion, and mean anomaly) converted to position and velocity, then plugged to a calculations. To confirm that conversion and simulation is "OK" was ported (as a second group) source code from http://www.projectpluto.com/source.htm (reverse conversion from position and velocity to Keplers element). After some sign adjustments was confirmed that dot mass simulation is correct == after one orbit (or 1 day flight) satellite's Keplers elements has had an error lower than 1.E-10. Then comes third group – old Fortran source code from "SPACETRACK REPORT NO 3" (http://www.amsat.org/amsat/ftp/docs/spacetrk.pdf) function for SGP4 gives better precision than SGP. Then model was switched from dot mass to a something better suitable like GEM-T3 or JGM3. Basically this model deals with "flatness" of the earth (or orange/pear like model of earth). All those models helps to calculate value a gravitation potential with assumption that earth is not a perfect sphere. Recursive function Pn(cosTtetta) does not take much time to implement. Was confirmed that coefficient J2=1082.6360229830E-6 has biggest impact. In first lines of a report was stated: "The NORAD element sets are "mean" values obtained by removing periodic variations in a particular way. In order to obtain good predictions, these periodic variations must be reconstructed (by the prediction model) in exactly the same way they were removed by NORAD." This statement was confirmed by analyzing source code – mean motion and semi major axis (eccentricity) was readjusted before use. Also by reverse conversion was confirmed that period/mean-motion of the orbit is different from "mean" mean motion (in terminology of a report), and different from "mean motion". Difference btw all 3 values is 4 seconds each. To avoid such errors was decided to use position and velocity from SPG4 model and compare that data with corresponded values from a model with J2-J8 and Sxx + Cxx coefficients to simulate gravity. Result was discouraging. Error was in 50km after one orbit. An attempt was made to minimize difference. It was found that best match (17km in position and 8m in velocity) was when instead of cos(co-latitude) was used cos(latitude) – which is a nonsense - it shows that earth has more flatness on equator instead on a poles. Then it was manual attempt to match SPG4 data by adjusting J2 coefficient (commented out code in function IteraSat in http://www.adobri.com/misc/tra/tra.cpp) – was found that J2 was to be increased in some moments 10 times. Was done attempted to adjust initial velocity and position from SGP4 model to match result after one orbit simulation – this also did not bring suitable match. Bug was in a model as it was found later.

Forward and backward conversion of (X, Y, Z) coordinates to latitude and longitude on the Moon surface was implemented. Optimization method: "target practice to a point on the Moon surface" was added. It is possible to set target as latitude and longitude point on the Moon and try to reach it by (a) adjusting time of a main impulse engine firing, (b) adjusting correction impulse (around 2kg) firing angle. Results are – adjusting (a) – gives the error 300km, combination (a) and (b) gives error 600m.

Last modification for orbit's simulation software was done in December 2012. Finally problems with orbits close 1/2 of earth diameter was solved. Gravitation potential was accounted.

*U (gravitational potential) = sum of Unk = fm/r (r0/r)**n Pnk(sinTetta) [Cnk*cos(k*lambda) + Snk*sin(k*lambda).*

Where Tetta is a latitude and lambda is longitude. Cnk/Snk is a constant from a model. fm= G*M, and r0 is equatorial radius. Partial derivative is (page 90 from [Akesenov's lectures] r0 is equatorial radius. Partial derivative is (page 90 from [Akesenov's lectures http://vadimchazov.narod.ru/lepa_zov/lesat.pdf]):

Then: Unk separates into 3 parts:

*(part 1) Rn(1/r) = fm/r *(r0/r)**n*

*derivative => Rn(1/r)' = (n+1) * fm * r0**n * (1/r)**n = (n+1) * fm * (r0/r)**n)*

from page 12 is

*Pnk(sinTetta) = (1-sinTetta**2) ** (k/2) * dk (Pn(sinTetta))/ d(sinTetta)**k = cosTetta **k * dK(Pn(sinTetta))/ d(sinTtta)**k*

*now cosTetta**k separated from Pnk(sinTetta) to get clear part 2 and CosTetta**k will end up at part 3.*

*(part 2) Znk(z/r) = dK(Pn(sinTetta)/ d(sinTetta)**k = Pnk'[n][k]*

Derivative => Znk(z/r)' = Pnk'[n][k+1]

also for K == 0 is:

Zn0 = Pn(sinTetta)

Finally cosTetta goes to a last part:

(part 3) Qnk(x/r,y/r) = cosTetta**k * [Cnk*cos(k*lambda) + Snk * sin(k*lambda)]

= [Cnk*cosTetta**k * cos(k*lambda) + Snk * cosTetta**k * sin(k*lambda)]

= [Cnk* Xk + Snk*Yk] and

Xk = (cosTetta)**k * cos(k*Lambda) Yk = (costetta)**k * sin(k*Lambda)

or recurcevly:

X0 =1, Y0 = 0

X1 = cosTeatta * Cos(Lambda) = x/r; Y1 = cosTetta * sin(Lambda) = y/r

X[k+1] = Xk * x/r - Yk* y/r Y[k+1] = Yk* x/r + Xk * y/r

D(Unk)/D(x) = d(Rn) / d(1/r)   * D(1/r)/D(x)   * Znk   * Qnk + Rn * d(Znk)/d(z/r) * D(z/r)/D(x) * Qnk + Rn * Znk * [D(Qnk)/D(x/r) * D(x/r)/D(x) + D(Qnk)/D(y/r)*D(y/r)/D(x)]

D(Unk)/D(y) = d(Rn) / d(1/r)   * D(1/r)/D(y)   * Znk   * Qnk + Rn * d(Znk)/d(z/r) * D(z/r)/D(y) * Qnk + Rn * Znk * [D(Qnk)/D(x/r) * D(x/r)/D(y) + D(Qnk)/D(y/r)*D(y/r)/D(y)]

D(Unk)/D(z) = d(Rn) / d(1/r)   * D(1/r)/D(z)   * Znk   * Qnk + Rn * d(Znk)/d(z/r) * D(z/r)/D(z) * Qnk + Rn * Znk * [D(Qnk)/D(x/r) * D(x/r)/D(z) + D(Qnk)/D(y/r)*D(y/r)/D(z)]

All Rnk, Qnk and it's derivatives calculates recursively, from pages 91-92 formulas to get partial derivatives for D(1/r)/D(x) and etc. At the end brackets was simplified for each partial derivative values fm * x/r**3, fm * y/r**3, fm*z/r**3. J2 constant really bigger 10 times for 75 degree, in Fx,Fy presents sin(co-latitude), in Fz presents cos(latitude)**2 / sin(latitude), and error drops from 30km to 685m.

Impulses was implemented in tra.cpp as a graph from engine thrust's tests, [i.e. http://www.canadianrocketry.org/motor_files/37148-O4900-BS-P.pdf] . The data was to be entered in a XML format:

```
<TRA:section name="Engine" value="0.0" >
   <TRA:setting name="EngineNumber" value="0" />
   <TRA:setting name="EngineOnSatellite" value="0" />
   <!-- convinent coeff - instead of entry real values just assume scaled version-->
    <TRA:setting name="PropCoeff" value="0.05" />
    <TRA:setting name="Weight" value="17.157" />
    <TRA:setting name="TotalWeight" value="21.996" />
   <!-- iteration per sec from engine's plot -->
   <TRA:setting name="DeltaT" value="5" />
    <!-- 2- EARTH 9-MOON for calculation distanses-->
```

```
 <TRA:setting name="NearBody" value="2" />
 <!-- calculates
     0 Perigee to a center of NearBody
     1 Apogee to a center of NearBody
     3 taget practice not far then distance earth-moon
     4 target practice to apoint on a moon's surface
     8 intial period -->
 <TRA:setting name="Calculate" value="8" />
 <!-- AngleType 0 - tangent line to orbit (elipse) oposit velocity
             1 - two angles set with reference to NearBody centre direction
             2 - 3 angles set vector fire (constant all fire)
             3 - oposit vector of velocity
             4 - same direction as vector of velocity -->
 <TRA:setting name="AngleType" value="0" />
 <!-- 2- EARTH 9-MOON for firing angle -->
 <TRA:setting name="AngleOnBody" value="2" />
<!-- first angle: in a plane over vector from the center of NearBody and Sat
    and vector of velocity. Angle: Centre,Sat,Direction
    (aggle == 90 degr is a Tangent line to elipse) -->
    <TRA:setting name="FireAng1" value="1" />
 <!-- second angle from projection of a velocity vector to a
    plane perpendicular to direction to centre of nearbody -->
<TRA:setting name="FireAng2" value="1" />
<TRA:setting name="XVector" value="-1" />
<TRA:setting name="YVector" value="1.1537307924011437" />
<TRA:setting name="ZVector" value="0.9986199954416527" />
 <TRA:setting name="OptimizationInitialStep" value="10" />
 <TRA:setting name="OptimizationDecCoef" value="10" />
 <TRA:setting name="OptimizationStop" value="1" />
   <TRA:setting name="Period" value="5494.000000" />
<!-- set impulses in a time -->
   <TRA:setting name="FireTime" value="11944" />
 <TRA:setting name="ImplVal" value="0.0" />
 <TRA:setting name="ImplVal" value="3400.0" />
 <TRA:setting name="ImplVal" value="3000.0" />
 <TRA:setting name="ImplVal" value="3100.0" />
 <TRA:setting name="ImplVal" value="3500.0" />
```

<TRA:setting name="ImplVal" value="3550.0" />

<TRA:setting name="ImplVal" value="3600.0" />

<TRA:setting name="ImplVal" value="3600.0" />

<TRA:setting name="ImplVal" value="3700.0" />

<TRA:setting name="ImplVal" value="3750.0" />

<TRA:setting name="ImplVal" value="3800.0" />

<TRA:setting name="ImplVal" value="3850.0" />

<TRA:setting name="ImplVal" value="3900.0" />

<TRA:setting name="ImplVal" value="3950.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="4000.0" />

<TRA:setting name="ImplVal" value="3980.0" />

<TRA:setting name="ImplVal" value="3960.0" />

<TRA:setting name="ImplVal" value="3940.0" />

<TRA:setting name="ImplVal" value="3920.0" />

<TRA:setting name="ImplVal" value="3900.0" />

<TRA:setting name="ImplVal" value="3885.0" />

<TRA:setting name="ImplVal" value="3850.0" />

<TRA:setting name="ImplVal" value="3815.0" />

<TRA:setting name="ImplVal" value="3780.0" />

<TRA:setting name="ImplVal" value="3745.0" />

<TRA:setting name="ImplVal" value="3710.0" />

<TRA:setting name="ImplVal" value="3695.0" />

<TRA:setting name="ImplVal" value="3680.0" />

<TRA:setting name="ImplVal" value="3665.0" />

<TRA:setting name="ImplVal" value="3650.0" />

<TRA:setting name="ImplVal" value="3600.0" />

<TRA:setting name="ImplVal" value="3500.0" />

<TRA:setting name="ImplVal" value="3500.0" />

<TRA:setting name="ImplVal" value="3400.0" />

<TRA:setting name="ImplVal" value="3150.0" />

<TRA:setting name="ImplVal" value="2000.0" />

<TRA:setting name="ImplVal" value="250.0" />

<TRA:setting name="ImplVal" value="0.0" />

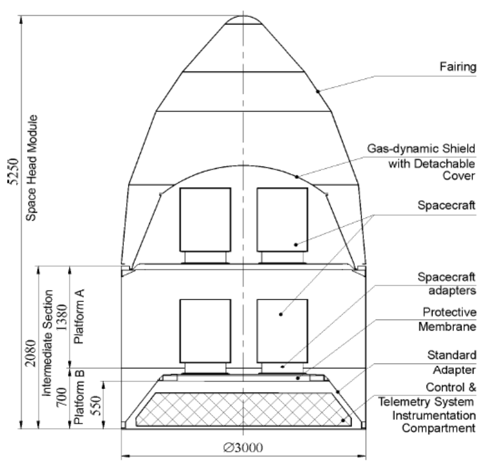
</TRA:section>

In trajectory calculation software for each satellite can be set different engines. It does not seems essential for a mission, but it was interesting in orbit calculation study to understand how will be possible to match speed and orbit of flying simultaneously by two satellites. For a convenience it is possible to use a parameter PropCoeff to adjust the impulse instead of re-entry different one. For sure a real test's data of a real engine could be use – but for an approximation and exercise it is ok. Also PropCoeff can be used to simulate real engine firing – satellite can be rotated and vector thrust can be controlled by rotation (траектория конуса), in that case impulse direction can be reduced to by 0-5% - and PropCoeff with corresponded value will be a good approximation for that firing.

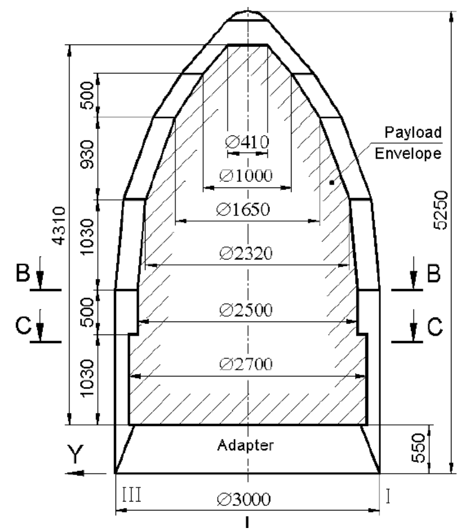
2.6 Launch Vehicle Details

Three launch vehicles available today on a market was considered as a possible launch vehicle. Below are pictures of payload envelopes of all of them.

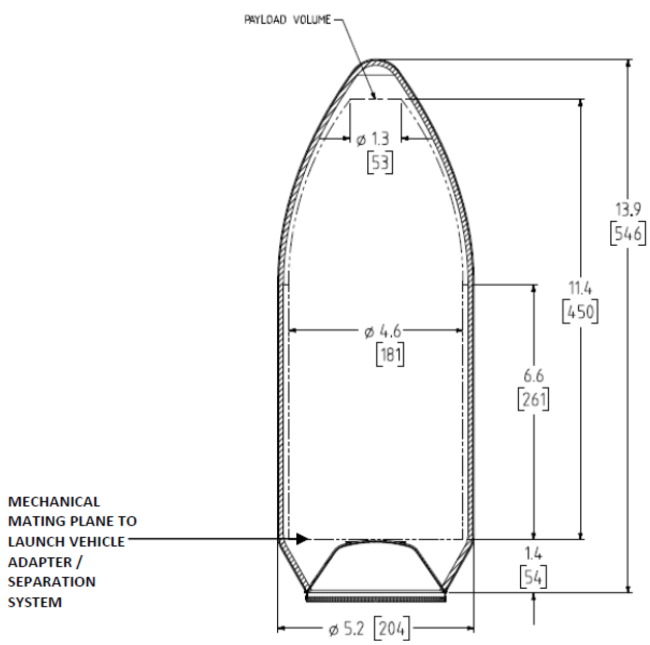
Pic 2.6.1 Dnepr payload bay



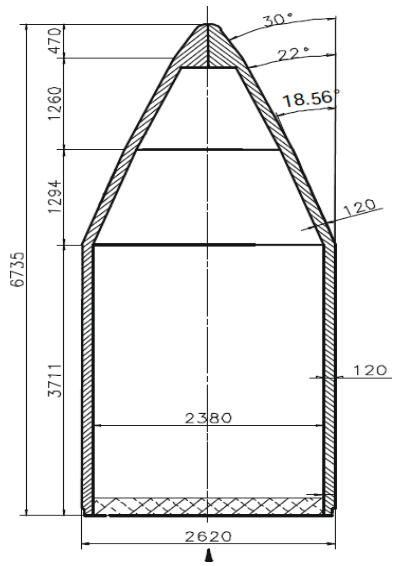
Pic 2.6.2 payload envelope



Pic 2.6.2 Falcon 9



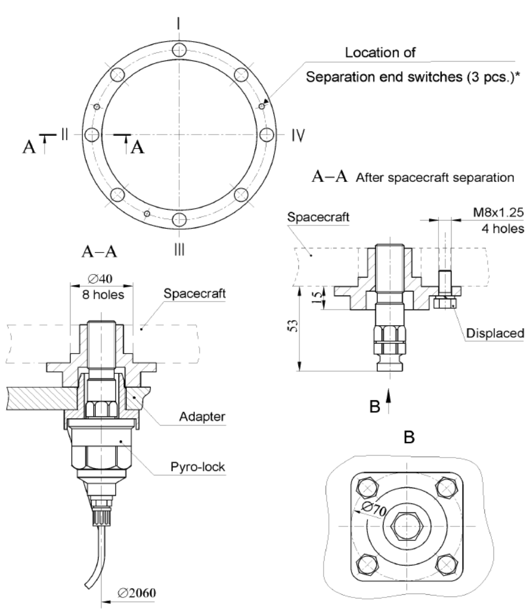
Pic 2.6.3 Rokot



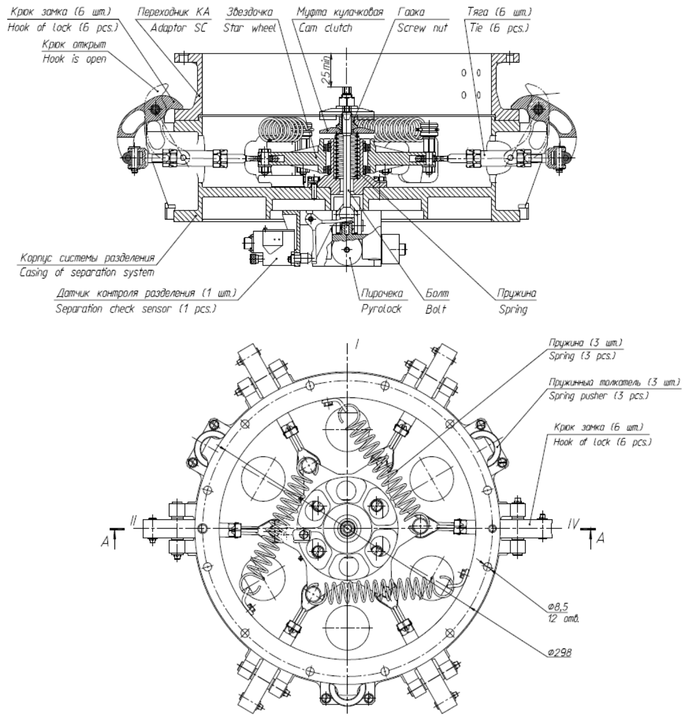
Launch vehicle will be connected with craft via Payload Adapter, Mechanical Lock System (MLS), Minisatellite Separation System (MSS), or Pyro-lock (without springs on Dnepr).

