Feasibility study of RUFS-1

WITH THE USE OF ORBITAL SIMULATION
DONE IN MATLAB

OSCAR HAG
Acknowledgement

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Abstract—A satellite, with the call sign RUFS-1, will attempt in the first half of 2016 a launch into low Earth polar orbit. This feasibility study shows that RUFS-1 is from an overall technical perspective capable of fulfilling its mission. This was determined by a break down of the mission requirements into a few key questions. These were answered through various means; firstly an orbital simulation, secondly a link budget and finally a risk assessment of the entire satellite project.

1 INTRODUCTION
In this article a feasibility study is performed on a CubeSat called RUFS-1 containing a 3D printed miniature of a stuffed animal (Rufs). This stuffed animal is the mascot of the project and is also what the satellite is named after. What this study entails is a broad examination of key aspects of the system: the orbital performance, communication, power and thermal variation. There is also a basic risk assessment because in order to reach a conclusion on whether the mission will succeed, both the technical requirements and the risks have to be reviewed in order to determine the overall likelihood of this mission.

This mission came about as a an idea for inspiring children and the young within the STEAM-field (science, technology, engineering, arts and math). The projet is run by a group of volunteers from various backgrounds all drawn together by a passion for both space and science. The reason a CubeSat mission was chosen was because the technological advancement that have concurred over the last 10-15 years (especially within computers and mobile phones) have enabled smaller satellites to become more and more capable. There has also been a larger push for enabling the launch of these so called pico-satellite. These things have collectively lowered the bar of entry for access to space and with this CubeSat mission the project group intends to show that space is within reach for anyone, even for a bunch of amateurs.

2 MISSION STATEMENT
The main goal of this mission is to launch a CubeSat to lower earth orbit. The preliminary time of launch is Q1/Q2 2016, which places some substantial challenge in terms of available time. A basic CubeSat-kit has already been procured with an included launch to polar orbit. This together with a tight budget has in effect locked the basic components of the satellite. There will, however, be a review of potential extra mission payload with the already stated constraints taken into account. Because of the size of the CubeSat (10 × 10 × 10 [cm]) with at least half of that volume taken up by the core systems, only a miniature version of the stuffed animal will be brought up to space.

So what will the satellite do when it reaches orbit, what is the mission? Well here the task is rather simple, it is going to try to establish contact with ground station start sending telemetry back to earth. This is also where the mission becomes a bit vague. Since an option for additional payload is included within the mission statement, ceratin tasks or mission goals will only be set in effect if their required systems are brought in to orbit. For example if a camera is brought onboard the CubeSat, the goal of the mission will be to take a picture of earth. On the other hand if a third party hardware is included then the mission goals shifts towards transmitting data back to them.

3 PROCEDURE & ASSUMPTIONS
In order to reach a conclusion on whether RUFS-1 will be able to complete its mission statement this feasibility study will make use of a procedure of several consecutive steps. The first one covered is an orbital simulation done in MatLab. The results are then used for a mission duration assessment, a power budget and a thermal budget. The next step is a linkbudget which will be done separately from the orbital simulation; it will be based on the worst case scenarios (the satellite being at maximum range from the ground station for instance). The final step will be a risk analysis for the entire mission, including everything leading up to launch.

Essential questions that need to be answered within this study:
- How long will the satellite remain in orbit?
• Will the communication system be able to sustain a link between the ground station and spacecraft?
• How often will there be line-of-sight from the ground station to the satellite?
• How much power will be available in orbit?
• At what temperatures will the satellite operate?
• Are there enough margins within the already existing parameters for a third party mission payload and/or a camera?

Throughout this feasibility study many assumptions are made, some are more specific for certain sections while others are more fundamental for the entire study. The specific ones will be covered in their respective sections whilst the fundamental ones, which will be referred to as "base assumptions" from here one are covered in this section.

The assumed orientation of RUFS-1 while in orbit as seen in figure 1 shows the front of the satellite aligning with the prograde vector ($\vec{e}_{vs}$), the rear being where the antennas are located. It is also assumed that the prograde vector is perpendicular to position vector ($\vec{r}_s$), who’s unit vector is the normal vector for the local plane ($\vec{e}_n$). $\vec{e}_{hs}$ is the unit vector of the orbit rotation vector for the satellite and is calculated by equation 1. These three unit vectors are used as the vectors of the satellites orientation, the front of the satellite corresponding to $\vec{e}_{vs}$, the top to ($\vec{e}_n$) and ($\vec{e}_{hs}$) being the side.

$$\vec{e}_{hs} = \vec{e}_{rs} \times \vec{e}_{vs} \quad (1)$$

Another important aspect that is assumed is the solar irradiance levels. Since the sun intensity varies it is very hard to predict the specific intensity at the time of launch and this issue effects several different parts of the study. Those being: density levels in the atmosphere, solar panel power output and temperature variations onboard the satellite. For both the power output and temperature variation a reasonable estimate is to set it at the solar constant of 1361.5 [W/m²].

The atmosphere on the other hand is much more linked to the variations in solar intensity. As seen in figure 2 the annual average shows s periodic behaviour but on a 30-day or single day mean it fluctuates greatly. This creates changes in the atmospheric density which is the root cause for the many uncertainties within low earth orbit prediction. Here the study will use three different atmospheric levels with low, medium and heavy mean solar irradiance in order to estimate how long the satellite will remain in orbit. This is covered in more detail in section 5.2.

It is also assumed that every time the satellite has a line of sight at the chosen ground station it will start transmitting data from its radio.

3.1 RUFS-1 (Hardware)

As was stated in section 2 a basic CubeSat-kit forms the basis of this mission. It consist of:

- Transceiver: Radiometrix (TR2M-436.50-10-ARS). RF-Power: 100 [mW], NBFM: 25 [kHz]. [1]
- Microcomputer: Arduino mini 04 [3]
- 1U Cubesat chassi
- Battery: Li-ion 3.7 [V] 5200 [mAh] Rechargeable Battery Module [5]
- Dipole antenna.

These components are going to be assembled into a CubeSat that will have the functionality to transmit basic data down to Earth via UHF radio. The kit is essentially what is often called a "Beep-sat". A cutaway rendering of the assembled satellite is shown in figure 3.
One thing to note is that as seen in figure 3 only two thirds of the volume is taken by the core components. This leaves space for the miniaturization of Rufs as well as a third party payload. What is missing from the figure is that the front and back will have thin polished aluminum sheets covering them. There are two reasons for this; firstly to dampen the thermal absorption by reflecting the Sun and secondly to increase the odds of RUFS-1 being observable from Earth. For this study two versions of RUFS-1 will be looked upon, one will be the basic kit (plus the aluminium sheets) with nothing else onboard and the second one is including an additional half PCB as a third party payload as well as additional batteries filling up the mass to a total of 1 kg.

3.2 Estimated ground station

Because of the unorthodox nature of this satellite project a specific ground station has not been chosen as of this writing. For this reason a so called "estimated ground station" will be used in this feasibility study. Its position will be a very typical site, namely northern Sweden (Luleå specifically) with the coordinates used in the study being 65.584 [deg.] North and 22.154 East. The capabilities of this ground station are taken mainly from an off the shelf system built by ISIS Space [14].