Adapters for connection to launch vehicle can be one of those types:

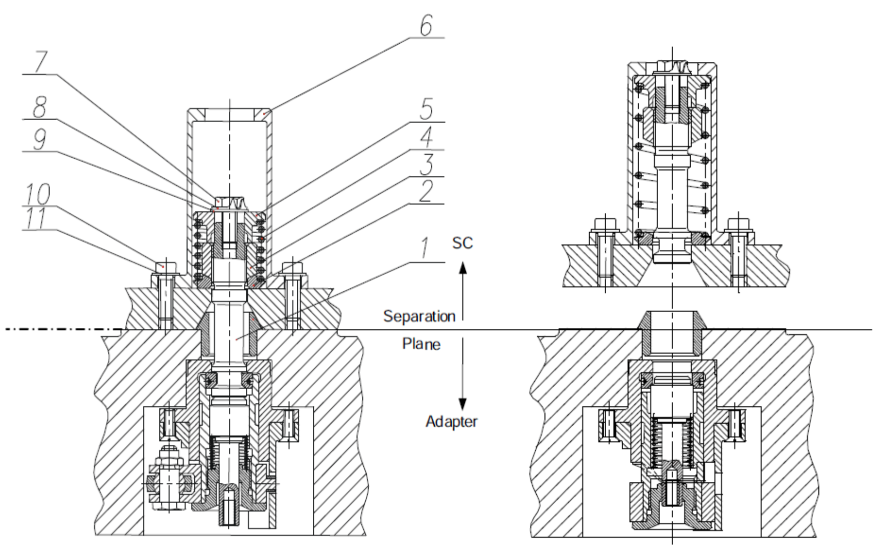
Pic 2.6.4 Dnepr payload adapter rings with pyrolocks



pic 2.6.5 Rokot Mini satellite Adapter System

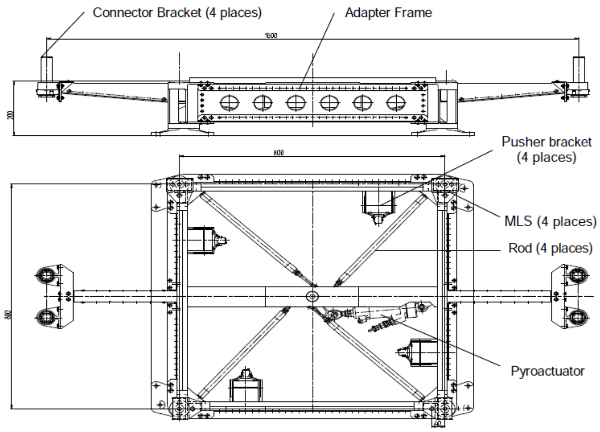


pic 2.6.6 Rokot's Mechanical Lock System



Note: The indicated parts remain on the spacecraft after separation: 1,7,10 = Bolt, 2,8,9,11 = Washer, 3 = Screw-nut, 4 = Spring, 5 =Support, 6 = Bolt Retainer.

Pic 2.6.7 Rokot's Mechanical System Payload Adapter



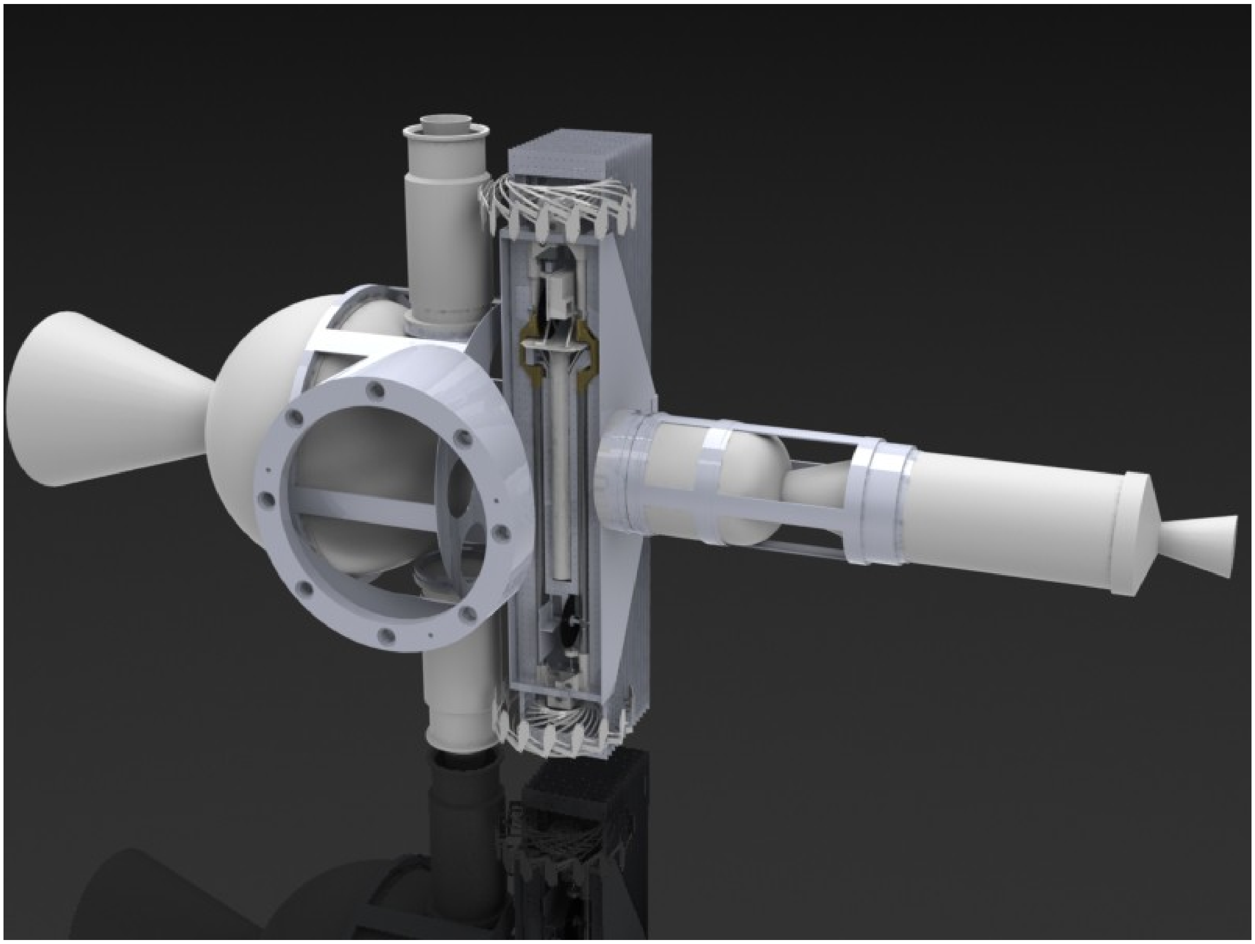
The craft will be a polygonal aluminum/carbon fiber frame mounted on the top of the rover frame. In table 1.1 in mission description appears weight properties of each fixed impulse engine. To minimize development was decided to take from available "from" shelf matching ATK engines. Table 2.6.8 consists the match for it:

table 2.6.8 List of engines from ATK catalog with matching parameters require for mission

Orbit correction	Catalog number	total mass + frame	propellant mass	Total impulse lbf sec	Diameter	
LEO correction 1	TE-M-541-3	6.0625 kg	4.85 kg	3077	0.16m	
LEO correction 2	TE-M-541-3	6.0625 kg	4.85 kg	3077	0.16m	
Main burn	TE-M-604-2	194.15kg	178.62kg	112400	0.63m	
correction 3	TE-M-1076-1	17.32kg	12.3kg	7430	0.21m	
break impulse	TE-M-956-2 (or TE-M-236)	18.5kg (27.6)	14.42kg (18.24)	9212 (10350)	0.23m(0.31)	
frame total		24.20kg				
TOTAL		272.3kg (282.4)			Max = 0.63m	
adapter/MSL		10kg				

Based on available drowning that makes craft as a combination of the frames and engines:

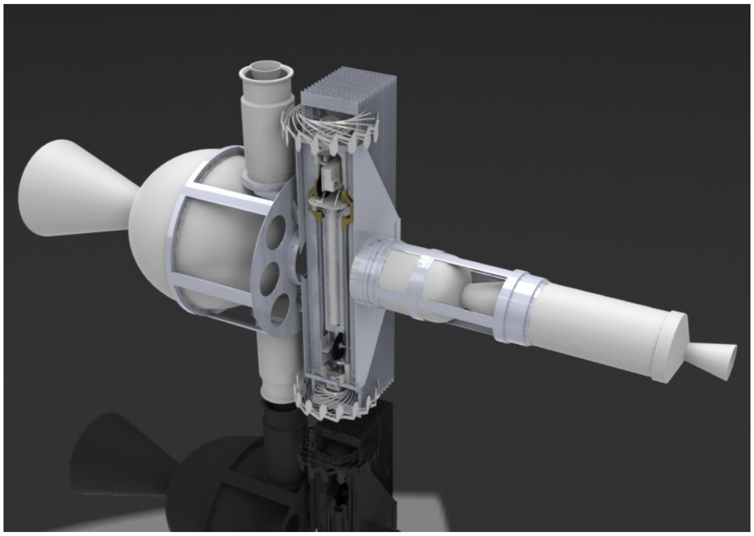
Pic 2.6.8.9 Craft with payload adapter



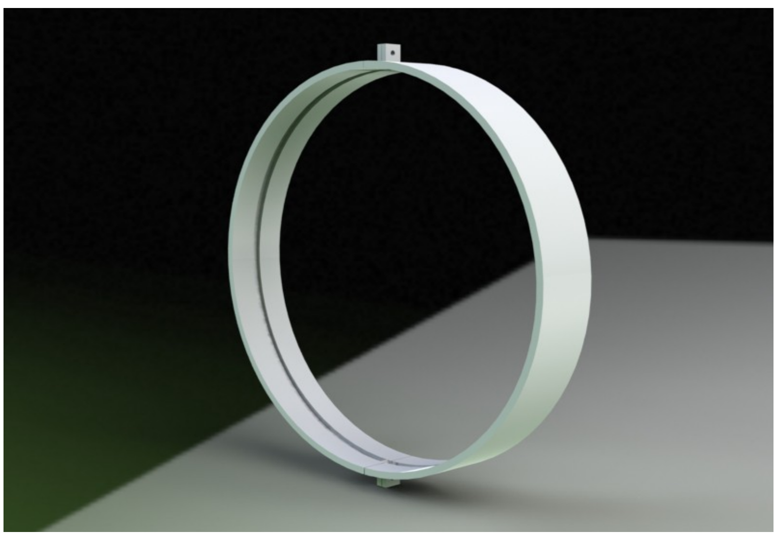
272g-282kg is not good result basically because of engines TE-M-604-2 (main impulse) and brake engine TE-M-956. Main impulse engine should be 10-15% less, with TE-M-604-2 engine trajectory will be less restrictive on accountancy of the sun/moon gravity, but that add 20-30kg on LEO. At the same time break engine needs to be 15-20% bigger.

2.7 Baseline System Design

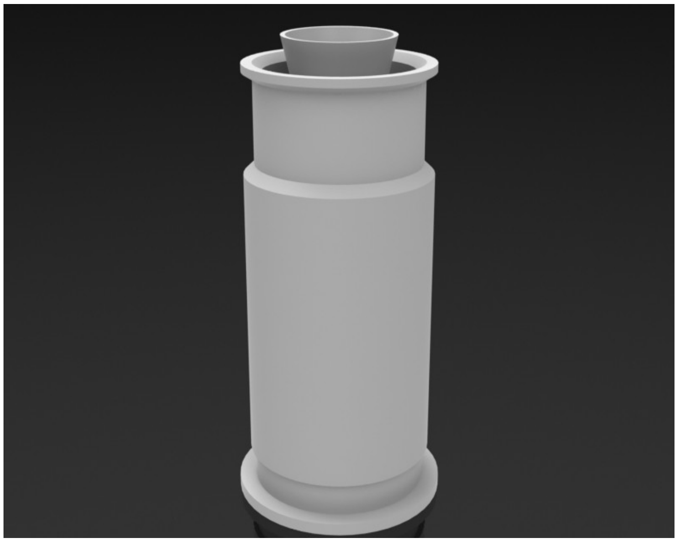
Pic 2.7.1 Frame of the craft with fixed impulses engines.



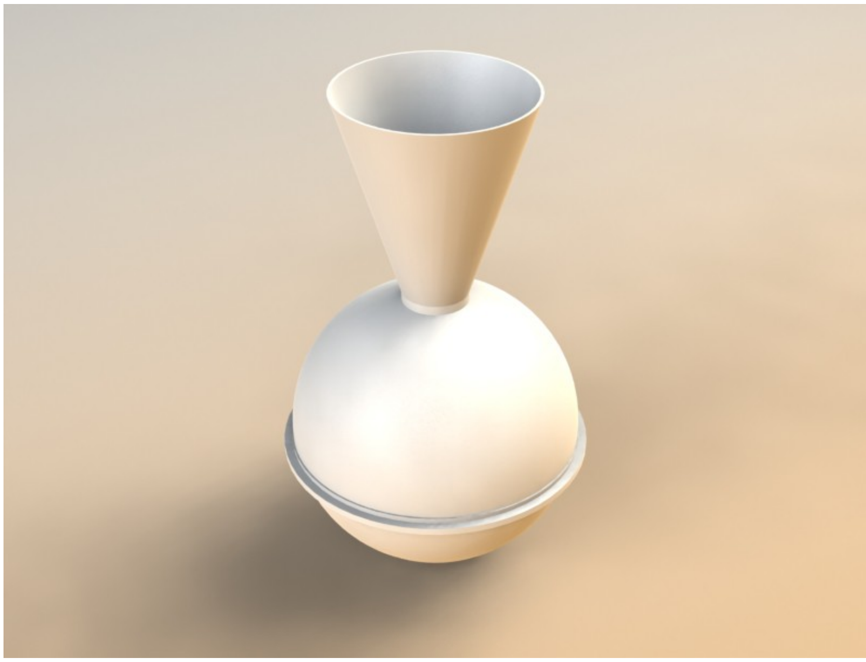
Pic 2.7.2 Engines is mounted on a frame by pyro-lock.



Pic 2.7.3 LEO correction 1 engine



Pic 2.7.5 Main impulse engine.



Pic 2.7.6 Orbit correction 3, engine.

Pic 2.7.8 Brake engine

Attitude control system

Attitude control consists of gyro-platform, rotation mechanism, and microcontrollers processing all data from sensors, and movements/rotations performed by rotational mechanism.

The following rotational mechanism is used for the landing of the craft. Rotation mechanisms use 3 devices. First - located in nozzle of the brake impulse engine. That device is mounted to the frame stepper motor, with the shaft lying on the line crossing center of the craft. A cylindrical object with the mass of 0.185g and the dimensions 25.4mm x 76.2mm is mounted to that shaft. The other two devices are used to control attitude of the craft are wheels of the rover. These wheels rotate freely; the superposition of the movement of all 3 devices allow for rotation momentum applied to the craft. The device mounted inside is the brake impulse engine it is responsible for the rotation. The speed of the rotation is 5 rounds per second for main impulse burn and for the break impulse. That device is ejected after achieving a rotational speed of 30 rpm before ignition of a brake engine. After the ejection, wheels of the rover will not be used for such attitude control. Instead they will be used for their prime purpose.

Gyro-platform is a two solid state gryro sensors mounted on a frame of the rover, 1 solid state gyro sensor is mounted on antenna mount, 1 solid state gyro sensor mounted on a low resolution camera stand mount, 1 accelerometer mounted on a frame of the rover, 1 accelerometer mounted on antennas mount, 1 accelerometer on low resolution camera stand, one magnetometer mounted on frame of the rover.

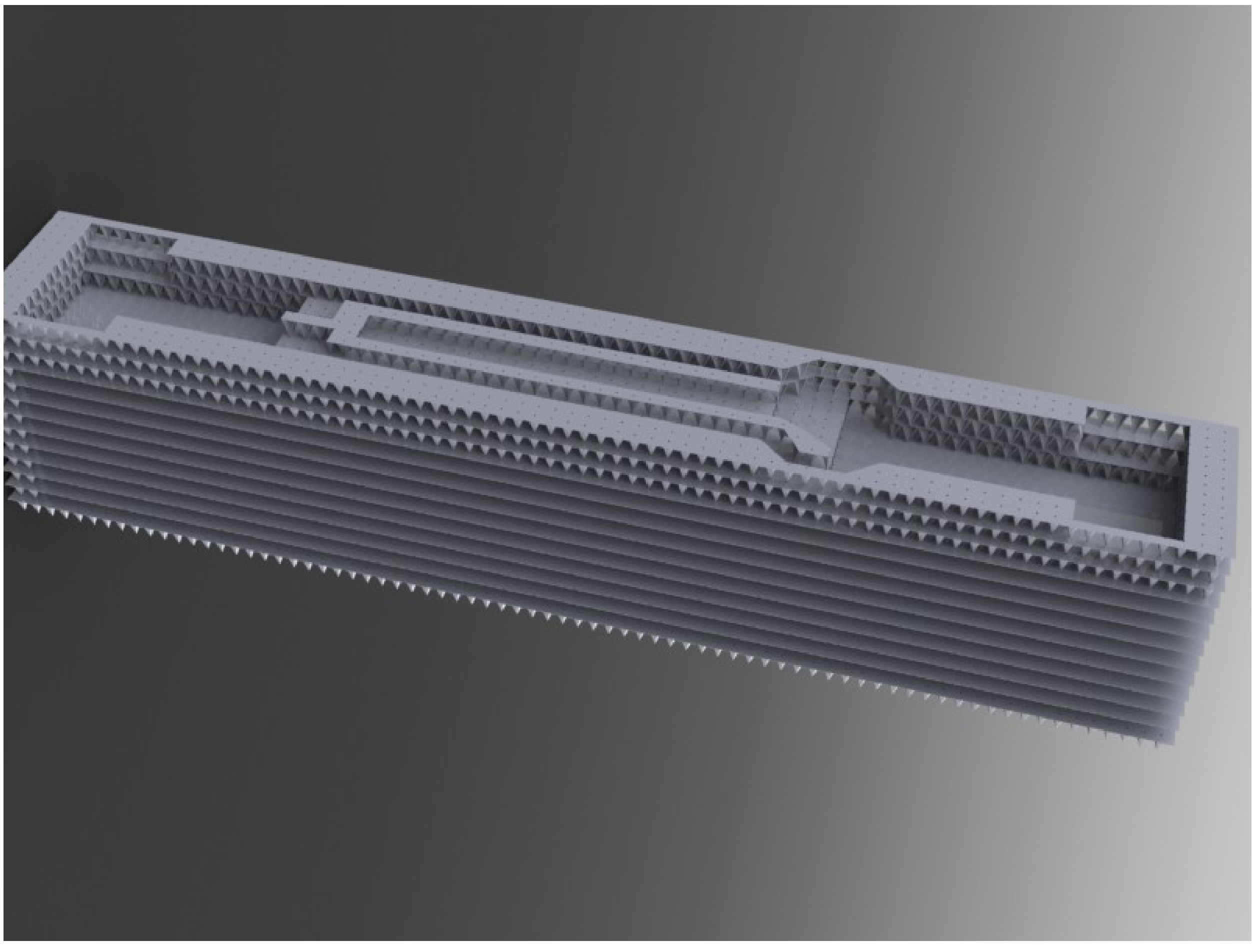
Processing all the data from 7 sensors is done by 2 microprocessors located on two independent PCB board.

Attitude control controls antenna direction for communication session for the craft. It is done by separate stepper motor mounted on the antenna stand of the rover. For "on-orbit" configuration, an antenna is set to a fixed position that orient's it to the ground station. This is performed by the rotation of the full craft. Nano-satellite antenna points to a fixed direction, all nono-satellite need to orient itself for communication session.

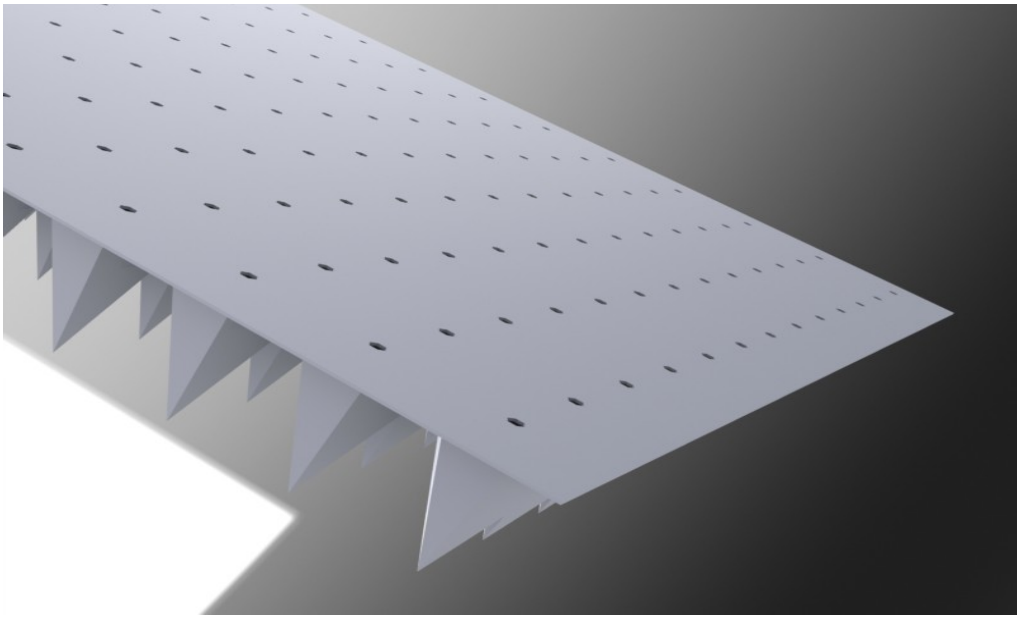
Attitude control also controls two low resolution camera fixtures on the rover. Low resolution pictures can be taken on "on-orbit" mode. In "on-orbit' mode the stand fix is opposite of the antenna direction. That is done to reduce the interference from electronics of the antenna and low the resolution cameras. On the nano-satellite, another low resolution camera is fixed. Attitude controls forces the rotation of all nano-satellite to make the desired orientation to take a picture.

Impact adsorption shield

Pic 2.7.9 All assembly from 13 individual shields



pic 2.7.10 one layer of impact shield

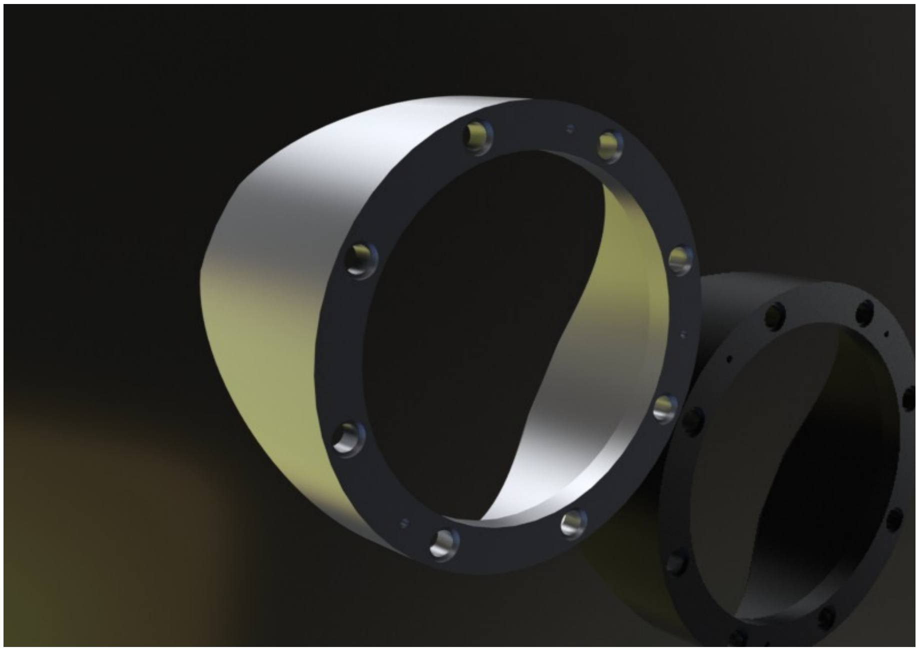


Impact shield consists from 13 layers of a carbon fiber sheets. Each sheet is designed to be crushed against each other and destroy the next layer. Destruction is supposed to suppress the kinetics energy of failing to the lunar surface rover, and reduce speed from worst case 35m/sec to 0m/sec. This makes acceleration on impact equal to 364g in bad case of a fall from 362m of the lunar surface. Descending with the speed of 25 m/s in a case of Luna-9 was around 260g on impact. Study performed in 2011 showed impact with a speed from 9-12m/sec makes the acceleration from 30 to 40g on impact.

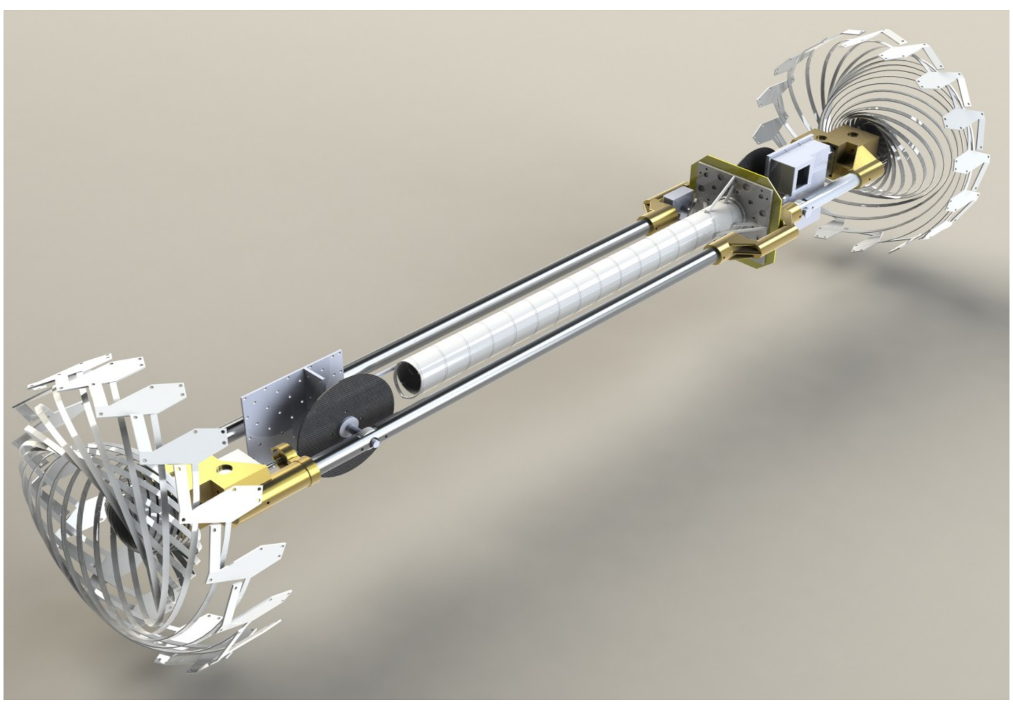
Each layer of impact shield connects to each other and to the rover by the carbon fiber bridge. Impact destroys the bridges which separates the rover from the impact shield.

To compensate for the rotation of the rover and impact shield used carbon fiber spring designed to give rotation moment in time of separation of the brake engine frame and rover. Spring is mounted on the engine and will apply rotation momentum on the impact shield and rover.

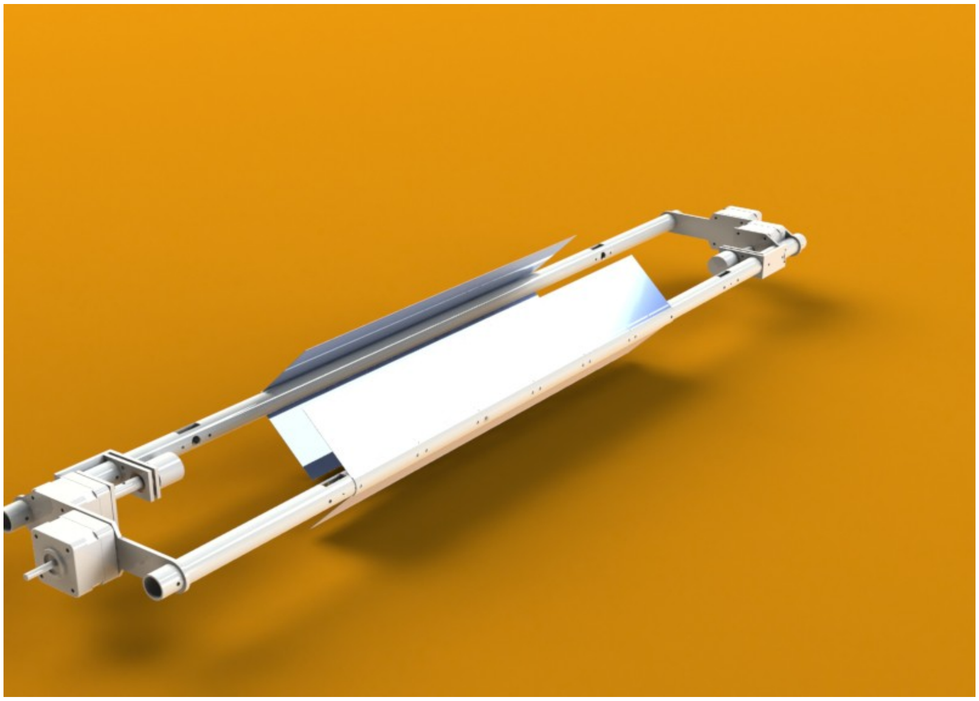
Pic 2.7.10 frame connector for secondary payload adapter



Pic 2.7.12 Rover/ Ground station



Pic 2.7.12 Rover. Flexible solar panel placement



Mobility subsystem includes rover weight of 4 kg, Consists of a frame made from carbon fiber. Two wheels mounted on a stepper motors. Antenna mount. Low resolution cameras box. Stand for low resolution camera box. Sealed HD camera box. Counterweight compartment for low resolution camera stand. Symmetrical gears for antenna, and low resolution camera's box. Two additional stepper motors to support movements of a antenna and low resolution camera stand. Gyro-platform with 2 gyro-sensors mounted inside frame, 2 accelerometers, and magnetometer. Additional gyro, and accelerometer sensors mounted on a antenna mount. Additional gyro and accelerometer mounted on low resolution cameras stand. Solar panels mounted on frame. Power energy storage capacitors inside tubes of the frame. Craft avionics inside HD camera box. Solar sensors mounted on a frame. Communication system electronics on reflector of the antenna. 3D printing test equipment for technological experiments on lunar surface includes lens mounted on HD camera box and indium metal dispensing holder.

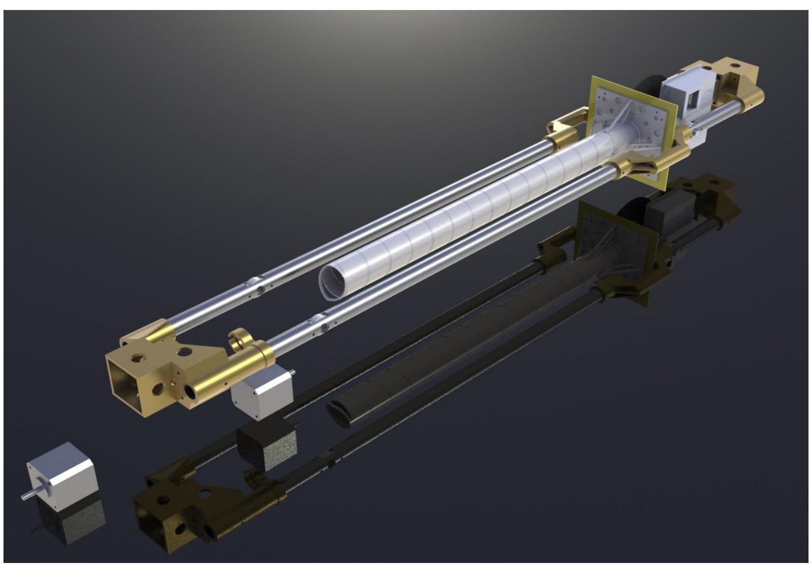
Mobility of the system supported by two wheels powered by two stepper motors and low resolution camera stand used as sliding leg in 45 degree rotated position. For guidance used a gyro-platform, allowing to keep desired direction. By retracting antenna and leg/camera's stand mobility of a rover converted from 3 touching surface points of the rover to 2 wheel configurations. That allow fast travel down on slop of carter to get as much as possible imaging (HD video) information of a geological observation.

At stops low resolution camera stand moves to a position with best observation to take pictures. Pictures can be transferred at stops in communication session performed by orienting antenna to the earth by gyro-platform based on solar sensor detected sun's direction and center of the moon direction detected by accelerometer of gyro-platform.

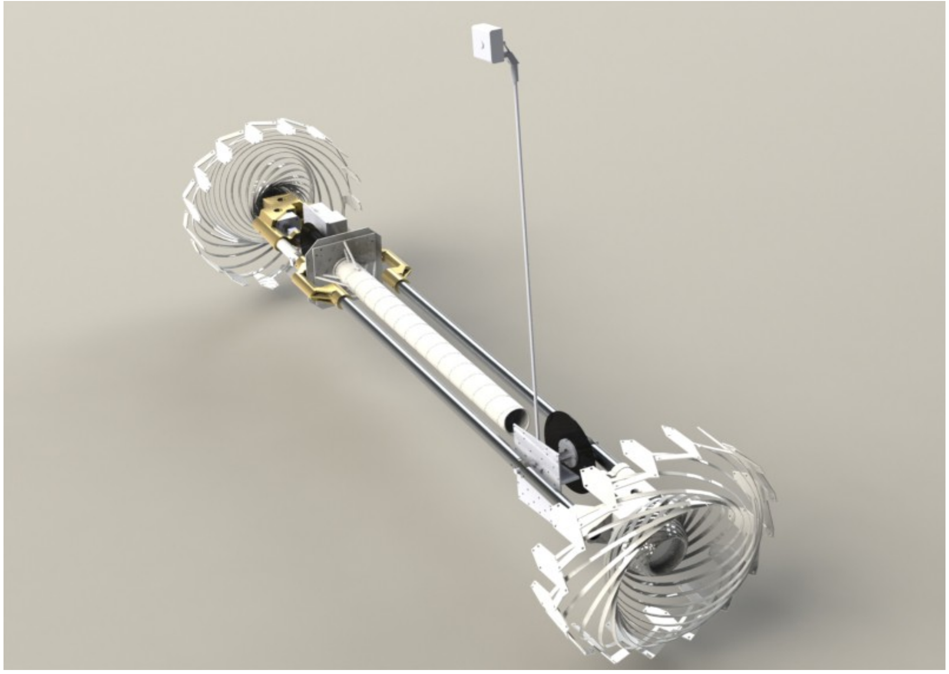
Mobility session of the rover is separate from a commination sessions, autonomous, and accepts command from a mission control to support movement in desired direction, with desired distance.