3.3 Targeted orbit

Launch services for this mission is provided by Interorbital Systems. RUFS-1 will be sent in to space upon a proprietary launch vehicles designed by Interorbital. This system is still in the development phase with a suborbital test launch scheduled for June/July 2015. An orbital test flight will take place in November/December the same year. RUFS-1 is scheduled to fly on the first or second commercial launch which are set for the first and second quarter of 2016 respectively. If the satellite project might happen to slip past those "dates" it would be moved to next upcoming launch. The rocket will be launched from a barge off the coast of California in the United States. The preliminary parameters of the target orbit are stated to be:

- Apogee \((r_a) = 310\, [\text{km}]\)
- Perigee \((r_p) = 305\, [\text{km}]\)
- Inclination \((i) = 90\, [\text{degrees}]\)
- Argument of perigee \((\omega) = -90\, [\text{degrees}]\)
- Longitude of launch site \((\Omega_L) = -125\, [\text{degrees}]\)

As of this writing there is no information on what time of day the launch will take place or an exact date. Since this is an unproven launch system it is also impossible to be certain on how accurate the rocket will be (how close the actual orbit will be to the targeted one). Here the study will assume that both the targeted apogee and perigee altitudes as well as both the argument of perigee and the right ascension of ascending node will be met. On the other hand three different inclinations will be tested, 85, 90 and 95 [deg.] in order to simulate some of the "margins" of the satellite system. This will be specified when it is employed.

4 ORBITAL SIMULATION

In order to ascertain the orbital performance of the satellite a simulation code, created in MatLab, is used. It runs a full scale model in a cartesian reference frame with a rotating spherical Earth and a yearly moving Sun. The reference frame is centered on Earth with the X positive direction set in the direction of vernal equinox and Z positive is parallel to Earth’s rotational axis.

Figure 4 shows how the entire simulation is setup up in a 3d cartesian coordinate system with \(\bar{r}_s\) as the satellite position and \(\bar{r}_{gs}\) being the ground station position. The Sun moves on a circular disk that is inclined 23.5 degree’s to Earth’s equatorial plane, \(\bar{e}_{sun}\) is the unit vector of its position.
An orbit that has a maximum altitude of 310 km, as seen in figure 5, will have two predominate effecting forces. The primary gravity from Earth including the J-factors and atmospheric drag.

\[ \ddot{a} = \vec{G}(x_0, y_0, z_0) - F_{\text{drag}} m_s \cdot \vec{v} \]  \hspace{1cm} (2)

Equation 2 is one of two main governing equations within the simulation. The acceleration vector (\( \ddot{a} \)) is the result of the two external forces taken into account. \( \vec{G} \) is the gravitational vector (including up to 4th level J-factors) as a function of the satellite’s position relative to Earth’s reference frame. \( F_{\text{drag}} \) is aerodynamic forces from the thermosphere acting in the opposite direction of the velocity vector. The other main equation is the change in temperature:

\[ dT = \delta Q(T, x, y, z) \cdot \frac{1}{C_{\text{tot}}} \]  \hspace{1cm} (3)

Where \( \delta Q \) is the change of heat as a function of current temperature, the satellite’s position and its orientation relative to Earth as well as the sun. \( C_{\text{tot}} \) is the thermal inertia of the satellite.

Equation 2 and 3 are solved numerically in Matlab with use of a program called ode45 [6] for a set time interval. To do this they need to be rearranged into a function (\( f \)) building an ordinary differential equation (equation 4) and then implemented in a simulation script (called m-files in MatLab).

\[ \frac{\partial f}{\partial t} = f \left( x, y, z, \frac{dx}{dt}, \frac{dy}{dt}, \frac{dz}{dt}, T \right) \]  \hspace{1cm} (4)

The initial conditions (\( \vec{x}_0 \)) are calculated from the preliminary orbital parameters supplied by launch services. The simulation will start at the right ascension of the ascending node with the initial orbital elements (apogee, perigee, inclination and argument of perigee) translated into cartesian position and velocity values by classical elliptical orbit principals. The standard gravity equation has been replaced by the “gravityzonal.m” function from MatLab (section 5.1) to better correlate with the simulation setup. One thing to keep in mind is that these orbital elements will begin to change almost immediately because of the oblateness model. For instance; the satellite will start with an indicated apogee of 310 [km]. On its way there the apogee may have become lower because of the change in gravity. These elements are unique to the specific point in time they where calculated.

\[ \vec{x}_0 = \left( x, y, z, \frac{dx}{dt}, \frac{dy}{dt}, \frac{dz}{dt}, T \right) \]  \hspace{1cm} (5)

Ode45 takes the function file and uses it together with \( \vec{x}_0 \) to integrate over a specified time interval \( T_0 \) with use of an explicit Runge-Kutta (4,5) formula [6]. Since the solver adapts its time steps based on the specified relative and absolute tolerances, only the start and end of the interval are stated. From this a solution matrix \( \bar{X} \) is produced with a corresponding time vector \( t \).

Since the simulation will run for several days of simulated time while at the same time fulfilling the desired accuracy, the solution matrix and its corresponding time vector will be very long. Actually storing all the created information can become a bottle neck for the program, thus hampering its effectiveness. In order to lessen these effects ode45 is run for a specified time interval, a segment of the total time, that is set at 300 seconds. The code runs initially from 0 to 300 seconds and then the final step in the result vector is used as a new \( \vec{x}_0 \) for the next interval. This interval starts at 300 [s] and goes up to 600 [s] and the process repeats itself. The program can be set to save the entire result vector or a scaled down resolution. Since the program will be run for a long time with the overall result being most important (total mission time and so). The resolution for saving the data be set at one data point for every 300 [s].

The time interval setup is also used as a “window” in to the numeric integration for checking how much the satellite’s original orbit has deteriorated. After each time segment has been solved with ode45 the code checks the satellite’s altitude above Earth. First if it is lower than 150 [km] which will trigger a shortening of the subsequent intervals to 60 [s]. Then the code starts to check if the satellite is below 100 [km] which will trigger a stop to the simulation and the total run is completed. The reason for implementing this at these specific heights is that at lower altitudes the atmospheric drag begins to have a very high impact on the simulation whilst at the same time the simplified aerodynamic model starts to lose its validity, the actual situation being more akin to re-entry dynamics. By first altering the interval length before placing a stop at 100 [km] it is ensured that the results stay valid and that the code does not overshot the stop with a too high margin.

The four main aspects calculated within the function file to create the ode system are:

- Gravity
- Atmospheric drag
- Line of sight determination
- Thermodynamics
4.1 Gravity

Since the satellite will be travelling on a low Earth orbit the main governing gravitational forces will only be Earth’s, while the contribution from the moon and sun will be neglected. To simulate the perturbation effects of Earth oblateness, a gravity vector is calculated by the “gravityzonal.m” function included in MatLab’s aerospace toolbox [7]. It takes the coordinates of a point relative to a fixed Earth reference system and calculates the gravitational forces acting at that point. Since Earth is rotating relative to vernal equinox a shift on the axial plane via a transfer matrix needs to be done in order to have the correct coordinate too use in the function.

\[
R(\phi) = \begin{pmatrix}
\cos(\phi) & \sin(\phi) & 0 \\
\sin(\phi) & \cos(\phi) & 0 \\
0 & 0 & 1
\end{pmatrix}
\] (6)

Where \( \phi \) is Earth’s axial rotation relative to the equinox, which is calculated by

\[
\phi(t) = \phi_{\text{earth}} + \phi'_{\text{earth}} \cdot t
\] (7)

with \( \phi_{\text{earth}} \) being Earth’s starting condition and \( \phi'_{\text{earth}} \) its rotational speed.

With the coordinates transferred in to Earth reference system, the gravitational accelerations are calculated by ”gravityzonal.m” and transferred back to the original system. This is done by simply multiplying the rotation angle with -1 and using it with the same rotation matrix. These accelerations are then used in equation 2 as the gravity vector \( \vec{G} \).

4.2 Atmospheric drag

The atmospheric drag is the main contributor for orbital decay. Equation 8 is the standard way for calculating the drag. It is built upon the integration of work in a fluid with \( \rho \) being the atmospheric density, \( A \) is the reference area, \( v \) is velocity of the object in the fluid and \( CD \) as the drag coefficient. This is a very basic model. Since RUFS-1 will be travelling at such high speeds and relatively low atmospheric densities the behavior of the fluid around the object is much simpler. The particles are essentially just colliding with the front of satellite and not curving around it. With this in mind the equation becomes a reasonable estimate of the force involved.

\[
F_{\text{drag}} = \frac{1}{2} \cdot A \cdot CD \cdot \rho(r) \cdot v^2
\] (8)

Atmospheric density \( \rho \) is dependant on two factors. The satellites altitude and the atmospheric “height”. This “height” is how the atmosphere scales depending on the current solar intensity. The simulation has a data table with three levels of scaling: low, medium and high. “Low” corresponds to the lowest measured levels of solar activity, “high” comes from the highest and “medium” is the average levels. This data table is taken from NASA Msis-E90 [19] database on 10 km intervals (ie 0, 10, 20 [km] and so on) with a corresponding value for \( \rho \). It goes from an altitude of 0 up to 900 [km]. Because of this an interpolation processes is needed in order to correctly simulate the changing atmosphere. This processes is done by a 4th level polynomial function.

The interpolation process is setup as a function file within MatLab. The code checks the desired altitude \( (h_i) \) and compares it to the data set and selects the closest lower value. For example if \( h_i \) is equal to 234 [km] it chooses 230 [km]. Since it uses a 4th level polynomial function 5 data points are necessary. Thus it takes 2 above and two below the found point. The code now has two five element vectors \( \vec{H} \) (altitudes) and \( \vec{\rho} \) (densities). It takes the logarithmic value of \( \vec{\rho} \) and uses the polyfit.m function in MatLab to create \( \vec{P} \) which is the corresponding polynomial coefficients, equation 9.

\[
\vec{P} = \text{polyfit}(\vec{H}, \text{log}(\vec{\rho}))
\] (9)

It then uses \( \vec{P} \) in the polyval.m function together with the sought altitude \( h_i \) to calculate the logarithmic \( \rho \) value, equation 10.

\[
\text{log}(\rho_i) = \text{polyval}(\vec{P}, h_i)
\] (10)

After this it removes the logarithmic state on result, which is sent out to the aerodynamic simulation. At the same time the function saves both \( \vec{H} \) and \( \vec{P} \). The reason for this is that for the next time the function is called upon to calculate a value for \( \rho \) it checks if the new \( h_i \) lies within the span of the previously created \( \vec{H} \). If it does only the “polyval” part of the process is preformed, otherwise the entire procedure is repeated.

As can be seen in figure 6 the data set (seen as circles) is followed well by the interpolated values. Naturally the red line matches all the data points but it also retains the general curved shape of the data set.

Since the satellite has a very simple shape the reference area is the side area of the cube. \( A \) is set to 1 [dm²]. CD is typically very high for a satellite since they are not very aerodynamically shaped and because the Reynolds number for these types of situations are very low (Re < 1). The drag
coefficient for a cube is roughly 1 but because of the low Reynolds number a reasonable estimate is to set it to 2.5 [8].

Another aspect that needs to be taken into consideration is the effect of Earth’s rotation on the velocity vector. The principal of atmospheric drag is that its force vector is in the opposite direction of the velocity vector in the medium creating it. What that means for \( \vec{e}_r \) in equation 2 is that it is a summation velocity vectors. The velocity vector stemming from the gravity simulation (RUF-1’s orbital velocity vector) and the velocity vector created from Earth’s rotation. This creates the complete velocity vector for the satellite when it travels through the atmosphere.

\[
\vec{V}_{rt} = \begin{pmatrix} -\sin(\phi) \\ \cos(\phi) \\ 0 \end{pmatrix} r_s \cdot \sin(\theta) \cdot \phi_{\text{earth}}
\] (11)

Equation 11 is the velocity vector from spherical coordinates with the assumption that all variables are constant apart from the longitudinal rotation angle. \( \phi \) is the longitudinal position of RUF-1 in Earth’s reference frame with \( \phi_{\text{earth}} \) as Earth’s angular speed. \( \theta \) is the latitudinal angle and \( r_s \) is the distance from the satellite to Earth’s center. This simulates the effects of an assumed atmosphere rotating in sync with Earth. When the satellite is above the equator the rotational contribution will be at its highest (488 [m/s]). \( \vec{V}_{rt} \) is added to the orbital velocity vector. The absolute value of the resulting vector is the velocity value (\( v \)) used in equation 8. The unit vector is the one used for the drag acceleration in equation 2.

### 4.3 Line of sight determination

Determining if the satellite is visible from certain points in space is needed for several reasons. The first being if it has line of sight to the ground station for radio communication. The second is its position relative to the Sun and Earth in order to calculate the power output of the solar panels as well as the thermal energy absorbed from the two celestial bodies.

With radio communication it is often talked about the angle over the horizon a margin for seeing the satellite from Earth. Within this study it is assumed to be 5 degrees. This value is taken from the general rule of thumb within the industry. That contact will be established when the satellite is between 5 and 10 degrees above of the horizon as seen from the ground station.

\[
\beta(t) = 90 - \arccos(\vec{e}_{gs}(t) \cdot \vec{e}_{ls}(t))
\] (12)

Equation 12 calculates how high the satellite is above the local horizon. Since both Earth and the satellite are in motion in the chosen reference frame the angle is a function of time. As seen in figure 7 \( \vec{e}_{ls}(t) \) is the unit vector of the relative vector \( \vec{r}_{ls}(t) \) between the ground station and the satellite. \( \vec{e}_{gs}(t) \) is the unit vector for the ground station. This angle is checked for every time step done with ode45 to calculate how much total radio time the mission will have. It is also used as a criteria for checking how much power is in use onboard the satellite for the thermal equation. Since it is assumed that the satellite will be broadcasting when it has line of sight to the ground station which triggers a higher power consumption onboard during that time. This is covered in greater detail in following section.

![Fig. 7: Angle over local horizon](image)

The other main line of sight determination needed to fulfill the goal of this simulation is between the satellite and the Sun. Only the blocking of Earth is taken into consideration with this setup. For the code to decide whether Earth is blocking the sun, two consecutive criteria have to be fulfilled. The first is the angle between the Sun’s vector \( \vec{e}_{sun} \) and the satellites position vector \( \vec{e}_{bs} \). The Sun’s unit vector can be seen in figure 4.

\[
\sigma_p(t) = \arccos(\vec{e}_{sun}(t) \cdot \vec{e}_{rs}(t))
\] (13)

If this positional angle (\( \sigma_p \)) exceeds 90 degrees the second criteria is checked, which is the positional angle relative to the total angle shown figure 8. This total angle \( \alpha_p \) is calculated by equation 14.

\[
\alpha_p(t) = 90 + \arccos(r_c(t)/\vec{r}_{earth})
\] (14)

\( \alpha_p \) is the maximum allowed angle between \( \vec{e}_{rs} \) and \( \vec{e}_{sun} \). It is calculated by the trigonometric relationship between \( rs \) and \( r_{earth} \). The added 90 degrees is for placing the satellite on the opposite side of Earth relative to the sun (see figure 8). If \( \sigma_p \) is greater than \( \alpha_p \) the code dictates that direct sunlight is blocked by Earth. This is used for two tasks within the code. The first being the thermodynamics simulation for both the sun heating up the satellite and the internal power estimation. The second is the total power estimation which is done externally from ode45.