Wheels of rover have springs to allow accumulate elastic potential energy of springs in motion's movements. Harmonic oscillation of individual springs holding loads of the rover will be detected by gyro-platform and can be used to achieve desired direction move on lunar terrain. On tip of spring mounted rims with flat surface. Surface of the spring's rim used to form a molding cavity for 3D printing technological experiment. There are 16 spring in each wheel, which makes possible to form 32 possible molds, or messages can be printed on a lunar surface.

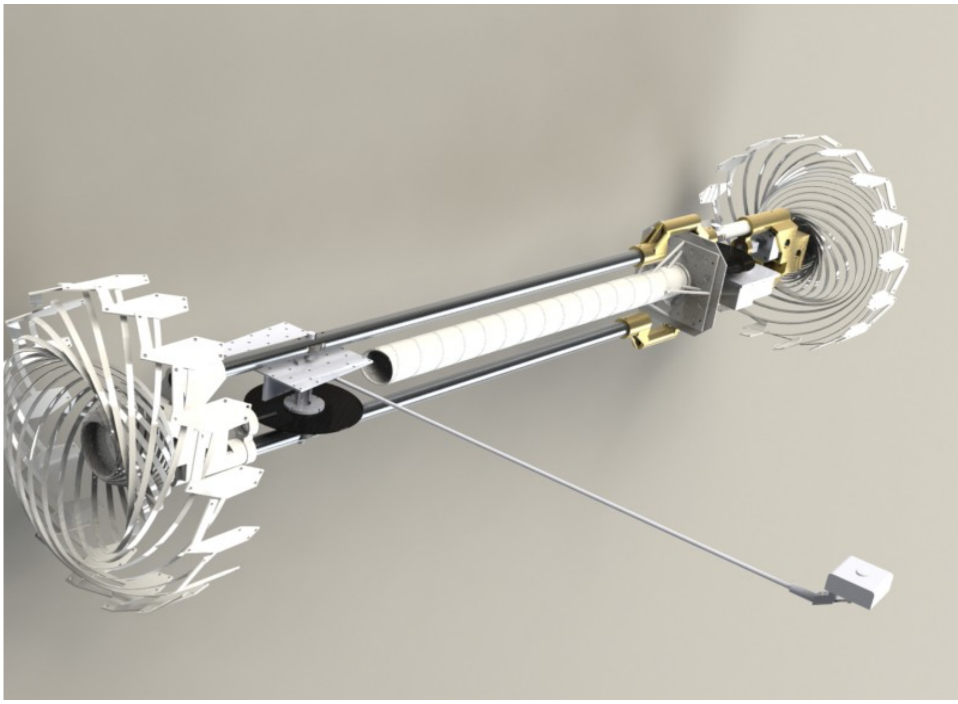
Pic 2.7.13 Main communication antenna



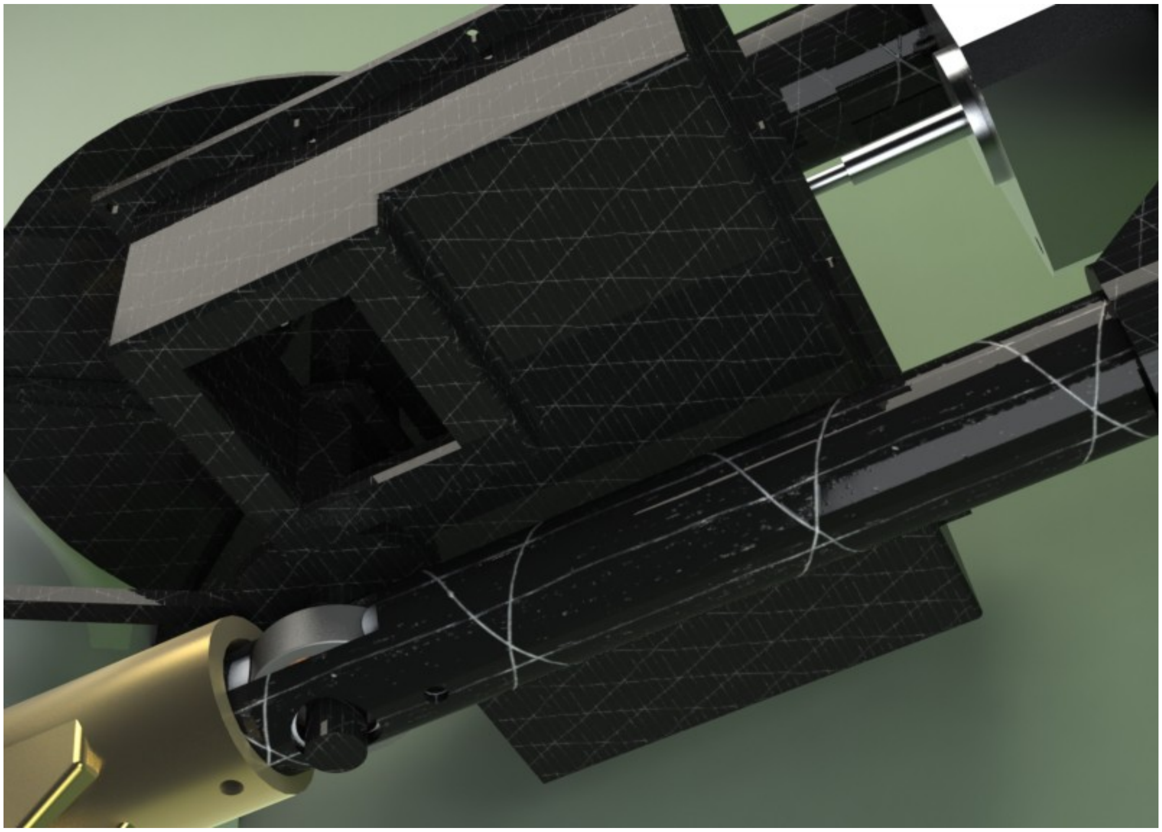
Pic 2.7.15 retractable mechanism for camera stand



Pic 2.7.16 mobility platform with 2 wheels



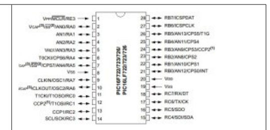
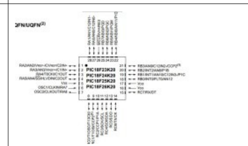
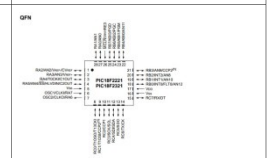
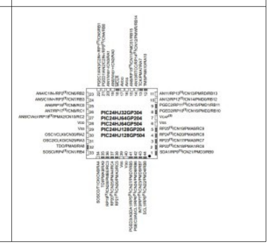
Pic 2.7.17 HD camera mounted on pointing mechanism for main antenna



Electronics.

For core development was chosen PIC family of micro-controllers. Such devices was tested and previously used for space flight on nano-satellite type missions. Micro-controllers allows to reduce power consumption to a non-watt level, support industry standards for communication interface like serial, SPI, I2C. Interrupts level and internal futures of onboard allows to build robust system. Compiles are not best but allows with some software tricks to have a portable source code not only for different type of micro-controllers inside the family, but for different type of micro-controllers. Another group of electronics is the sensors.

Micro-controllers:

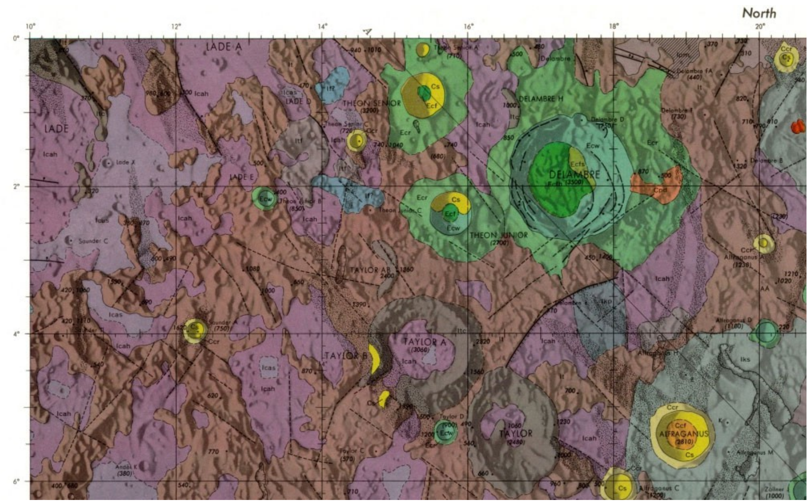
PIC16LF724-E/P	PIC® XLP™ 16F,20MHz,I²C, SPI, UART/USART,7KB (4K x 14),NO EEPROM,192 x 8,power=1.8 V ~ 3.6 V,A/D 14x8b	
PIC18F25K20-E/ML	PIC® XLP™ 18F,48MHz,I²C, SPI, UART/USART,32KB (16K x 16),eeprom=256 x 8,1.5K x 8,power=1.8 V ~ 3.6 V,A/D 11x10b	
PIC18LF2321-I/ML	PIC® 18F, 8-Bit, 40MHz, I²C, SPI, UART/USART, i/o=25, 8KB (4K x 16), eeprom=256 x 8, 512 x 8, 2 V ~ 5.5 V, A/D 10x10b,	
PIC24HJ64GP504-E/PT	PIC® 24H,40 MIPs,CAN, I²C, IrDA, LIN, PMP, SPI, UART/USART,64KB (22K x 24),8K x 8,power=3 V ~ 3.6 V,A/D 13x10b/12b	

All lists of used electronics can be found in a DB on- http://24.84.38.192/dbadobri/List.aspx

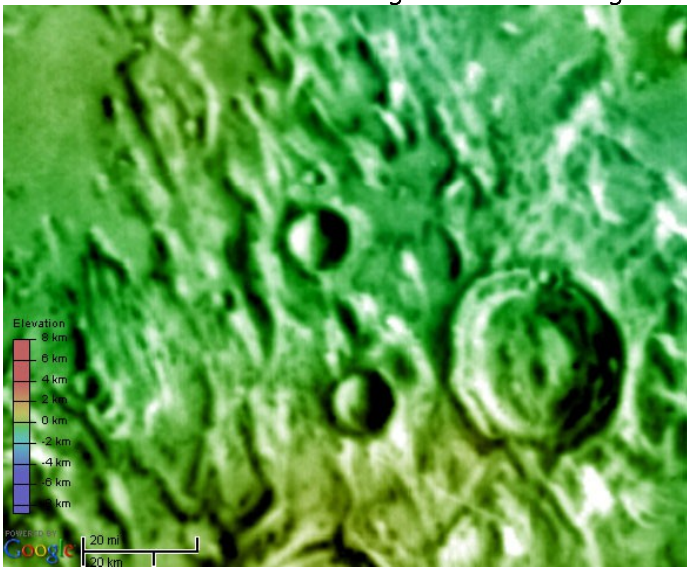
2.8. Lunar activity top level design.

Planned landing point is S2E15. That area is in highland area with steady inclination. Landing 1km from any point around it will give steady sloops to travel required 500m in any directions.

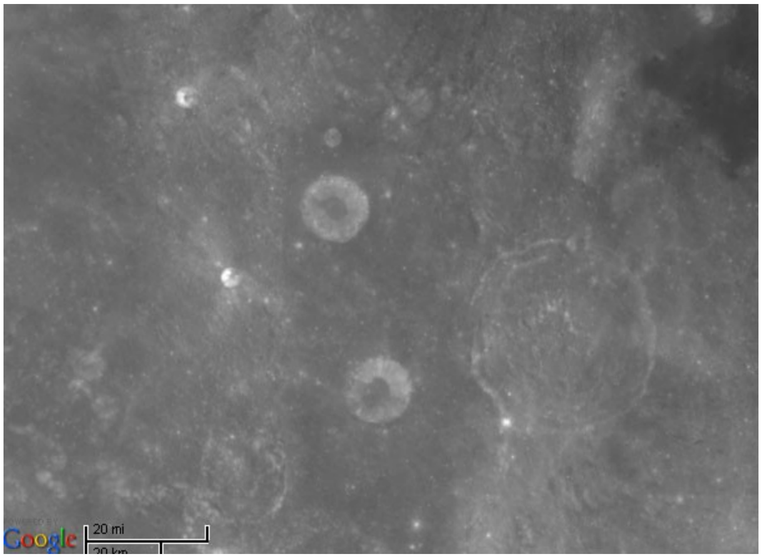
Pic. 2.8.1 Landing area S2-E15



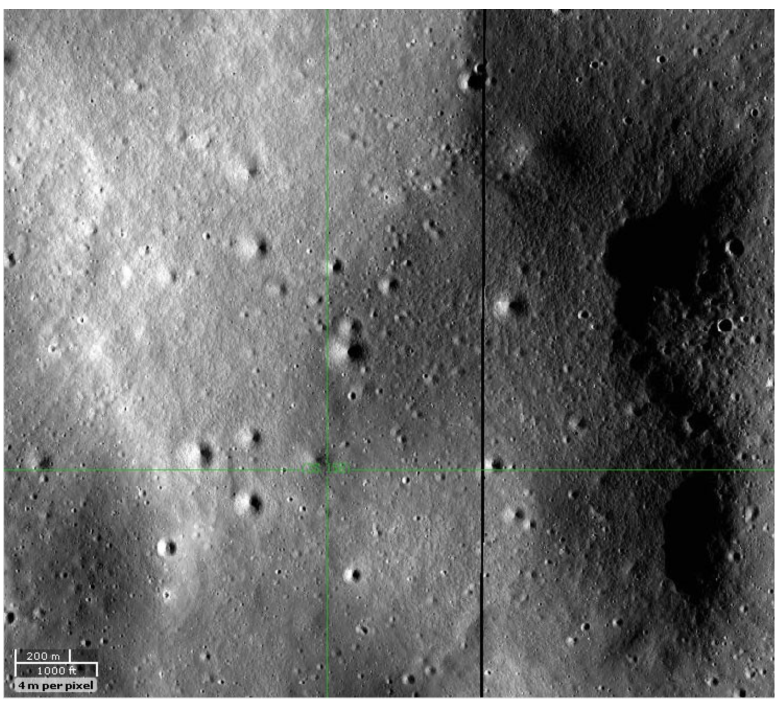
Pic 2.8.2 elevation in landing area from Google map



pic 2.8.3 image of landing areas from Google map



Pic 2.8.4 main landing area from LRO/LROC

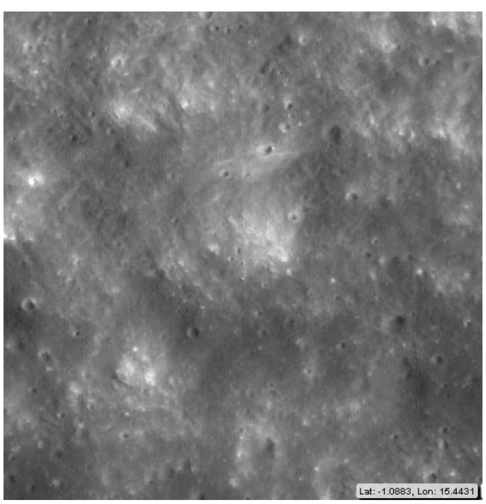


30km on east from the planned landing point there are two interesting sites Theon Junior, and Theon Sinior. Both landmarks are around 18-19km in diameter, with rim to floor difference of 3km and length of the slope around 5 km. In that case choosing landing point on the rim of Theon Junior S2.1312E15.7745 mission will have ability to travel distance 4-5 km in short period of time (8 minutes) and HD video can give detailed pictures of a geological structures on all way from the rim to the floor.

pic 2.8.5 second landing area

Shifting the landing point 30km to NE also desirable. Pictures of high resolution slope of the Theon Senior available from LRO/LROC. It shows rough slope. Landing on the tip of the rim will give ability to travel 5-7 km with short period of time, less energy, and through 3 km of sliced geological structure.

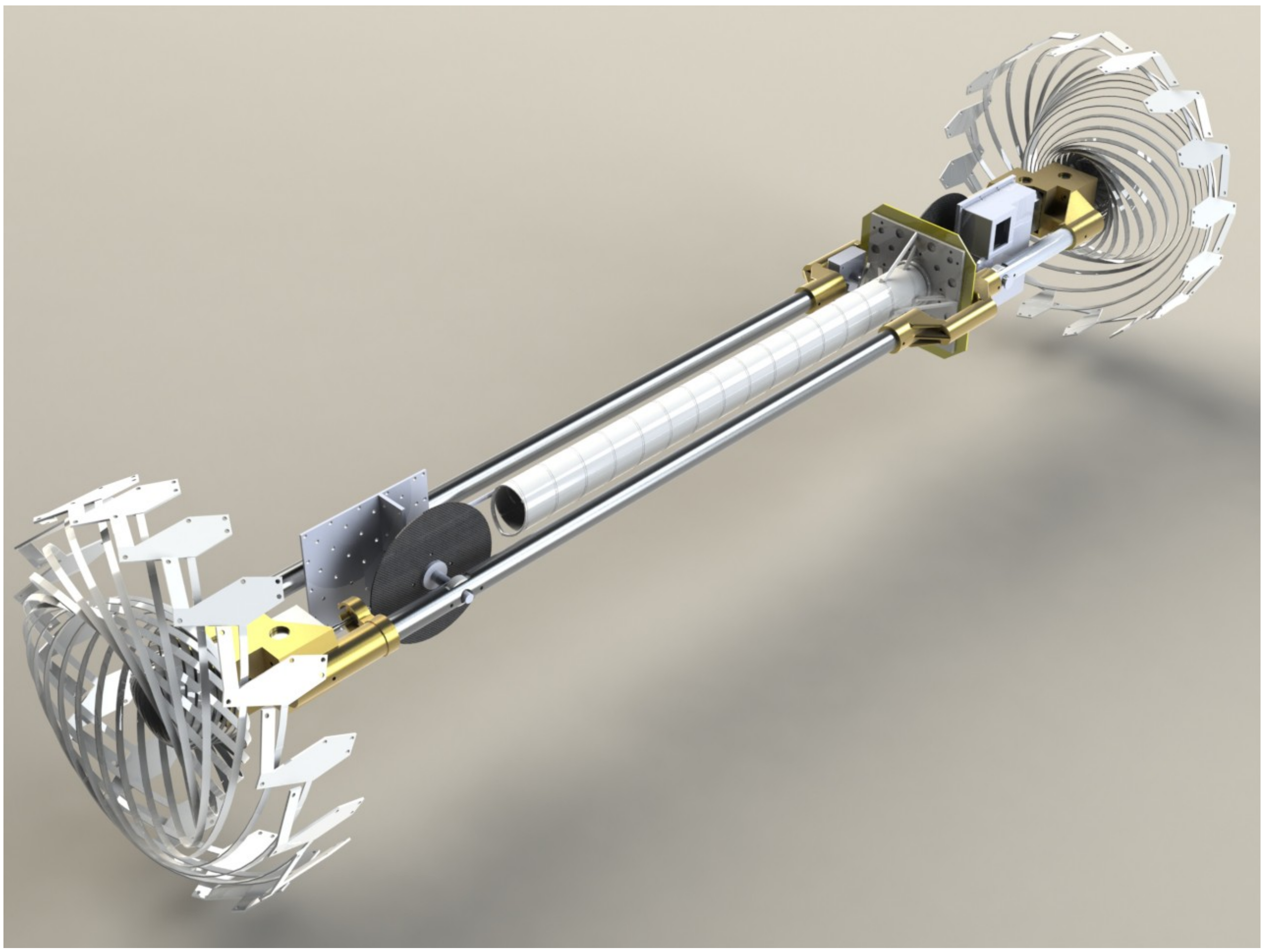
pic 2.8.6 third preferable landing area



Selection of landing area can be done by correction on trans lunar trajectory, ether it will be a safe attempt to win competition, or risky (50/50 chances to get correct point on tip of the rim) attempt to receive required by MTA HD video with valuable information to analyze.