![Fig. 8: Maximum allowed angle (\( \alpha_p \)) for \( \sigma_p \) based on altitude above Earth](image)
4.4 Thermodynamics & Power generation

An important aspect needed to be calculated is how the temperature onboard the satellite varies during the mission. This is done because the electronics onboard have different functionality limits in terms of the temperature they can withstand and still be fully operational.

Equation 3 calculates the rate of change of the satellite’s temperature. As was stated in the section 3 the simulation does not look at individual components but instead sees the spacecraft as one complete unit. The heat capacity of the system $C_{tot}$ is calculated from the different heat capacities of the components that make up the satellite:

$$C_{tot} = \sum_{i=1}^{j} c_i \cdot m_i$$ (15)

$c_i$ is the specific heat capacity of a certain component, for example the batteries are set at 1070 [J/(kgK)], with $m_i$ being the components mass. This is summarized for all the individual pieces of the satellite and for the second version of the satellite there is more mass for the batteries, see table 1.

**TABLE 1:** Data on specific heat capacity and mass for the various components that make up RUFS-1 (version 1 & 2)

<table>
<thead>
<tr>
<th>Component</th>
<th>specific heat capacity [J/(kgK)]</th>
<th>mass V.1 [g]</th>
<th>mass V.2 [g]</th>
</tr>
</thead>
<tbody>
<tr>
<td>solar panels</td>
<td>710</td>
<td>14</td>
<td>14</td>
</tr>
<tr>
<td>chasse</td>
<td>910</td>
<td>300</td>
<td>300</td>
</tr>
<tr>
<td>battery</td>
<td>1070</td>
<td>100</td>
<td>434</td>
</tr>
<tr>
<td>PCB</td>
<td>600</td>
<td>251</td>
<td>251</td>
</tr>
</tbody>
</table>

The second portion of equation 3 is $\delta Q$, the heat path function. It is the difference between heat absorbed and heat emitted from the satellite.

$$\delta Q = J_s \alpha A_{sun} + J_d \alpha A_{alb} + J_p \epsilon A_{pht} + P - \sigma T^4 \epsilon A_{surf}$$ (16)

As can be seen in equation 16 $\delta Q$ is built up by several separate contributing sources.

**TABLE 2:** The different heat contributions.

<table>
<thead>
<tr>
<th>Heat Source</th>
<th>Expression</th>
</tr>
</thead>
<tbody>
<tr>
<td>heat received directly from the sun</td>
<td>$J_s \alpha A_{sun}$</td>
</tr>
<tr>
<td>albedo contribution</td>
<td>$J_d \alpha A_{alb}$</td>
</tr>
<tr>
<td>planetary radiation contribution</td>
<td>$J_p \epsilon A_{pht}$</td>
</tr>
<tr>
<td>internally dissipated power</td>
<td>$P$</td>
</tr>
<tr>
<td>heat radiated to space</td>
<td>$-\sigma T^4 \epsilon A_{surf}$</td>
</tr>
</tbody>
</table>

These different contributions as seen in table 2 are summarized in equation 16 to calculate the change in heat. The table itself stems from [8]. Each of the three external contributions is built upon a heat source factor ($J$), an absorption coefficient ($\alpha$ & $\epsilon$) and a corresponding surface area ($A_{(i)}$). The fourth is the internally created heat within the satellite: the wiring and circuits heating up from the electrical power. This happens while running the onboard computer (Arduino chip), charging the batteries, broadcasting or receiving signals. Here is where the code uses the previously described line of sight equations to determine which case of onboard power levels are in play within the satellite. The final part is the how much heat is radiated into space from the satellite.

4.4.1 Heat source factors

The $J$-factors seen in table 2 are case dependent. $J_s$ is the solar radiation intensity at 1 AU distance, the same as the already stated solar irradiance constant of 1371 [W/m²]. The second is albedo factor $J_d$ which represents the solar radiation reflected off of Earth’s atmosphere.

$$J_s = J_o \alpha F$$ (17)

As seen in equation 17, $J_s$ is estimated from the solar radiation factor together with an albedo reflection factor $\alpha$ and a visibility factor $F$ [8]. The reflection factor could range from as low as 0.05 all the way up to 0.80 depending on if the satellite is passing over clouds, water or forests (with forest giving low reflection and clouds giving high). For a long running simulation such as this one the average value of $\langle 0.34 \rangle$ can be used [8]. The visibility factor is a little more complicated. Depending on the altitude and the angle between local vertical and the Sun’s position unit vector ($\vec{e}_{sun}$) it will vary from $10^{-4}$ to $10^{3}$. The estimated average value of the remaining range will be 0.1 [8]. $J_o$ represents the planetary radiation. For engineering purposes an average level of 237 [W/m²] can be used as a base value and that it emanates uniformly from Earth [8]. Since the intensity falls with inverse-square law $J_p$ can be approximated by equation 18, see reference [8].

$$J_p = 237 \left( \frac{r_e}{r_s} \right)^2$$ (18)

4.4.2 Internally dissipated power

**TABLE 3:** Internal power criteria.

<table>
<thead>
<tr>
<th>Case</th>
<th>Sunlight</th>
<th>Line of Sigh of to Ground Station</th>
<th>Outcome</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>No</td>
<td>No</td>
<td>Base consumption for receiving signals and running onboard computer.</td>
</tr>
<tr>
<td>2</td>
<td>Yes</td>
<td>No</td>
<td>Power from solar panels unless case 1 is a higher wattage.</td>
</tr>
<tr>
<td>3</td>
<td>Yes</td>
<td>Yes</td>
<td>Broadcasting power consumption minus the power transmitted to antennas.</td>
</tr>
<tr>
<td>4</td>
<td>No</td>
<td>Yes</td>
<td>The same as case 3.</td>
</tr>
</tbody>
</table>

Table 3 shows how the code decides which of the different levels of internally dissipated power is chosen based on whether the satellite has line of sight with ground station and so on. One thing to note with case 2 is that the charging power from the solar arrays is dependant on the satellites orientation. Since this might be lower than the base consumption of all the systems onboard, the code checks which is higher and chooses the option with the highest power.

$$P = A_{panel}((\epsilon_{sun} \cdot \vec{e}_{rs}) + (\bar{\epsilon}_{sun} \cdot \bar{\epsilon}_{hs})) \epsilon_{panel} S_E$$ (19)

Equation 19 calculates the power output of the solar panels based on the satellites orientation relative to the sun. $\epsilon_{panel}$ is the efficiency of the panels, $S_E$ is the solar irradiance and $A_{panel}$ is one side area of the solar panels. Each solar panel
is made up by shard shaped cells (see figure 3). There are 56 in total with 16 cells for each panel. Thus:

\[ A_{\text{panel}} = 16 \cdot A_{\text{cell}} \]  

(20)

with \( A_{\text{cell}} \) taken from [4] as 2.277 [cm\(^2\)] and \( \epsilon_{\text{panel}} \) is also taken from the same source with the value being 27%. \( S_E \) is set at 1371 [W/m\(^2\)] [8].

4.4.3 Surface heat absorption

The heat absorbed form the sun is similarly to the solar panel power output orientation dependant. Equation 21 calculates the total area seen by the sun.

\[
\alpha_{\text{sun}} = A_{\text{srf}} (\epsilon_p | \epsilon_{\text{rs}} | + \alpha_p | \epsilon_{\text{rs}} | + \alpha_p | \epsilon_{\text{hs}} |) \]  

(21)

This equation uses the same mathematical principals as equation 19 but instead of the solar panels it takes the total area of each side (\( A_{\text{srf}} \)). Together with a surface dependant absorption factor the total absorption area can be calculated. Since each side has different absorption intensities there are separate absorption factors \( \alpha \) for each surface type. As can be seen in figure 3 four of the six sides are covered by the solar arrays with the remaining two (front and back) being covered by polished aluminium. The absorption factors used for these surface in the simulation are shown in table 4.

### Table 4: Surface factors [8].

<table>
<thead>
<tr>
<th>Surface</th>
<th>Absorbance</th>
<th>Emittance</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Cells, GaAs</td>
<td>0.88</td>
<td>0.80</td>
</tr>
<tr>
<td>Polished Alumínium</td>
<td>0.21</td>
<td>0.08</td>
</tr>
</tbody>
</table>

Both for the albedo and planetary radiation contributions the surface area \( (A_{\text{alb}} \) and \( A_{\text{plt}} \) respectively) is 1 [dm\(^2\)]. This is because of the assumed orientation of RUPS-1.

4.4.4 Heat radiated

The final “contribution” is the heat radiated from the satellite. It is build upon the internal heat of the satellite \( (T) \), the Stefan-Boltzman constant \( (\sigma) \) and the total surface area and its corresponding emittance factors.

\[
\epsilon_{\text{Aeff}} = \sum_{i=1}^{N} (A_{\text{Aeff}})|\epsilon_i| \]  

(22)

Equation 22 calculates the total surface emittance of the satellite. The emittance factors are taken from table 4. As seen in table 2 \( (\sigma) \) which is set at to 5.67 \cdot 10^{-8} [W/(m\(^2\)K\(^4\)].

It is taken from [8].

Equations 15 and 16 are used in equation 3 to calculate the change in temperature within the ode45 function file. This means it is solved in parallel to equation 2.

5 SIMULATION RESULTS

As was stated earlier, the simulation calculates speed and altitude for the satellite. Since the calculations are highly dependant on the initial values it becomes hard to tell if the results are reasonable. In order to test if the code is preforming as advertised, certain orbits with known behaviors are needed to be run.

All of the required resulting plots are included in appendix A, with only some specific plots placed within the section 5, mostly as tables containing the needed total outcome will present in this section 5.

5.1 Verification of simulation

In order to verify that the simulation is working two different types of trials are done. The first is specifically testing the validity of the gravity model. This is done by running the simulation with CD set to zero, in effect removing the aerodynamic drag from the simulation. Two special orbits will be run for this, the first is a sun synchronous orbit and the second being a Molniya orbit, both of which will be run for a quarter of a year \( (365/4 = 91.25 \text{ days}) \). The initial parameters for each simulation can be found in table 5. Each of these orbits have different purposes. The sun synchronous is a low eccentricity orbit that has an inclination specifically tuned for shifting its ascending node. This shift is done at the same rate as Earth’s movement around the Sun, hence the name sun synchronous orbit. The Molniya orbit is a highly elliptical orbit with apogee placed above the north pole. Its inclination is chosen so that the argument of perigee remains unchanged for its orbital life time.

### Table 5: Table for orbits

<table>
<thead>
<tr>
<th>Orbit Type</th>
<th>Sun Synchronous</th>
<th>Molniya</th>
</tr>
</thead>
<tbody>
<tr>
<td>Inc</td>
<td>99</td>
<td>63.4</td>
</tr>
<tr>
<td>Apogee</td>
<td>800</td>
<td>46000</td>
</tr>
<tr>
<td>Perigee</td>
<td>775</td>
<td>450</td>
</tr>
<tr>
<td>Argument of Perigee</td>
<td>−90</td>
<td>−90</td>
</tr>
<tr>
<td>RAAN</td>
<td>150</td>
<td>150</td>
</tr>
</tbody>
</table>

The desired outcome for these two test runs are somewhat different from one another. For the sun synchronous orbit a 90 degree change shift in the orbital plane is expected whilst for the Molniya orbit the argument of perigee shall remain unchanged.

Fig. 9: 3D plot of a Molniya orbit run in the simulation code.

As can be seen in figure 9 the argument of perigee remains the same throughout the entire simulation. The
double outline of the orbit track is due to the change of RAAN (figure A.3). For the apogee and perigee there are some relatively small variations but nothing of note. (figure A.2)

Figure 10 shows the change of RAAN for the orbit and on a initial glance it seems to have fullfilled the stated goal. Although as seen in figure A.6 the movement of RAAN is greater than the desired outcome, $\delta \Omega$ being equal to 91.25 degrees, overshooting with 1.25 degrees. As was stated in section 5.1 the Earth oblateness model causes the orbital characteristics to change. Figure A.5 shows that both apogee and perigee varies from their original starting point. This could have the effect of altering the rate of change for RAAN since the specified inclination (which causes the change in RAAN) is dependant on both apogee and perigee. It is also known that satellites placed on these types of orbits tend to need station keeping protocols [11] in order to sustain their specific orbits over longer periods of time. Overall the gravity model seems to deliver a reasonable result from the simulation.

The second trial is for the atmospheric model. To test it, CubeSats that have been ejected from the ISS are used as test cases. This creates very clear initial conditions, since both the exact date and orbital conditions for the "launch" are fully known. It is also known how long the satellite remained in orbit. One thing to note is that the atmosphere varies a great deal throughout the year/years. Because the simulation can only handle one atmospheric setting per run, for each test case three simulation are performed: one for every intensity.

Table 6 shows the two test cases used for this trial. The first ones is ArduSat-1 and the second is F-1. They are very comparable to each other with both being 1U in size, having a mass of one [kg] and an inclination of 51.6 degrees. This is also very comparable to RUPS-1, except for the inclination. Since the only variances between the two satellites are the time of launch and re-entry as well as apogee and perigee the code will be setup to run in six different initial conditions. For ArduSat-1 apogee is set at 414 [km] and perigee 410 [km]. For F-1 the initial apogee is 410 [km] with a perigee of 402 [km]. All of these trial runs will be run at the same inclination ($i = 51.6$ degrees), argument of perigee ($\omega = -90$ degrees) and RAAN ($\Omega = 150$ degrees).

As seen in table 7 the results vary greatly depending on the chosen level of intensity for the atmospheric model. Both ArduSat-1’s and F-1’s three trials either overshoot or severely undershoot their actual orbital durations.

Table 6: CubeSat test cases

<table>
<thead>
<tr>
<th></th>
<th>Launch</th>
<th>Re-entry</th>
<th>Days</th>
<th>Apogee</th>
<th>Perigee</th>
</tr>
</thead>
<tbody>
<tr>
<td>ArduSat-1</td>
<td>2013-11-19</td>
<td>2014-04-16</td>
<td>148</td>
<td>414 [km]</td>
<td>410 [km]</td>
</tr>
<tr>
<td>F-1</td>
<td>2012-10-04</td>
<td>2013-05-09</td>
<td>217</td>
<td>420 [km]</td>
<td>402 [km]</td>
</tr>
</tbody>
</table>

As seen in table 7 the results vary greatly depending on the chosen level of intensity for the atmospheric model. Both ArduSat-1’s and F-1’s three trials either overshoot or severely undershoot their actual orbital durations.

Table 7: Trial results

<table>
<thead>
<tr>
<th>Atmospheric intensity</th>
<th>Low</th>
<th>Medium</th>
<th>Heavy</th>
</tr>
</thead>
<tbody>
<tr>
<td>ArduSat-1</td>
<td>605</td>
<td>115</td>
<td>12.4</td>
</tr>
<tr>
<td>F-1</td>
<td>602</td>
<td>114</td>
<td>12.4</td>
</tr>
</tbody>
</table>

Figure 11 shows that during the launch of both of these satellites the sun had below mean activity. This could be one reason for the simulations result being 2/3 or 1/2, for ArduSat-1 and F-1 respectively, of the actual orbit time. Since the F-1 was launched before ArduSat-1, at the time of lower activity, it remained in orbit for a longer time. This is not conclusive proof that the aerodynamic simulation is perfect but it does show that it can give a reasonable indication on how long the satellite will remain in orbit.