3.0 Rover design

pic 3.0.1 rover version 3.0



Rover consists of a 4 stepper motors mounted on carbon fiber frame. Two stepper motors controls two wheels made form carbon fiber. Two additional stepper motors controls antenna mount and stand with two low resolution cameras. On a frame mounted flexible solar panels. HD resolution camera sealed inside a box mounted on an antenna mount. High capacity capacitors as power storage sealed inside carbon fiber frame of the rover. Another box with power storage capacitors is located on mount of cameras. Communication system electronics mount on antenna reflector. Gyro system for movement includes 3 tri-axes gyro-sensors and 3 tri-axes accelerometers. Each pair of gyro & accelerometer is located on antenna mount; frame of the rover, and on low resolution camera's stand. Main computer located inside sealed HD camera box. Gear for rotating both mounts (antenna and camera stands) are symmetrical. Gears allow rotating antenna's mount and pointing antenna to desired place on the sky. Movements of the rover and turning frame to desired direction, together with antenna's gear allow orienting antenna to the earth from moon. Additional degree of freedom allows choosing more stable position of a rover in communication session. Angle of the earth on a lunar sky depended on a latitude and longitude. Extends of a HD camera box (length from rotation axe of the antenna mount to the tip of the HD cameras box) allows to touch the lunar soil in a position of a communication session. That allows keeping temperature of a sealed box and lunar soil in balance. That also allows saving energy required by antenna mount stepper motor in time of communication session.

For travel antenna mount located in "retracted" position. HD camera box allow to record video during travel. Command from mission control set direction of movement. From that moment of time low resolution camera's stand moved into position of "45 degree". That allows using stand as additional "leg"/"slide". Stepper motors tried all possible movement and first tried to move rotation frame to desired direction used to orient rover on lunar surface. Then rover made attempts to travel and accelerator checks for movements was successful or not. Algorithms to move around obstacles based on data collected by accelerometer. If for some period of a time accelerometer did not calculated that rover traveled in decided direction that will be made attempt to bypass obstacle by avoiding route. That includes full stop, rotate low resolution camera into a "capture images" position, make photo in "stuck" position, travel back direction for 2 m, take a picture, then turn left 90 degree, move 5 m left, take a picture, continue move to direction set from mission control. Last on that route will be right turn and then travel 5 m back to required point. Each time on turning point low resolution camera makes reconescens picture. Mission control can analyze pictures and instead of one command to travel to some direction (i.e. on tip of rim of the crater command can set to travel all way down to the floor of the crater, which can be 5 km). Accelerometer used inertial calculation to roughly calculate performed movements. When traveling (target) point was reached, or when after timeout rover stops rover performs sequence of movement to orient antenna to an earth. In communication session picture and recorded accelerometers data can be retrieved by commands from mission control. At the same time of the session HD video can be transferred by chunks.

Power system before any movements calculates power required for such move. It is done by accounting previous movements performed. If power storage considered not enough for performing such move it waits for solar panels to charge storage capacitors. When target point or timeout reached in a sequence of an orientation antenna to the earth, the same procedure to calculates power used based orientation requires mechanical moves. Then when orientation of an antenna is done, it needs to have enough power to support communication session. To do this in communication session embedded data with a level of a stored power. That will give mission control 1.5 second delayed data about session sustainability. If power is not enough then session interrupted for a required period of time and continue until all data will be delivered to mission control, and rover will wait next command to perform. Thermal control of the rover monitors temperature measurements to estimate that move/action/communication is possible.

4.0 Power Equipment List

It was considered that only solar panels capable to withstand landing impact are flexible solar panels. In 3 years of a project it was tried different type of solar panels and was found that each half year new solar harvesting devices and power equipment for harvesting become available. Mounting of solar panels assumed all over the frame of rover. Performance of the solar panels will degrade after trans lunar flight. Assumption was made that on a lunar surface it can be reduced to a level of 3Wt and low. To accommodate all movements and communication session account harvested energy stored in high volume capacitors.

Stepper motors for a mobility/attitude control 8Wt

Gyro-platform 0.12Wt

Communication 25Wt pick power

Electronics 0.001Wt-0.2Wt (depend on a mode of operation controllable by a software)

4.1 System Level Summary

Craft consists of such system

- attitude control weight 2kg, with a task to perform orientation of the craft, attitude control system ejected before brake engine burn after performing stabilization rotation.

- Communication system, works on 2.4GHz on "hopping" frequencies. Its transmitter is 10Wt transmitting power with PAE 40%, that require 24Wt for pick transmitting, proprietary protocol of transferring data allow to send data using one physical transmitter over 3 different channels. Automatic system adjusts drifting frequencies because of temperature drift. System allows restoring broken packets with 10% of damage.

- command system includes microprocessors performing tasks - to control power plant, to control cameras/storage/GPS's orbit determination devices, to control gyro-platform, to control attitude system of the craft. Commands is a sequence transferred from a mission control to a craft. Each microprocessor in a command system has unified system of commands.

- Gyro-platform with set of solid-state gyro and accelerometers, magnetometer

- Solar sensor / near celestial body detector is a set of IR sensors with partitions to observe part of the sky, and a software to process data from gyro-platform and readings from IR sensors.

- power plant with harvesting capabilities from solar panels, and storage of the power inside high volume capacitors for a pick use in case of mobility sessions and communication sessions.

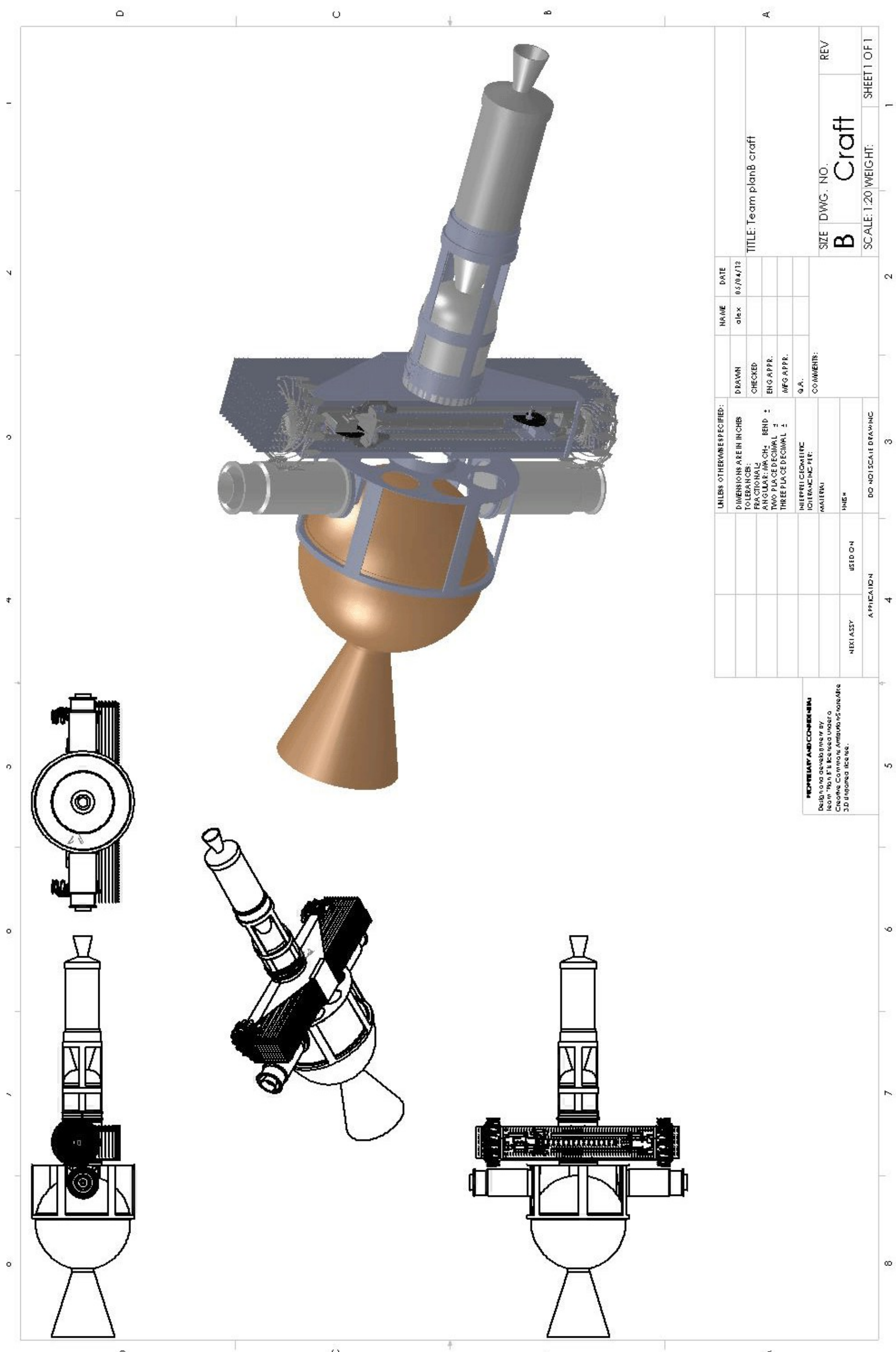
- Engines fixed impulses

- Frame with mounting brackets for engines. Mounting done by pyro fastening bolts

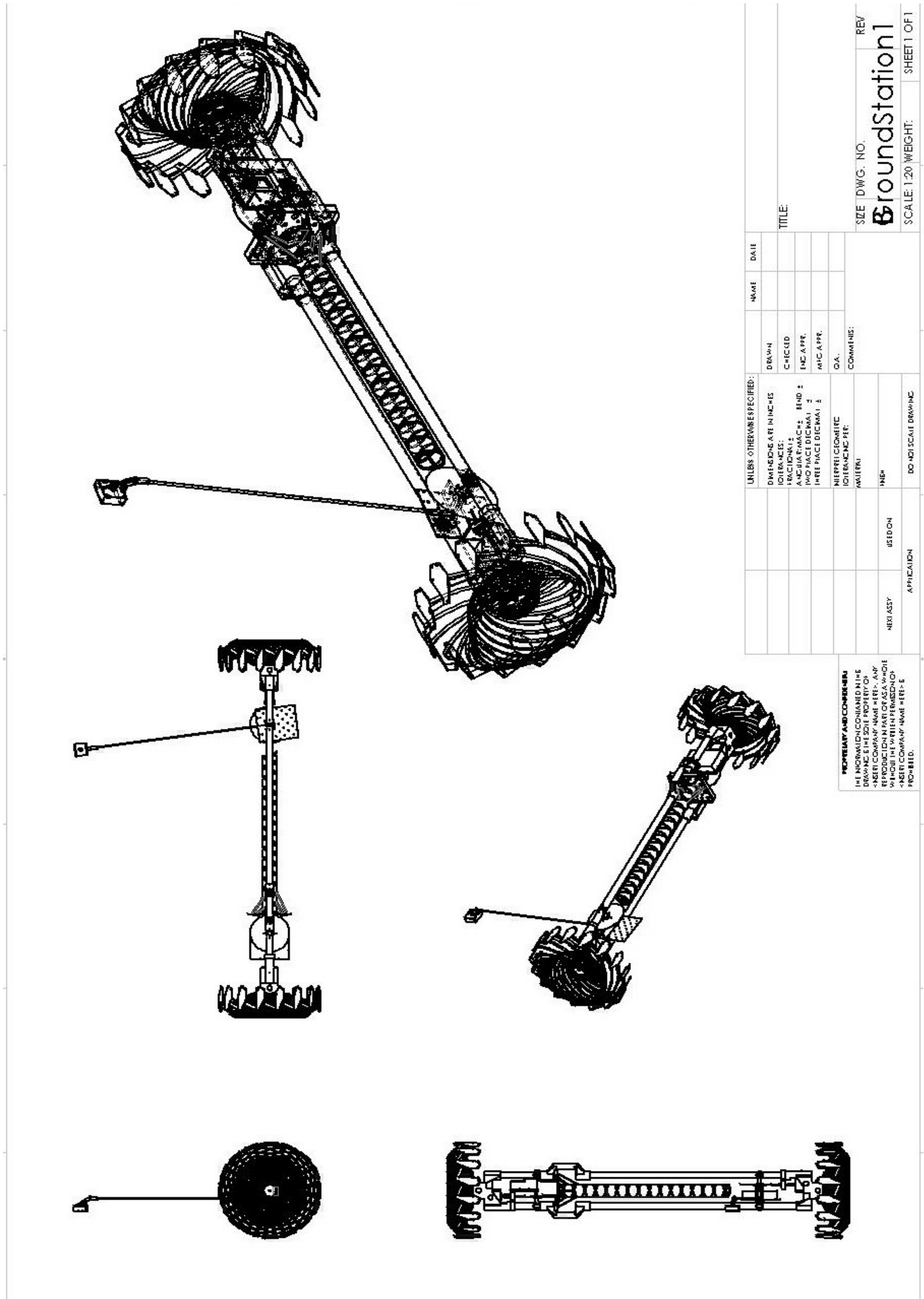
- landing system including rover, impact adsorption shield, and orientation system used to turn shield and rover in time of impact to face lunar surface from impact shield side.

4.2 Design Concept Drawing and Description

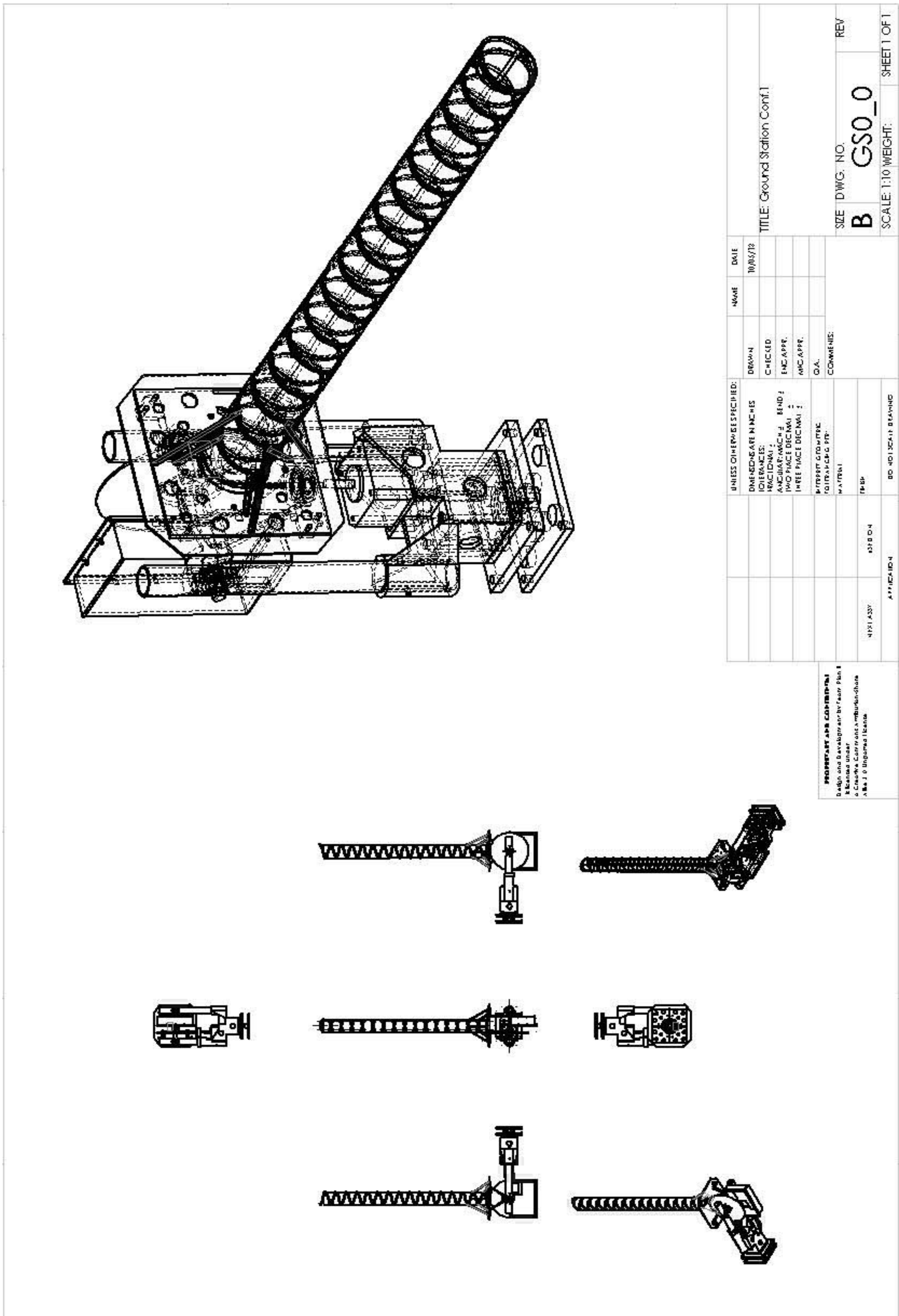
Pic 4.2.1 craft



Pic 4.2.2 Rover



Pic 4.2.3 ground station (simplified configuration)



5.0 Subsystem Breakdown

5.1 Attitude Control Subsystem (ACS) - mounts inside nozzle of a brake engine. It has cylinder shaped weight of 0.160g grams (hockey puck other name "face-off device") mounted on axis of a 0.21kg stepper motor. Another element of Attitude Control system is wheels of the rover. Wheels mounted on a stepper motors. At a time of a flight, the rotation of wheels will creates moment of rotation and will allow turning the craft around axis parallel of wheel's rotation axis. At the same time rotation of a cylinder shaped weight inside break engine nozzle will create rotation's moment and will allow making rotation around another axis - perpendicular to wheels axis. Attitude control system get commands from its micro-controller, that commands come from gyro-platform. Gyro-platform detects rotation based on data provided by two solid state gyros and used quaternions mathematics to perform integration of rotation speed data. Two gyro devices mounted such way that it's 3 axes compensate each other. For detection of a zero drift of gyro-devices was used similar to Kalman's filter mathematics. Attitude control system used special future - it can repeated sequence of rotation commands in backward order with time precision of 1 mks. That feature allowed gyro-platform to be in sleep mode (saving power) in a case of request to restore last craft's orientation.