5.2 Orbital simulation of RUFS-1

In the final simulation there are a number of different initial parameters that affect the sought after performance results. For the orbital life time there are two deciding factors: the mass of the satellite and the atmosphere. Since only the
maximum and minimum outcomes are desired only two
mass inputs will be used, the mass of the basic kit and
the upper limit allowed by the launch provider which is
one kilogram. This creates six different initial conditions.
With the radio performance, on the other hand, inclination
starts coming in to play. At first glance this might seem
like the needed initial conditions now greatly increases but
that is not the case. A longer or shorter orbital life time
will essentially only have the effect of either lengthening or
shortening the radio plot. Since the assumed ground station
is in northern Sweden, the relative amount of radio time
will be roughly the same (the ratio between total mission
time and total radio time). Thus only the initial parameters
with the longest total mission time will be run with different
inclinations.

5.2.1 Orbital life time
As was stated in the previous section all the different desired
performance characteristics are dependant on specific initial
parameters, with only two factoring in for orbital life time.
The setup used for this: two mass levels, one at 0.65 [kg] and
one at 1 [kg]. Each of them run three times, one for every
atmospheric intensity. All of the initial orbital characteristics
will be the same, as the ones seen in section 3.3, with the
time of launch set for Q1 at 6 am.

Table 8 shows the six different outcomes for the orbital
simulation with the corresponding plots placed in appendix
A.5 as figure A.13 & A.14. The results clearly show that with
higher mass and lower solar intensity the longest duration
orbits happen, which was the expected outcome. The most
likely outcome for the satellite is between the results for
low and medium intensity. This was discussed in section 5.1 in regards to the validity of the results depending the
chosen solar activity setting. It also appears that the sun is
in a transition towards lower activity in the coming years
(figure 11) which shifts the most probable outcome more
into to the results shown with low intensity. Although the
highest results should still be seen as an upper “ceiling” for
how long the satellite will remain in orbit.

5.2.2 Radio time
The main controlling component for radio time is the timing
between Earths rotation and RUFS-1’s orbital position. That
means inclination might start to play a part in the outcome
but factors effecting the duration of the orbit will most likely
not. The reason for this is that the duration aspect will only
length (or shorten) the plots, they will not have an impact on
the general behavior of orbit. The orbital period will change
but the characteristics will still be dependant primarily
on Earths oblateness. Since the launch site is known, the
desired argument of perigee and both targeted apogee as
well as perigee are assumed to be met, only variations in
inclination are to be tested. Initial parameters are the same
as the ones used in section 5.2.1 apart from the atmosphere
and mass will be set at “low” and 1 [kg] respectively. The
inclinations tested are: 85, 87.5, 90, 92.5 and 95 [deg].

<table>
<thead>
<tr>
<th>Inc</th>
<th>85</th>
<th>87.5</th>
<th>90</th>
<th>92.5</th>
<th>95</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ratio [%]</td>
<td>0.78</td>
<td>0.78</td>
<td>0.78</td>
<td>0.77</td>
<td>0.80</td>
</tr>
<tr>
<td>Time [h]</td>
<td>7.5</td>
<td>7.5</td>
<td>7.6</td>
<td>7.4</td>
<td>7.7</td>
</tr>
</tbody>
</table>

Table 9 shows that there is very little variation between
the different test runs. Both the ratio of radio time relative
to total mission time and the total radio time in hours show
a similar outcome far all 5. All have a ratio of less than 1
percent.

5.2.3 Average power and temperature variation
Here both the variation in temperature and the available
power are covered together, the reason for this is that these
two are inherently linked. As was shown in section 4.4 essentially the same equation is used for calculating the
governing factors for this simulation run will be the time
of launch. RUFS-1 has a targeted launch date of either the
first or second quarter of 2016 (Q1 and Q2 respectively) but
the time of day is unknown at this point. Also the two mass
variants of the satellite will be tested. The simulation run,
with a total of 16 different initial conditions are shown in
table 10.

<table>
<thead>
<tr>
<th>Yearly quarter</th>
<th>6 am</th>
<th>8 am</th>
<th>10 am</th>
<th>12 am</th>
</tr>
</thead>
<tbody>
<tr>
<td>Q1</td>
<td>0.65</td>
<td>1</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Q2</td>
<td>0.65</td>
<td>1</td>
<td>0.65</td>
<td>1</td>
</tr>
<tr>
<td>Q3</td>
<td>0.65</td>
<td>1</td>
<td>0.65</td>
<td>1</td>
</tr>
<tr>
<td>Q4</td>
<td>0.65</td>
<td>1</td>
<td>0.65</td>
<td>1</td>
</tr>
</tbody>
</table>

Only a quarter of the day is looked upon because form
the orbits perspective it is all about the angle between the
orbital plane and the sun vector. Because of the assumed
orientation of the satellite together with its design the
results occurring at 8 am, for example, can be assumed
to be equal at 2 pm, 8 pm and 2 am. With regards to the
yearly quarter (Q1 and Q2 shown in the table) here the
two chosen are based on a similar assumption. The suns
angle above (or below) the equator varies throughout
the year but from the satellites perspective there is no
difference between 15 degrees below or 15 degrees above
the equator. Hence only two quarter periods are looked
upon. All the other parameters are the same as in section 3.3.

The upper limit for the allowed temperature onboard
the satellite is set at 60 [C] based on the limits of existing
electronics (reference [1] through [5]). The lower limit is set
at −10 [C] because of the battery’s limits [5]. For available
power, the average power output from the solar panels will
be looked upon. The code average the power for every
saved time interval (which is set at 300 [s] mentioned in
section 4) and this value is plotted (which can be seen in
appendix A.6). In this section the average power available
of the total orbital life time is presented.
Table 11 shows both the maximum and minimum temperature for all of the tested orbit variants. The one set for launch at 12 am is the one that breaks the lower temperature criteria for all initial condition variants (time of year and launch mass) with the 10 am launch failing only when the lowest launch mass is used. For all of the initial conditions the upper limit is never broken, only reaching an overall maximum of 40.75°C. As seen in the plots in appendix A.6, some of the orbits happen to have periods where the satellite has constant sun light. The most likely reason for this is that the 12am group to fail is that the smallest surface area of the satellite is hit by the sun with two of the sides being the front and back panel. These two, as was stated in section 3.1, are fitted with polished aluminium which has a lower thermal absorption rate than the solar panels. This decreases the amount of heat received from the Sun and is most likely the cause of the lower onboard temperature.

The average available power, as seen in table 12, varies greatly depending on the time of launch. It has a highest value, 1.28 [W], occurring on the 8 am for Q1 with a launch mass of 1 [kg]. The lowest, 0.56 [W], is with the 12 am Q1 with a launch mass of 0.65 [kg]. These results coincide with the temperature results (although the maximums narrowly miss one another). This goes back to the long standing problem within spacecraft design: if one desires a higher average power the design has to be able to handle a higher operating temperature or a cooling systems have to be brought along. A cooling system that might bring issues with the mass will have to be brought along as well. The total power needed for both situations are calculated by:

\[ P_{sum} = \sum_{i=1}^{N} I_i \cdot U_i \] (23)

Only the consumption of transceiver, amplifier and onboard computer (Arduino controller) will be looked upon here. The budget will show if there is enough margins to allow a third party payload onboard. All the specifications are taken from their respective fact sheets.

The budget will show if there is enough margins to allow a third party payload onboard. All the specifications are taken from their respective fact sheets.

Table 13 and 14 show the needed power for transmitting \( (P_t) \) and reception \( (P_r) \) respectively. With the ratio for LOS to ground station \( t_{los}/t_{tot} \) already presented in section 5.2.2 it is now possible to calculate the total average power consumed onboard the spacecraft.

\[ P_{tot} = P_t \cdot \frac{t_{los}}{t_{tot}} + P_r \cdot \left( 1 - \frac{t_{los}}{t_{tot}} \right) \] (24)

If the ratio is assumed to be 0.008 (which is rounded upwards from the data shown in Sec. 5.2.2) the total power consumed becomes 0.355 [W]. With the now known average power output the margin is very simple to calculate. The margin is the average available power subtracted with average power used.

Table 15 shows the margin of the power budget. Here the outcome naturally is similar to the one seen in section 5.2.3 in regards to maxima and minima. The important difference is that the margins stays positive for all orbits. This shows that from a power availability perspective a third party payload and/or camera are possible options.

### 6 Power Budget

The power budget is built upon the average power available by the solar panels and the assumed general behaviour of RUF-1. The assumed behavior is based on the expected tasks that the satellite will perform during its mission. As was stated in section 3, it is assumed that RUF-1 will be transmitting every time it has line-of-sight (LOS) of the ground station. What remains is the question of what is RUF-1 doing for the remainder of the orbit. Here it is assumed that the spacecraft will shift to “reception mode” which means that both transceiver and amplifier will be using less power. The total power needed for both situations are calculated by:

\[ P_{sum} = \sum_{i=1}^{N} I_i \cdot U_i \] (23)

Only the consumption of transceiver, amplifier and onboard computer (Arduino controller) will be looked upon here. The budget will show if there is enough margins to allow a third party payload onboard. All the specifications are taken from their respective fact sheets.
7 Radio Performance

A satellite’s radio performance is an essential part of any mission study (if not the most important one after celestial mechanics). For a satellite this performance evaluation consists of two key aspects: how often can it be observed from the ground station and how good is the communication link. The means to which this study calculates the line of sight has already been described in section 5.3 and how this is used for determining the total radio time together with its results is covered in that section. The link budget on the other hand is a separate procedure and is not directly connected to the time dependent orbital performance of the system (although the range between the ground station and the satellite is one key element in the budget).

7.1 Link budget

The link budget is separated into two parts, up-link and down-link, with each one consisting of a ground, space and on-board segment with a link performance summary in the end. All of this is done in decibels. In the budget included in Appendix B the performance summery is named the Eb/No-method which is the technique used for evaluating if the radio link fulfills the desired performance with enough margin [13]. The segments are set in the order of transmission. That means the basic principal for organizing the aspects of a radio link. The first part is the broadcaster, next is the medium propagation and lastly the receiver. Both RUFS-1 and the ground station are looked upon in the broadcast as well as the receiver segments since both will be acting in those roles. Thus for the down-link the order becomes: on-board segment, space segment, ground segment and finally the performance summery. For the up-uplink, ground and on-board segment switch places.

Through out the link budget the estimated ground station will be used, the reason for this is covered in section 3.2. It is based on [14] and all the parameters for the ground segment are taken from there unless stated otherwise.

7.1.1 broadcaster

The broadcaster performance is dependant on four characteristics of the system. Transmitter power and frequency, on-board losses and antenna gain. These need to be evaluated for both the ground station and the satellite. This is done by the EIRP (Equivalent isotropic radiated power) equation.

\[
EIRP = P_T - L_c + G_a
\]

(25)

With \( P_T \) being the transmitter power, \( L_c \) is the onboard losses, \( G_a \) is the antenna gain and again this is done in decibel. Since transmitter power is often listed in Watts it needs to be converted to decibels, which is done by:

\[
dBW = 10 \cdot \log_{10} \left( \frac{\text{power out}}{1 \text{ W}} \right)
\]

(26)

The antenna gain value for the spacecraft is taken from [18] which is set at 2.15 for the spacecraft and 16.3 for the ground station with its antenna being switched from a dipole array type to a dish setup but with the same gain as seen in [14]. "Total Transmission Line Losses" (\( L_{\text{TL}} \)) is estimated to be relatively small for the satellite (0.1 [dB]) since the physical distance between amplifier and antenna is very short. For the ground station, on the other hand, it is set at 0.5 [dB] partly because the distances will be longer while at the same time more precautions can be taken to keep the losses low.

7.1.2 Medium propagation

For medium propagation there are only losses. Some of these losses depend on the frequency of the transmission whilst others are orientation dependent losses. All losses are assumed to be at their greatest unless stated otherwise.

\[
\text{FSPL}(\text{dB}) = 20 \cdot \log_{10} \left( \frac{4\pi df}{c} \right)
\]

(27)

Equation 27 calculates the free space attenuation loss (FSPL) in decibel. It is dependant on two main factors: the frequency (\( f \)) as well as the distance (\( d \)) between transmitter and receiver with \( c \) being the speed of light. Since the losses will be higher if the distance is increased, the maximum distance is used. This distance is the absolute value of the vector \( \vec{r}_{ls} \) seen in figure 7 when the angle above the horizon is exactly five degrees. The value used for this is listed as maximum distance in table B.1 and B.2. Ionosphere and atmospheric loss are taken from [13] based on link frequency.

The pointing loss is the change of reception between the two antennas. To accurately asses it for the satellite, experiments need to be done with a mock up of RUFS-1 radio system. For this study an equation is used that estimates the loss depending on the satellite’s roll (\( \sigma_{\text{roll}} \)) angle relative to the radio beam. This is the angle of rotation around \( \vec{e}_r \) seen in figure 1.

\[
P_{ls} = 10 \cdot \log_{10} \left( 1.5 \sin^2(\sigma_{\text{roll}}) \right)
\]

(28)

In equation 28, \( \sigma_{\text{roll}} \) is estimated to be 45 degrees. The equation itself is taken from [15]. For the ground station the pointing loss is more a question on the quality of the actual mechanical design since a typical station will track with a moving antenna. Here the loss listed is more of a caution and because of that, is an estimated value.

Polarization loss is taken as the maximum potential loss between a dipole antenna and a circular receiver (and vice versa) [16]. All of these losses (including the total decibel level from the previous EIRP) are summarized as “Signal power at antenna”.

7.1.3 Receiver

On the receiver end the first three listed entries in the budget have already been covered in section 7.1.1 for antenna gain and \( T_{\text{TL}} \) as well as section 7.1.2 for pointing loss. The three remaining, namely “Effect Noise Temperature”, “Figure of Merit” and “Signal-to-Noise Power Density, are covered here.

“Effective Noise Temperature” (\( T_s \)) for the ground station is calculated by

\[
T_s = T_0 \cdot N_F + T_g
\]

(29)

with \( T_0 \) being “standard temperature” (290 [K]), \( N_F \) being “system noise factor” (2 [dB]) and \( T_g \) being “galactic noise
temperature” (84 [K]). For RUFS-1 the noise is estimated to be 372 [K]. This is then used for the “Figure of Merrit” (G/T) which is calculated for both receivers by:

\[ \frac{G}{T} = G_a - T_{ll} - 10\log_{10}(T_s) \]  

(30)

which is then summarized together with the “Signal Power at Antenna” and Boltzman constant (in decibel K = −228.5) as the “Signal-to-Noise Power Density” in equation 31.

\[ \frac{(S/No)}{dB} = P_s + G/T + K \]  

(31)

### 7.1.4 Performance summary

There are several ways of doing a performance summary with EB/No-method being the one used here. It compares the S/No with the “System Desired Data Rate” and “System allowed error bit rate” in order to see if the data link has enough margin to sustain a good connection with the ground station without losing too many data packages.