6.1.1 ACS Trades

If for rover mobility will be decided to improve mobility performance by placing additional gears to stepper motors rotation wheels, that attitude control subsystem will be added with two additional rotation wheels, each of the weight of original one.

6.2 Orbit Determination Subsystem

For orbit determination was designed trajectory calculation software. It can work in distributed calculation mode with mission control distributed data from DB for processing "loop" records with time measurements performed by RF communication system, and raw data from GPS/Galileo system.

It is 3 methods of orbit determination. Global Navigation satellite's based and distance measuring system. Global Navigation System used two different type of orbit determination but both of them used raw Global Positioning System signal. First is a regular GPS device with capability to output raw data from GPS satellite. That device will be not able to determine orbit (position vector + speed vector) of a flying craft, but time stamps and GPS satellites position and velocity will be recorded, record will be transferred via backup communication system to process that raw signal to the mission control. Second system used for orbit determination is based on front-end RF global navigation system device. In that case raw signal require to be processed to extract raw signal from digitized data provided by front-end RF device. Last system for orbit determination is embedded into a communication system. Time measurements allow having a 35 m precision of measurements of distances from ground station to craft, and from craft to a ground station. Combined with trajectory calculation software that will allow using many measurements to estimate orbit with required to flight to the moon precision.

6.2.1 ODS Trades

6.2.2 ODS Analytical Methods

Trajectory calculation software was developed to work in destitution calculation mode. System allows determining orbit by trying some initial orbits parameters like velocity and speed at initial moment of time. Other measurements like distances measured and time stamps of received raw global navigation satellite data can be entered to try to match second observed record. Distribution of the data for calculation delivered from mission control server. Each small task to try different orbit for match all available data can be processed on separated computers.

To get orbit with 6 unknown parameters (speed vector is 3 dimensions and position is also 3 dimensions) needs to have 6 measurements to determine orbit.

6.4 Communications subsystem

For communication we chose a 2.4GHz frequency. That range is allowed to use (on earth) as long as the frequency channel is periodically switched in a 1 second interval. The core device used is a Blue Tooth front end Nordic semiconductor nRF24L01+. For power amplifiers was chosen standard WiFi power amplifier.

6.2.1 Communications Requirements

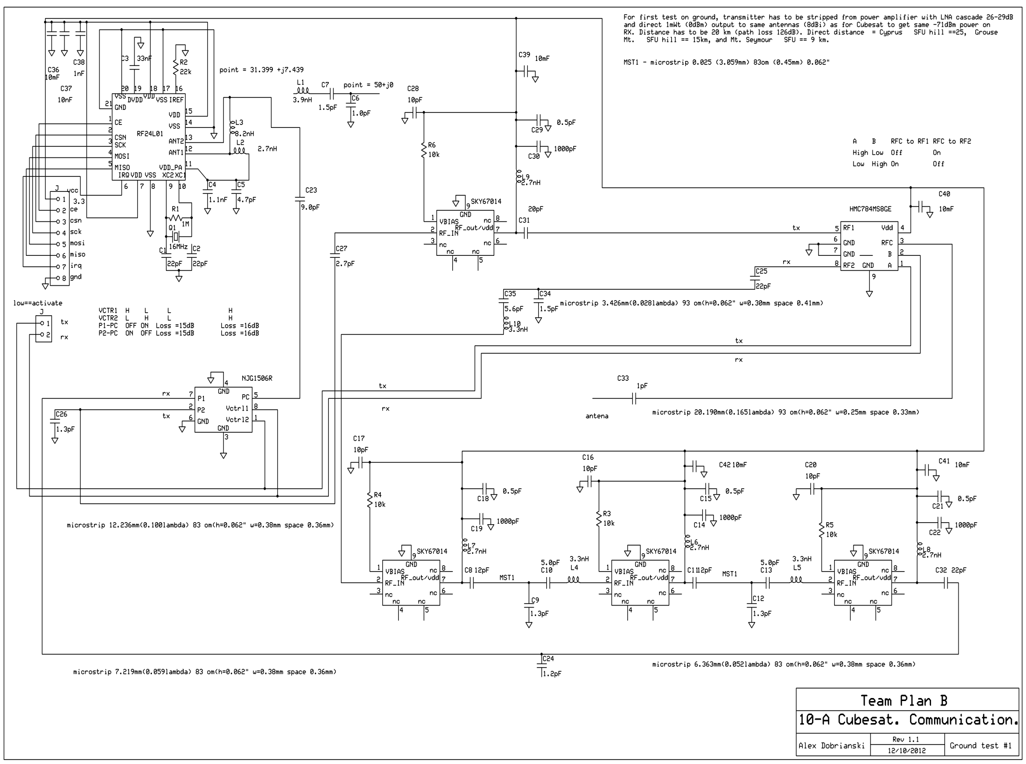
Operational conditions -40+125

Transmitting data by tripling packets over 3 communication channels.

Communication system require to communicate with probe on a distances ranged low earth orbit (200-300km), high orbit (>500km), earth-moon flight (< 400,000km).

6.2.2 Communications Design and MEL

Pic 6.2.2.1 schematic of the ver 2.0



6.2.3 Communications Trades

No trades.

6.2.4 Communications Analytical Methods

See 2.5.3.

6.2.6 Communications Risk Inputs

Backup communication system works only on LEO. Communication subsystem was not tested on a space flight

6.3 Command and Data Handling (C&DH)—Avionics

On board autonomy is supported by command subsystem state machines on each micro-controller. It is consists of a power plant automatic, thermal subsystem requests generation, gyro-platform automatics, nearest to celestial body center determination, sun direction determination, and orientation of the communication antenna to the earth ground station.

6.3.1 Avionics Requirements

To work under -40+125C temperature conditions.

Accepting commands for orientation from mission control web server

Serial communication loop 56000bit/s

Logical equivalent to ground station command system.

Automatic support of the orientation for communication system.

Automatic support for active thermal control.

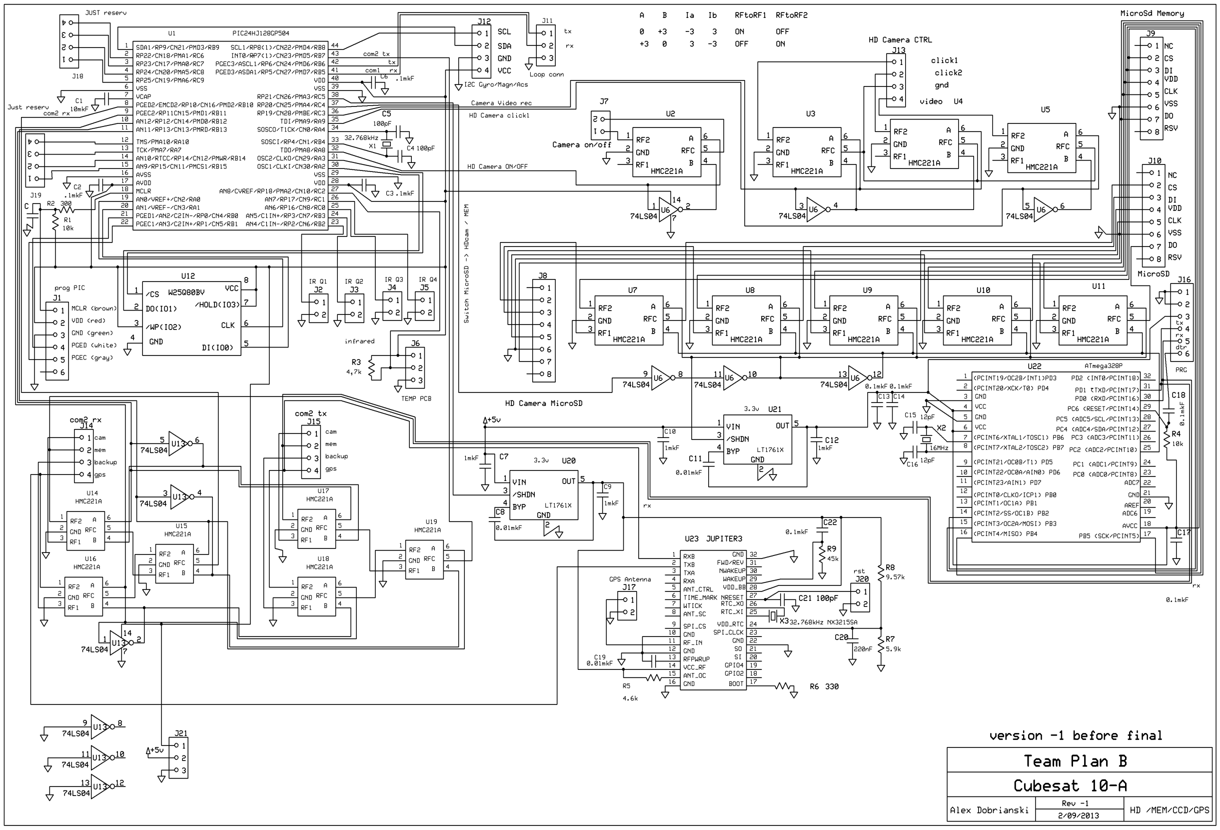
Gyro-platform 0.5 degree "zero-drift" under 1g environments.

6.3.2 Avionics Assumptions

Gyro-platform based on solid state gyro-sensors. Accelerometers 0.004 m/s2 precision measurements. Magnetometer with precision 230 LSb/gauss.

6.3.3 Avionics Design and MEL

Source code and schematics and PCB design available (gyro-platform) https://github.com/alexdobrianski/STM_BT (main PCB) https://github.com/alexdobrianski/STM_HDC



PCB board with microcontrollers on it 0.15kg

Sensors with PCB's 0.1kg

6.3.4 Avionics Trades

no trades

6.3.5 Avionics Risk Inputs

None was tested on real space flight

6.4 Electrical Power System

On board power plant supposed to be with flexible solar panels harvests energy under control of a micro-controller. Software controllable switches commutate best performed solar panel with harvesting energy controllers. Stored energy accumulated in group of high volumes capacitors. Switches controllable by software connect charged capacitors to distribution devices.

6.4.1 Power Requirements

8Wt power on lunar surface, 1 Wt Power on test nano-satellite flight.

Stepper motors for a mobility/attitude control 8Wt

Gyro-platform 0.12Wt

Communication 25Wt pick power

Electronics 0.001Wt-0.2Wt (deppend on a mode of operation controllable by a software)

6.4.2 Power Assumptions and Power trades

Flexible solar panel side is 63.5 mm. assuming it is hypotenuse - that geometrically gives in prime triangle with 45 degree angle, side sizes 45+45mm – which is bigger that original 63mm.

Sizes of each cell will be smaller (like 25.4 mm x 44.9 mm) but on same frame now it is 46 cells with 0.052 m^2.

On 2 frames can be placed 4 groups of 46 cells, calculating same efficiency and same Sun's power, that surface allows to harvest total 18.8 watt.

Same top-bottom and max outcome power is 9 watt. That is better, than previous design, and it mean == 1 minutes to transmit + 2 minutes to harvest.

Placing additional 2 stand (length 0.5m) which can hold 19x4 = 76 small size panels outcome additional 0.17m^2 with total power 15wt (and half of tit is 7wt). Gives the grand total is 34wt from all surfaces, and halving gives max s 17 watt power. Which is not bad – on top of each minute to transmit it will require 35 seconds to harvest.

Now the question where to place that two stands. Plus needs to balance weight- i.e. - 350grams for each 4 stepper motor, but gear motors can be reduced to have less torque. That can save around 50-100 grams on a motors in favor for stands. Reducing clearance makes smaller dimensions and also saves weight.

Total amount solar cell (sizes 45x25mm) to accommodate on rover today is 168. With weight limit of 400 grams for all system it is 2.3 grams per flexible cell including weights of holders and mounts. Plus needs to account that after 3 week flight cells will lost 1/2 of it power capability. Realistically it will be 8 watt of power at best.

6.4.3 Power Design and MEL

Power system design postponed as much as possible to a day of mission flight. New technology appears before old design and electronic components are obsolete. After research decision was made that power plant's design needs to be started 2 month before nano-satellite test flight and 3 month before main mission flight.

6.4.4 Power Trades

Power consumption controlled by a software running inside micro-controller (first powered) with nano-watt consumption. Additional power will be provided for such controller by harvesting methods embedded into a passive thermal control system. In absence of such system (based on Pelter elements) will be using solar cell for initial power to poer plant’s micro-controller.

6.4.5 Power Analytical Methods

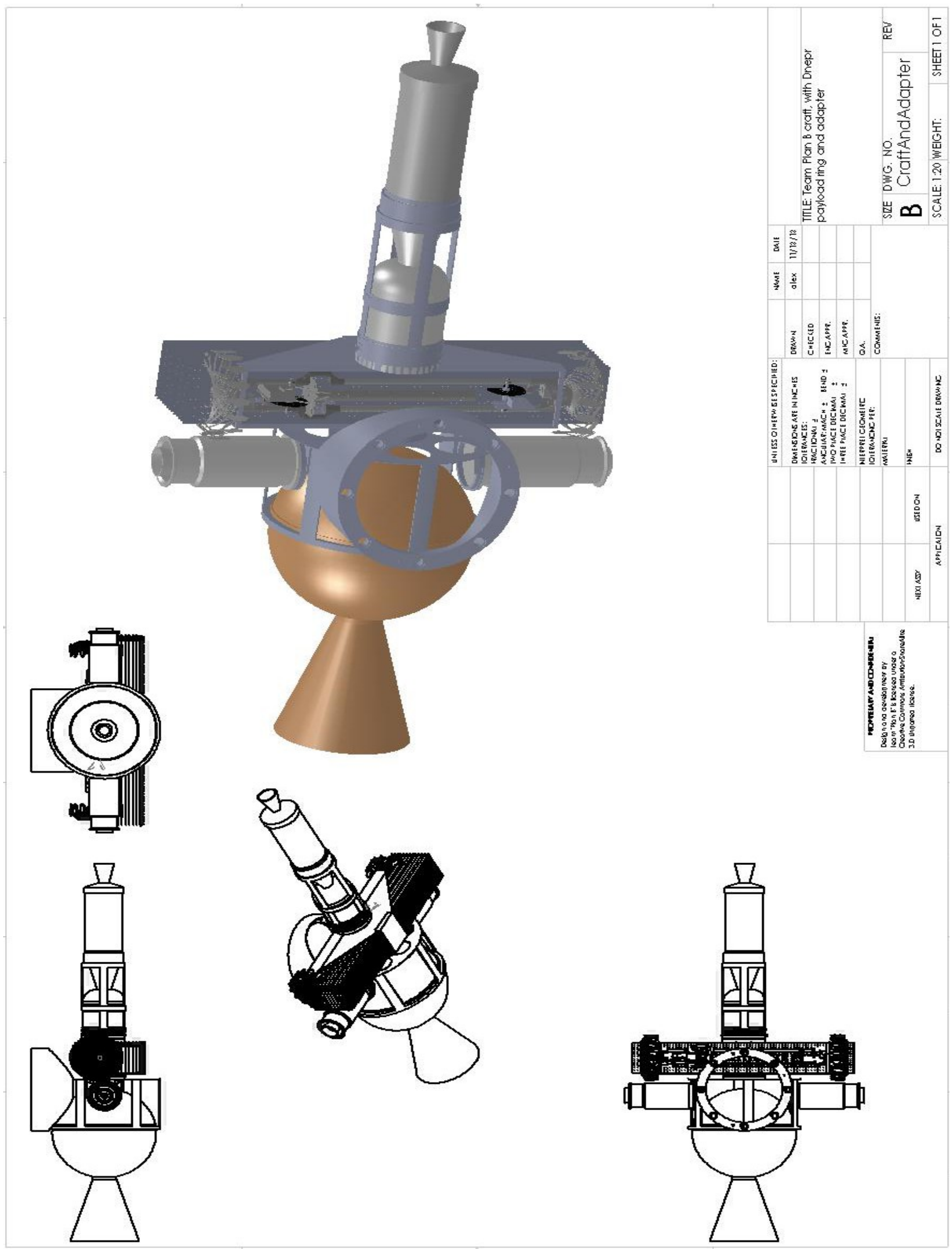
In low power consumption all micro-controller will use 0.001wt. With harvesting on sun side of the orbit 25 minutes will be enough to support all functionality of a on-board avionics on "night" part of the orbit.

With full powered avionics consumption of the power will be 0.5Wt, that will be enough to keep functioning nano-satellite on night part of the orbit with 1wt power produced by power plant on sun side of the orbit.

6.5 Structures and Mechanisms of the craft

The follow section describes the design structures and craft's details.

Pic 6.5.0.1 frame with payload adapter



6.5.1 Structures and Mechanisms Requirements

Acceleration applying to a frame, fixed impulse engines, rover and impact shield will be applied sequentially. First will be launch by itself.

Accelerations specific to launch vehicle-

- Dnepr structural stiffness with fundamental frequency hard mounted on separation plane no less 20Hz in longitudina and lateral axes

- Falcon 9 fundamental axial mode grater than 25Hz, fundamental bending mode grater than 10Hz

- Rokot payload minimum natural frequency axial >33Hzm and lateral >15Hz

- Dnepr max longitudinal <8.3g lateral <1g

- Falcon 9 longitudinal <8g lateral <2.1gt;2.1g

- Rokot longitudinal <8g radial <4g

Vibration profiles are-

for Dnepr -

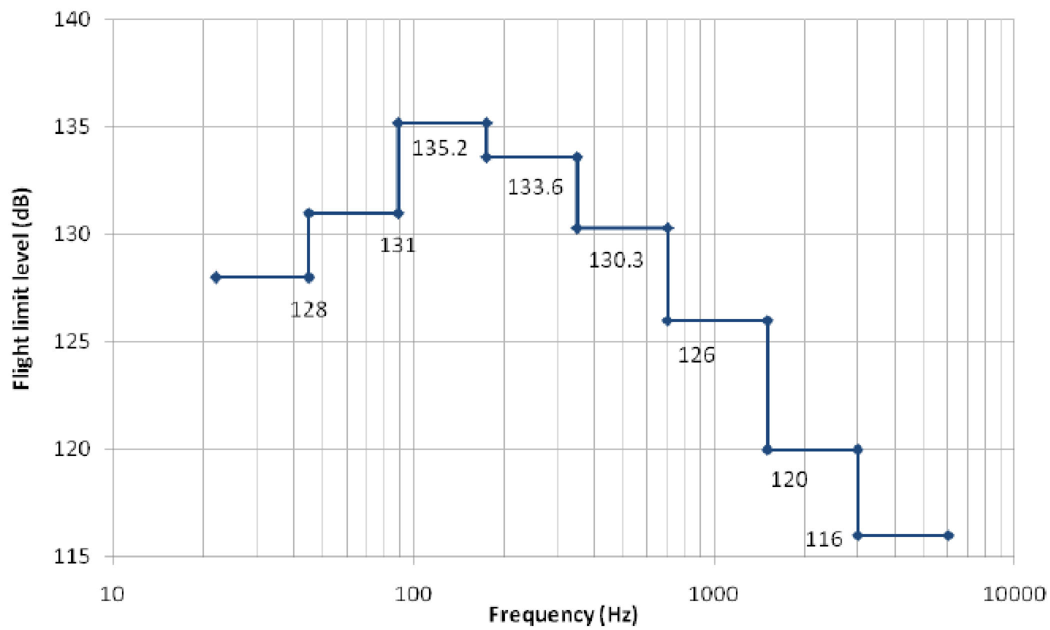
Pic 6.5.1.1 Amplitude of Harmonic Oscillations at SC/LV Interface. Longitudinal Axis (X)

Frequency sub-band, Hz	5-10	10-15	15-20
Amplitude, g	0.5	0.6	0.5
Duration, sec.	10	30	60

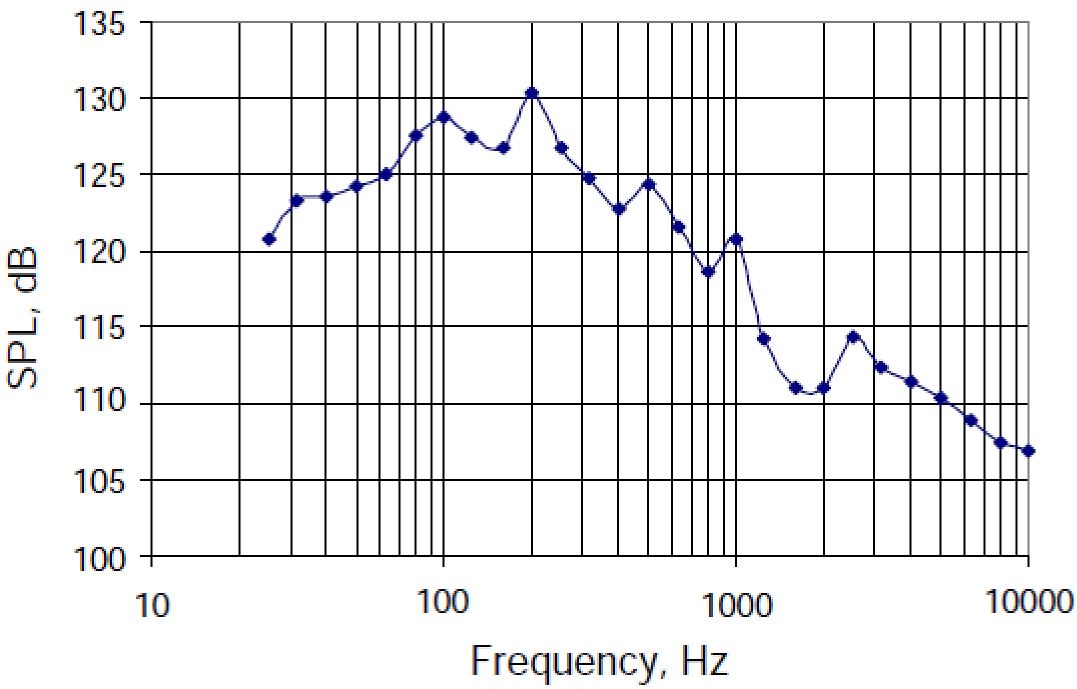
Pic 6.5.1.3 Spectral Density of vibrations -accelerations at SC/LV Interface

Frequency sub-band, Hz	Load Source	
	Liftoff, LV flight segment where M=1, q_{max}	1st stage burn (except for LV flight segment where M=1, q_{max}), 2nd stage burn, 3rd stage burn
	Spectral Density, g^2/Hz	
20-40	0.007	0.007
40-80	0.007	0.007
80-160	0.007-0.022	0.007
160-320	0.022-0.035	0.007-0.009
320-640	0.035	0.009
640-1280	0.035-0.017	0.009-0.0045
1280-2000	0.017-0.005	0.0045
Root Mean Square Value, σ, g	6.5	3.6
Duration, sec.	35	831

Pic 6.5.1.4 for Falcon 9



Pic 6.5.1.5 for Rokot



On main engine impulse acceleration expected to be <10g

On a brake impulse acceleration expected to be <10g

Impact shield needs to adsorb energy at impact from max 362m above lunar surface, free fall of max 21 second and at max speed 35m/sec at impact. That is equivalent of free drop in earth gravitation from 60m high - 24 floor of standard apartment building. Kinetic energy 7095kg * (m/s)^2 = 7.095 kJ. That is equivalent of a drop of 200kg from 1.76m on concrete floor.

6.5.2 Structures and Mechanisms Assumptions

Impact shield adsorbs - energy 7 kJ

Rover - F=4.7N (gear) F=3.244N (no gear, slope 10 degree, wheel deformation 80% = 0.108m)

6.5.3 Structures and Mechanisms Design and MEL

6.5.3.1 Rover drawings

See pic 4.2.2

6.5.3.2 Craft drawings

See pic 4.2.1

6.5.4 Structures and Mechanisms Trades

no trades

6.5.5 Structures and Mechanisms Analytical Methods

Engines – to be asked from manufacturer

Deformation of the each part under stress – to be asked from manufacturer.

6.5.6 Structures and Mechanisms Risk Inputs

None was tested in real flight

6.6 Main Mission Propulsion

6.6.1 Propulsion Management Requirements

удельный impulse better than:

Orbit correction	prpellant mass	Total Impulse lbf *s
LEO correction 1	4.85 kg	1538.5
LEO correction 2	4.85 kg	1538.5
Main burn	178.62kg	5620
correction 3	12.3kg	3715
break impulse	14.42kg	4606

6.6.2 Propulsion Management Assumptions

In Canada design and testing of a fixed impulse engines can be a serious question/problem/task related to political will of a government, as a result all what is possible to do is to make mockup engines, study and research of the manufacturing process, and purchase the engine available ether from government controllable agencies/companies. Purchased engines in different countries (US, Russia, Ukraine, China, and India) depend on political situation too. Ability to fit available fixed impulses engines resources to a space craft in matter of weeks is mandatory for success of the flight.

To accommodate difference of "what was calculated" to "what was bought" used the calculated time and direction of the firing of each engines on different path of the orbit. Impulse of a fixed engine cannot be reduced, but the firing direction and a time of firing of the second low earth orbit correction can make craft close or far from earth surface on a main impulse burn. That can change trajectory on the way to the moon and that second LEO correction will be equivalent of a reducing the main impulse. Proper calculated balance makes flight to a destination possible.

6.6.3 Propulsion Management Design Trades

Trades in design depend on trades of engines. Companies producing engines expressed only willing to discuss matter of trades up today.

6.6.4 Propulsion Management Design and MEL

Was considering available in public domain data and match was done based on a it:

Break impulse engine	weight of the engine, fastening bolts, frame connects to rover and impact shields	16.2 kg
2 correction orbit's engines	weight of the engine, frame, fastening bolts, connected to a frame of a brake impulse engine	6.5 kg
main impulse engine	weight of the engine, frame, fastening bolts, connected to a frame of the brake impulse engine	194.8kg
pre-main burn engine	weight of the engine, frame, fastening bolts, connected to a frame of a main impulse engine	3.2 kg
1 LEO orbit correction	weight of the engine, frame, fastening bolts, connected to a frame of the main impulse engine	3.2kg

6.6.5 Propulsion Management Risk Inputs

Critical risk is in brake engine ignition time and precision of the impulse. It is desired (nothing more) to have all engines produced by same manufacturer. To verify impulse of non-critical engines for a time and precision by recording impulse performance in flight together with temperature conditions will give estimates for a brake engine performance. First data for adjustments can be done on first LEO correction, then performance can be conformed on the LEO and main impulse burn. Finally precision can be measured on trans lunar orbit correction.

6.6.6 Propulsion Management Recommendation

It is hard to give recommendation of something which is out of your control. Be healthy and wealthy is better than to be a lunatic and a broken. Measurements of the performance of the engine can approximate health’s status of engines, and question about prices of lunar resources better be addressed to Columbus.

6.7 Thermal Control

Thermal control includes a combination of passive and active methods to keep electronics and mechanism under operation temperature. In passive measures includes a layer of gold on a box of the low resolution cameras and on HD camera box to reflect solar radiation. Passive system includes thermal heat conductors for all electronics surface mount components to be thermally coupled with copper layer on PCB. Heat conduction provided by carbon fiber composite with thermally conductive epoxy.

Active control includes two modes "on-flight" and "on-moon". Temperature measurements works together with gyro-platform to record temperature derivatives [see IS_TRA for detailed descriptions of the design] Attitude control of the craft can automatically orient craft to best direction keep temperature under operation conditions, or to support cooling/heating process. The same active thermal control used in rover, in this case on tip of HD camera box located temperature sensor allowing to measure difference of the temperature inside box and a tip of the box with contact to with lunar surface.

In passive thermal regulation used composites with filled hallow glass spheres. Nano-satellite passive thermoregulation will include Peltier elements to shield internals of the nan-satellite from sun thermal radiation. For temperature stabilizer used 6 water filed containers in walls of HD camera box with total amount of 0.01kg water. Containers filled with water and sealed under 5 mmHg vacuum.

Each part of electronics including HD camera, low resolution cameras, communication system transmitter’s equipped with temperature measurements devices. Temperature measurements used by low power consumption micro-controller to recommend active measures of orientation of the craft/rover, and to predict operation conditions. Out of temperature conditions used in decision making to used required devices. I.e. out of a temperature conditions of the communication transmitter gives indication to adjust series of burst of transmitting packets.

For thermal dissipation of communication transmitter will be used radiation ability of a reflector of the antenna and one Peltier element to harvest released thermal energy back. Patches of thermal conductive epoxy with carbon fiber threads and graphene will be used to adjust decapitations ability of reflector.

6.7.1 Thermal Requirements

Mechanisms -40+125C

Electronics -40+125C

HD camera -20+85C

Low resolution cameras -20+85C (-40+105C)

Communication system transmitter (-40+100C).

Sensors -40+105C

6.7.2 Thermal Design and MEL

Gold leafs total 0.001kg

Water filling containers each 0.01kg

Peltier elements 0.03kg

6.7.3 Thermal Trades

Active control provides balance for a use the power on a craft / rover. Test nano-satellite will provide data to such trade.

6.8 The Ground Station

In 2012 was made decision to use for a ground station ether rover by itself of some "simplified" version of the rover. Testing mission of a nano-satellite flight (appendix XX) in this case gives unique ability to test all-in-one systems of the rover/craft in time frame of couple minutes of communication session. Each failure in such tests can be analyzed and corrected on next orbit will be verification of a fixed problem. That will include functionality of the avionics, software algorithms, attitude control, and mobility of the rover. Such tests can provide debugging ability for systems on board, because all micro-controller are designed to be capable to upgrade its software in-flight. On ground station look of microcontrollers connects to ground station hub software to support communication to mission control.

6.8.1 Ground Station's Requirements

Gravity 9.8 m/ sec^2

Pressure 1atm.

Communication 0.dBm, 30dBm, 40dBm transmitting power

Two configuration - rover itself and ground station simple configuration with vertical mount of a frame.

6.9 Mission Operations

Mission control was developed as a core support for craft/rover/nano-satellite control and operation. It has browser based interface with distributable running orbit simulation/determination software.

Mission control (http://www.adobri.com/SatCtrlR.aspx) available over IP connection as HTTP/HTTPS server , it connects to ground stations hubs (also proprietary web servers running on a Windows based computers). Hub software connects via serial link to a loop of ground station micro-controllers. That micro-controllers performing various tasks like communication and orientation of the communication antenna. Task performed with the same set of commands acceptable by loop of micro-controllers on craft/rover/nano-satellite. That approach allows skipping development of separate subsystems for control nano-satellite, rover, craft, and ground station. Software and hardware on ground station and craft / rover, nano-satellite are the same. Compactness of a ground station is supported by using the same rover's mechanics in ground stations.

6.9.1 Mission Operations Requirements

9 ground station around the world to support 24 hours communication on trans lunar trajectory and lunar surface

Vancouver, BC, location 49.257735, -123.123904

Sarasota, Florida (TBD), location 27.306121, -82.569112

Joao Pessoa, East Brasilia (TBD), location -7.153526, -34.88966

Cape Town, South Africa(TBD), location -33.916743, 18.402658

Donetsk, Ukraine (TBD), location 47.971091, 37.712352

Karaganda, Kazakhstan (TBD), location 49.789355, 73.070583

Perth, Western Australia (TBD), location -31.967891, 115.883274

Hilo, Hawaii (TBD), location 19.707243, -155.064983

Rarotonga Aws, Cook Islands (TBD), location -21.255302, -159.768333

6.9.3 Mission Operations Design and MEL

Mission control available under Creative Commons Attribution-ShareAlike 3.0 Unported License on:

https://github.com/alexdobrianski/SATCTRL/tree/master/SatCtrl

https://github.com/alexdobrianski/GrStn https://github.com/alexdobrianski/VISUALTRA

https://github.com/alexdobrianski/TRA

7.0 Cost, Risk and Reliability

Boris Chertok in 2009 wrote:"...экономисты, получив задание доказать преимущества многоразовости космических систем подсчитали, что вывод в космос 1 кг полезного груза на «Спейс шаттле» будет обходиться первое время в $5000, затем в $1000 и при более чем 100 полетах в год дойдет до $100. В действительности американцы к 2010 году намерены прекратить эксплуатацию «Спейс шаттлов». Реальная стоимость по разным полетам составляет от 15 до $20 тысяч за 1 кг полезного груза, доставляемого

«Шаттлами» на МКС. Билет для полета не на Луну, а на международную орбитальную станцию с помощью российского транспортного корабля «Союз», стоит не $5000 обещанные фон Брауном для Луны, а $30 миллионов. Если инженер-конструктор или современный программист ошибаются в расчетах параметров и оценке технических характеристик, создаваемых ими сложных объектов в два раза, то его наказывают или даже увольняют. А экономисты ошиблись в 100 и более раз! Это происходит по причине полной некомпетентности или в угоду коррумпированным чиновникам и политикам."

"...economists , having been instructed to show the advantages of reusable space systems estimated the price of 1 kg of payload for the Space Shuttle system will cost: the first time around $5,000, then $ 1,000 and with more than 100 flights per year, price can dropped to $100. In fact, by 2010, NASA is planning to stop using the Space Shuttle fleet. The real cost for different flights ranges from $15,000 up to $20,000 per 1 kg of payload delivered by Shuttle to the ISS. A ticket for a flight to the Moon, not as well to the international space station with the help of a Russian cargo ship "Soyuz", is not the $5,000 promised by Von Braun for the Moon, but $30 million for IIS. If the design engineer or modern programmer mistakes in the calculation of the parameters and the evaluation of the technical characteristics of a system in half, then he will be fired. And economists were wrong by more than 100 times! This is due to complete incompetence or for the sake of corrupt officials and politicians."

He also points - "Для космонавтики начала XXI века стоимость вывода 1 кг полезного груза (по беспилотным программам) составляет 20–$25 тысяч, на геостационарную орбиту соответственно 30–$50 тысяч. Я не могу прогнозировать существенное удешевление вывода в космос полезных грузов в ближайшие 50 лет."

"For the beginning of the XXI century price of 1 kg of payload (for unmanned programs LEO) is 20 - $ 25,000, and for geostationary orbit, 30 - $ 50 thousand. I cannot predict a significant reduction in price of payload in the next 50 years." (see http://tvroscosmos.ru/frm/zhurnal/chertok.php)

We cannot say anything better than it was done in 2009. And taking situation "as it is", based on that approximation, the required for a mission 250kg on LEO will cost $5,000,000 to $6,200,000

Two missions planed for guaranteed reach the moon output price tag $10,000,000 - $13,000,000

For testing mission (1 kg of a nano-satellite) it should be in a range of $20,000

Cost of insurance - that parameter directly connected to a reliability of a launch vehicle and varies from 10% to 25% of a cargo price.

Another consideration for insurance - insurance companies will reserved their ability to have a deal, when they did not see any perspectives for a business involved (in space exploration). Government agencies does it business based on a budget, and budget can fluctuate, but it has some stability and tendency to increase. But for a business, even with clear picture what to do AFTER competition, insurance can create obstacle as high price of the insurance, probably in a range of 25% - 50% of a cost of a launch mission.

All that output price of two missions are in a range of $15,000,000 - $18,000,000.

7.1 Mission Risk

All risks with detailed description and plans to mitigate this risk by development and tests are in [IS_TRA] [IS_DVP] [LS_TRA] [LS_DVP] [MS_TRA] [MS_DVP].

Risk is also is releated to a time frame of a competition, needs to condider avalable scedules fron different launch providers.

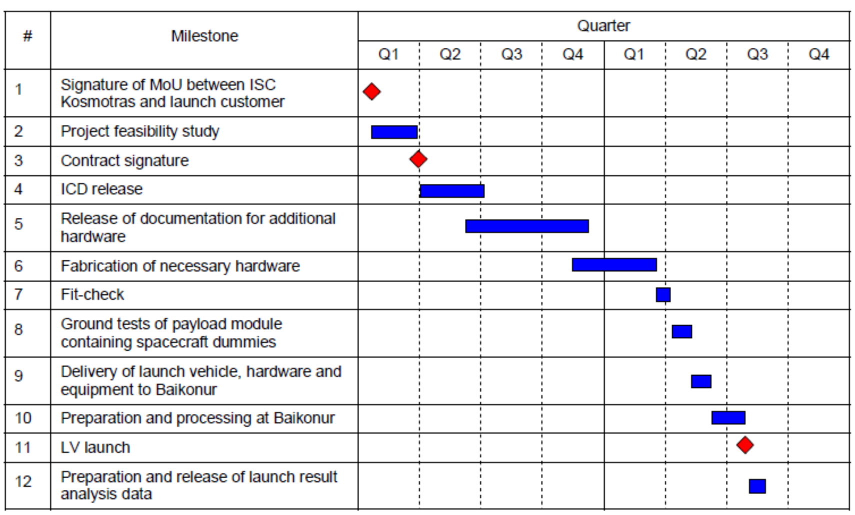
For Rokot launch vehicle:

Meetings / reviews schedule	Date
Contract signature meeting	L - 24 months
Technical kick-off meeting/ IRD review	L - 24 months
Launch vehicle to spacecraft system requirements review + ICD outline	L - 22 months
ICD review (draft issue)	L - 21 months
Launch Vehicle to Spacecraft Preliminary Design Review incorporating the preliminary mission design and analyses	L - 18 months
ICD review (issue 1)	L- 12 months
Spacecraft Operations Plan/ Joint Operations Plan review	As necessary, combined with other meetings
Technical interchange meetings	As necessary, combined with other meetings
Safety reviews (phases I, II and III)	As necessary, combined with other meetings
Launch Vehicle to Spacecraft Critical Design Review incorporating the final mission design and analyses	L - 12 months
ICD review (issue 2)	L - 11 months
Design qualification review	L - 5 months
ICD review (final issue, if applicable)	L - 5 months
Shipment Readiness Review	To be agreed
Launch Readiness Review (LRR)/ State Commission	L - 3 days
Launch Evaluation Review meeting	L + 2 months

for Falcon 9:

Launch – 18 months or more	**Contract signing and authority to proceed** • Estimated payload mass, volume, mission, operations and interface requirements • Safety information (Safety Program Plan; Design information: battery, ordnance, propellants, and operations) • Mission analysis summary provided to the Customer within 30 days of contract
Launch – 6 months	**Final payload design, including: mass, volume, structural characteristics, mission, operations, and interface requirements** • Payload to provide test verified structural dynamic model
Launch – 4 months	**Payload readiness review for Range Safety** • Launch site operations plan • Hazard analyses
Launch – 3 months	**Verification** • Review of Payload test data verifying compatibility with Falcon 9 environments • Coupled payload and Falcon 9 loads analysis completed • Confirm payload interfaces as built are compatible with Falcon 9 • Mission safety approval
Launch – 4-6 weeks	**System Readiness Review (SRR)** • Pre-shipment review (per stage; prior to shipment to the launch site) have occurred or are about to occur • Verify launch site, Range, Regulatory agencies, launch vehicle, payload, people and paper are all in place and ready to begin launch campaign
Launch – 2-4 weeks	**Payload arrival at launch location**
Launch – 8-9 days	**Payload encapsulation and mate to Launch Vehicle**
Launch – 7 days	**Flight Readiness Review (FRR)** Review of LV and payload checkouts in Hangar. Confirmation of readiness to proceed with Vehicle rollout
Launch – 1 day	**Launch Readiness Review (LRR)**
	Launch

for Dnepr:



In a [IS_DVP] in Appendix B there is a consideration of a time frame available today for a main and test missions.

7.3 Reliability

Test mission and main mission reliability are based on people participating in the project.

Appendix A. Test Mission.

10.1 Test Mission Top Level Design

During the 2011 team summit, there was an interesting development. One of the Google X Prize teams presented that its partner is currently developing a launch vehicle, capable of delivering a small payload to a low earth orbit.

Why not to try to use such an opportunity?

Maybe it will be a failure, but to spend 14 grand is not the same as to spend millions on a development of a satellite, and then either wait for an available rocket to place it into orbit, or watch undisclosed amounts of money burning in the atmosphere because of a simultaneous failure of the main and backup computers.

Before attempting to perform the main mission flight, the decision was made to use this opportunity with Interorbital Systems, where they offer the launch of a nano-satellite type nano satellite. For such a test mission, there will be the following key goals:

First – the orientation of a craft/satellite needs to be calculated with the precision of 0.3 degrees. If the is an error over 0.3 degrees, than the Moon is missed. If the nano satellite will achieve this precision, then it will be a green light for a craft itself.

Second – communication over two the channels (main and backup) has to be proved capable of transferring data to and from ground station(s).

Third – the requirements for the satellites needs to be passed, prior to placing a satellite inside a launch vehicle, and long before signing launch agreement. Passing ground tests is a good step, especially in an unknown territory, from the cradle of civilization, to the lunar surface.

Forth – ground station communication has to be developed and tested (ideally it will communicate with this satellite on low Earth orbit, in compact form, before the ground station in the form of a rover is sent to the Moon).

After brief consideration a decision was made – to try. As a result, trajectory parameter calculations were tailored for a proper low orbit (an essential part on the path to the Moon). Then another development was done - to create a working gyro platform.

Subsystems on the rover will be the same as on the nano-satellite: a camera unit (2 units looking in opposite directions), a gyro unit, a main computer with storage of data, a Communication unit, a backup communication unit, a power plant unit, an attitude control/orientation unit. If the development of such components will be successful, then the software can be re-used on a rover.

Backup communication: it is better to be capable of communicating with the nano-satellite even it is out of ground station reach. There is a decision to use existing satellite communication. Modems with such function appeared on the market over the last years: the restrictions are temperature requirements. To overcome these, we need to place the modem into a sealed epoxy box and traverse connectors outside the improvised compartment. Power requirements on transmitter will be 1.5 Watt instead of 10 Watts for the main mission.

Main communication will be at 2.4GHz. This band is designated as a hopping frequency band: as long as the transmitter does not stay on the same frequency channel, there is no license required to operate on this frequency, and no restriction on antenna with power up to 4 Watts (in Canada) or 1 Watts in US. There are recently developed OEM amplifiers modules for this band, with transition power up to 1-2Watts. The key module for a 2.4 GHz transmit and receive is just regular a Bluetooth device. The software required to be able to do error corrections on at long distances, with the "standard" BT repeating packet, or "standard" send/receive ACK/NACK, is not a practical way to do communications.

The power plant harvests as much energy as possible from the solar panel, and stores it in 6 capacitors for the test mission. High volume capacitors were chosen instead of batteries because of their better performance in temperature ranges. Range in temperature of -50 C to +125 C makes it suitable for the flight. The power plant needs to check which solar panel performs better, and what unit needs to get most power, then switch the source of power to discharge the capacitor's unit, delivering the power to the required unit. Performance of the power plant will be the main “source” of the information for a rover's redesign. The test flight of the power plant will be an ideal test of the latest harvesting techniques, and latest development of the solar panels. It was chosen to postpone the development of the power plant as much as possible. Industry developments in solar power harvesting sector is booming (something we cannot say about the space industry), and waiting for a flight for one two years can allow to use new technologies during the flight. This approach is risky - in the space industry it is assumed that design of any system has to be frozen 1-2 years before a flight. Engineers responsible for the integration satellites into a flight vehicle can be reluctant to accept such an approach.

The camera module can be a prime element for orientation – it will not only deliver pictures, but the software in the main computer can detect the horizon and that detection can help to calculate the direction to the center of the Earth. The camera needs to be stripped from of optics, and a pin hole will be the main optical element. The quartz glass on the sensor has to be drilled to make a hole to avoid breaking in a vacuum. Such a system will work alongside a solar sensor and infrared celestial body detector.

The orientation and attitude control module will have 3 small stepper motors (instead of 4 on the rover). The task for the modules will be to adapt rotation to keep required orientation calculated by gyro platform module.

The gyro - platform was the main development of 2011, we are happy with how it is working.

The frame of the satellite, according specifications of a nano-satellite, has to be built within specific standards. On the market now there are a lot of different frame and nano-satellite kits available. Team Plan B decided to build our own frame: an antenna for a communication module has to be deployed; this is not a standard configuration. One of our partners, Jetasonic Technologies Inc, a company, based in Coquitlam BC, Canada, developed such a frame.

On the ground station, one of the tasks performed by the rover on its main to the lunar surface is to orient the antenna to the Earth for communication sessions. If we were to combine ground station with the rover (the rover is the same as the ground station), then both tasks designed of the rover and design of the ground station will be achieved simultaneously.

A test mission is considered essential by our team before any attempt to reach the Moon. It is premature to talk, to plan, or to build anything before the mission can be successful.

Goals for a testing mission

a) Test mobility of the rover by applying the same design (of a rover) to a ground station

b) To learn how to orient satellite on the orbit

c) Test communication system

d) Test command system and all loop of data transfer from satellite to a mission control website.

e) Check the ability for independent ways to determine orbit parameters

Missing Growth, Contingency and Margin Policy

In nano-satellite world nano-satellite has pretty good, and clear specification/requirements. It weight is less 1.3kg and sizes specified as 0.1m x 0.1 m x 0.1m. Power plant has to be switched of before flight. Special off/on switch powers up all onboard electronics after separation from a container mounted on launch vehicle.

limitation of the weight for a mission is 1.3 kg sharp. Anything high then 1.3 kg is not accepted for a flight.

Contingency is 0 gr, no margin policy assumed.

10.5 Test Mission Description

The nano-satellite will be discharged of all high volume capacitors, before being placed into a container, mounted on the launch vehicle. During design, it was was assumed that the power switch creates a short-cut for the electrical circuit, to satisfy the requirements of a nano-satellite class launch.

After reaching orbit, nano-satellite will be separated from the other nano satellites in the container, and the power switch will disconnect the short-cut circuit. At the same time, a small solar cell will start to charge the high volume capacitor of the main microprocessor. The charge used in the operation be ready after 30 minutes, to allow main microprocessor to run for next 1 minute at full speed, or with reduced speed continuously. Changing the speed from full to slow will be done by the microprocessor itself.

After main microprocessor reached its functionality, it will switch on the power plant microprocessor. The power plant will control attitude control and the orientation subsystem, to maximize current of the solar panel(s). It is the task of the power plant microprocessor to determine the orientation based on best solar power harvesting. It is not required on this step to orient or rotate the nano-satellite to face one of the surfaces to the best illumination by the Sun.

After power storage, consisting of high volume capacitors, achieves some level (approximate 0.25 of max power), all subsystems of the nano-satellite are powered and self-tested. This gives the command for opening the helical antenna. The antenna does not need to be opened if the nano-satellite launches with an already extended antenna. In this case the nano-satellite will use two units inside a container instead of one.

3 nichrome wires will be burned, one after another, and the antenna will be released. To extent the antenna to a working position, the attitude control and orientation subsystem will perform a prerecorded turn to lock the antenna in a fixed position, by two permanent magnets. A backup communication system antenna will be mounted on the opposing side of the helical antenna's reflector. From that moment, the nano-satellite is ready for backup communication with mission control, and to send a message "I AM OK".

In a case of antenna opening failure, or failure of locking the antenna in fixed position, the backup communication will be ready to send messages to mission control.

The backup communication is performed using an Orbit Communication M-10 modem. It is not certified to communicate on aircraft, but Orbit Communication representatives confirmed that on-board a satellite, it was used in the past successfully. Communication of the Orbit Communication M-10 modem is initiated by applying power to its power pins, performed by the power plant subsystem. Then it needs to monitor the signal on the "satellite detected" pin. That pin provides detection of an Orbit Communication satellite in range of the M-10 modem. Each modem has unique identification. After sensing the Orbit

Communication satellite, it is possible to send a 140 byte message to mission control, confirming status of the nano-satellite on the orbit.

After sending the "READY" message, it can be 15-20 minutes before the Orbit Communication satellite reaches the ground station, to deliver the message to Earth. In this time nano-satellite will collect raw GPS signals from two global navigation subsystems, and 5 different raw GPS satellites, collecting position and velocity together with time stamps. This will be next message to be transferred via backup communication subsystem.

Mission control can send commands to nano-satellite using Orbit Communication's infrastructure, but the main task on this stage will be to process raw GPS data from the nano-satellite, and attempt to determine orbit using raw data and trajectory calculation software.

After the parameters of the orbit are determined it will be possible to make an attempt to communicate by a ground station. Trajectory calculation software will determine the time and window of visibility for each ground station, and from the ground station, calculate the sequence of orientation vectors to be performed by the antenna of the ground station. The same calculation will produce angles of orientation of the nano-satellite.

Now it is the task for the orientation and attitude control subsystem to achieve and hold orientation of the nano-satellite. First there will be an attempt to use the solar sensor. The attitude control and orientation subsystem will try all possible movements performed by the 3 stepper motors, in all combinations. The gyro-platform will provide quaternions of rotations after each possible movements are performed by 3 stepper motors. The values of quaternions will be stored for later use. A period of the function of voltage, measured from one of the four infrared solar sensors will be performed, and movement by stepper motor will be used to maximize period. After a period reaches same predetermined value, there will be fine tuning, as all four infrared sensors have to produce the same photo-voltage value, which will be an indication of the Sun's position. That orientation is stored by the gyro platform, as a direction to the Sun. The next step will be attempting to detect, by two infrared sensors, the edge of the Earth. Here, the nano-satellite will be put into a slow spin. Values from the two IR sensors will be recorded. A peak in the values will indicate the edge of the Earth. The orientations at the time of 4 peaks will give 4 vectors, enough to determine direction to the Earth. By knowing the time when direction to the Earth is detected, orbit parameters, and solar direction will give all data to orient nano-satellite for communication session with the ground station. Indication to mission control that nano-satellite is ready orient for a session is confirmed by sending 2 vectors: the Solar direction, and the center of Earth's direction, and the time when that direction was found.

Mission control calculates the angles, and sequences of the antenna's directions based on Solar and Earth orientation. That sequence is send to the nano-satellite using Orbit Communication infrastructure. The sequence of directions for the ground station will be sent encrypted, over internet. Both objects ready to perform the main task and main test for the mission.

The nano-satellite orients itself based on Solar direction, and center of Earth direction (achieved automatically) and sequences of direction vectors transferred from mission control previously. The ground station (which is mechanically equivalent to a lunar rover) will turn it's antenna to the nano-satellite. Communication session will be performed by a 2.4 GHz communication subsystem. All waiting data from nano-satellite (like images made by low resolution camera, and HD video) will be delivered to the ground station. If the communication session is successful, than it is a green light for the main mission to the Moon.

Some additional tests will be performed. The ground station will be configured to a compact version. Here, the mobility of the rover will be stripped by removing part of the frame with one wheel, then another wheel will be removed, and the stepper motor shaft will be mounted on horizontal rotation holder. In such configuration one stepper motor is controlling horizontal rotation, and the antenna mount is controlled the second stepper motor. Software on the ground station has to adapt itself for 2 instead of 3 degrees of freedom, and properly orient the antenna for the next communication session with the nano-satellite.

In another test, the communication system (located on a back side of the reflector of the antenna) will be switched to a 1mW ground station transmitter. Here the nano-satellite must support the communication system because its low noise amplifiers must be capable of supporting communication.

Finally the communication system will be switched to 1 low noise amplifier, where the BT module with one 12.5 dB low noise amplifier will be capable of picking up signals from a 400 km distance.

Trajectory Baseline for the mission.

Trajectory is not determined yet, even payments were made in full.

Launch Vehicle Details

Unknown today. Launch vehicle in a development now.

Test Mission. Costing:

After short investigation was found available launch vehicles - SPACEX - price after discount as a GLXPRIZE competitor $180,000, European Space agency - 90,000 - 120,000 EURO, Russian space agency $90,000.

It is desired, possible and visible for a private space business to support "big" players, connected their business with taxpayer money, but it is nothing new in space industry for a last 60 years, it will be "preferable" to support "space enthusiasts" willing to risk but create something new, new rockets, new systems. We decided to give a chance in our test mission to new player and it was arranged payment for 1 nano-satellite flight $14,500. Up today luck is not on our side and definitely not on the side of a "new player" launch vehicle by "Inter rbital" LTD is not in "ready to flight" state.

We give another 9 months to a "Interorbital" before considering money lost.