The “System Desired Data Rate” is the sought after connection speed between the spacecraft and ground station. As seen in [1] the maximum bit rate (R) for the radio is 5 [kbps] and in order to achieve that speed the margin becomes:

\[ E_{db} = 10\log_{10}(R) \]  

(32)

For estimating the error bit rate a function from MatLab called “beawgn.m” is used. It returns the value for various modes of modulation. As seen in [1] the modulation for the tr2m transceiver is a so called “Frequency Shift Key” (FSK), but the modulation order is up to the designers. For a bit error rate of 10⁻⁵ and a modulation order of 4 the required margin becomes 10.5 [dB]. The final entry is the implementation loss which is taken from [13] and is essentially a cautionary extra margin in order to be safe that all the criteria are met. The error bit rate, desired data rate and implementation loss are summarized and subtracted from the previous determined (S/No).

With all of these different gains and losses now determined the total system margin for both the up-link and down-link are 36 [dB] and 11 [dB] respectively. There is no industry standard for how much margin is needed in order to have a good data link. The margins calculated here are high enough so that even with some extra interference the satellite can still be communicated with. One thing that can not be stressed enough is the fact that a lot of entries in the linkbudget are estimated or simplified. This is a preliminary result which gives the indication that the satellite will be able to sustain contact with the ground station as long as the LOS requirement is met. But further studies should be performed in order to fully be certain that the link margin is held at positive value.

### 8 Risks

The risks involved with a satellite project are numerous and varied. Here they have been broken down into three categories based on the stages that RUFS-1 will be in during this project: Pre-launch, Launch and In-orbit. Each of the areas have their own unique challenges which will be discussed further on. But one underlying fact is that a successful satellite project is built upon thorough planning and thorough testing.

#### 8.1 Pre-Launch

- The main risks involved with the project before the launch are mostly linked to the organisation of the project. Since this project is purely done on a voluntary basis, the chance that someone of the key people involved either not having enough time or entirely dropping out is higher than a typical industry project. To mitigate this a more thorough planning is needed with as many of the involved parties engaged and informed early as possible on the overall progress of the project.

- Another risk is severe delays of various types that start pushing the testing phase too close to launch. These could be for example failure with the vibration tests. This will most likely have the effect of increasing the risk of failure either during the launch or in orbit. Again here careful planning is the key. By adding extra time between different phases of the project buffer zones are created that will lessen the loads on the schedule, keeping it on time.

- System integration problems is also a risk. This can be due to either parts not being available, not functioning with the planned system layout or getting broken by poor handling. To mitigate the possibility of parts not being available, close communication with the suppliers is needed in order to get early information on the status of key parts. Good pre-research and the use of already tested components will lessen the likelihood of parts mismatching. Broken components can be avoided through careful handling and stocking on spares when possible due to budget.

#### 8.2 Launch

- One of the greatest and most obvious risk in the entire project is the launch rocket. The system chosen for this is not yet fully operational. As of this writing only one basic suborbital test flight has been preformed. There is a risk of it never flying or exploding during the inaugural flight. These risks are very hard to control, let alone lessen their likelihood. One possible way is to build a second satellite as back up but budgetary constraints might not allow it. An alternative launch provider is also an option if the rocket program suffer failure before RUFS-1’s scheduled launch. Although that would most likely push the entire timetable back several months, if not years.

- The second major risk during the launch phase is the sustained loads and vibrations suffered whilst reaching orbit. These could either inflict minor damage on for example a non vital subsystems. Making the satellite not responding to commands
correctly. Major damages would be for instance an entire system or component of the satellite completely breaking and endanger the entire mission. One way of preventing this is to get good easements of the loads and vibrations of the launch and designing the satellite with these criteria in mind. To confirm that the design can withstand the challenge of launch, testing needs to be done through a centrifuge and a vibration platform.

- The rocket not reaching the desired orbital parameters is the last major risk with the launch. The main issue here is that when the final burn has been performed by the upper stage and the carried satellites have been jettisoned, there is no way for RUFS-1 to alter its orbit. The only option is to change the mission profile to suit the new orbit where possible. For instance, if the orbit has a different inclination than originally planned, new uplink times has to be calculated. Preventing something like this to happen is very hard and completely out of the RUFS-1 satellite project’s hands. Again, a different launcher with a proven record of reliability could be chosen instead but as stated earlier it could delay the project considerably.

8.3 In orbit

- When the satellite has reached orbit it will be in a very hazardous environment. Radiation, space debris, magnetic flux, temperature; all of these things could cause major system failure and lead to total mission loss. Each of them have their own different way of effecting the satellite and are dealt with accordingly. How this is done is a report all on its own, but what they all have in common is that they are all about preparatory work. Testing for magnetic flux, thermal variation, radiation and so on has to be done in order to be sure that RUF-1 will survive its time in orbit. Nothing can be done if these issues are discovered once it has reached space.

- There is also a risk for software failure. The consequences could range from something like the satellite not responding but still sending information to a complete loss of communication. This could be caused by a multitude of different things but it can be summarised into two major categories: failure due to programming and failure due to hardware malfunction. Preventing hardware malfunction has already been covered in previous segments but failure due to programming is equally serious. The main way of insuring that it does not happen is to perform several tests on the ground including one or more full mission simulations. With programming failure there still remains the possibility of reprogramming the satellite while it is in orbit. This is a highly circumstantial solution and should not be seen as a strong possibility.

- Another risk that needs to be handled is the potential failure of the ground station. Since ground stations are a combination of several subsystems the ways of which it may fail are equally numerous. The fallout of a failure during the ongoing mission is much less volatile compared to many of the previous stated events. As long as the failure can be traced and fixed with a reasonable amount of time, the consequence is only a loss in the total orbital radio time.

9 Conclusion & Discussion

From an overall technical perspective the mission is feasible. All the budgets done in this study shows enough positive margins in order to say that both the communication system and power available are sufficient to fulfill the mission requirements. There is however one aspect that needs to be further examined and that is the temperature variation. With some of the initial conditions used for the simulation run, the minimum temperature dipped below the allowed levels. The thermal study is very basic and needs to be done on a more thorough level in order to give a conclusive answer whether or not the satellite would “freeze up”. Since the initial conditions are such a heavily controlling factor for the outcome, close communication with Interorbital Systems must be held. When the final intended orbit has been announced a better examination of that thermal environment can be done. The link budget is also something that needs to done in greater detail. When a ground station has been found/chosen and an initial build of the satellite is completed, many of the assumed or estimated losses present within the budget can be changed to known.

With regards to third party payloads, here again the margins are shown to be good enough to allow them. The power budget especially shows that even during the worst conditions there is power available for some additional cargo (albeit half of a 10x10 [cm] PCB). Because of the longer orbital time achieved with the heavier satellite extra weights will be placed onboard to increase the mass to one kilogram. If too many issues start to arise from integrating extra batteries onboard RUFS-1, the extra mass will be added in the from of steel plates behind the front and back polished aluminum sheets.

10 Summary

All the various methods used in this study show that RUFS-1 will be able to fulfill the mission requirements. The orbital simulation was used for assessing the orbital life time, available power and temperature variation. A link budget was also calculated based on the orbit and the onboard radio system. All of them showed favorable results except under a few specific circumstances. For instance the temperature simulation showed that depending on the initial orbit RUFS-1’s internal temperature would drop below the desired levels. The methods used for some of the results present are a based on assumptions that simplify the situations and
events. Further examinations of both the onboard temperature and link-budget is needed but the overall conclusion is that RUFS-1’s mission is feasible.

**References**

[1] TR2M-436.50-10-ARS Radiometrix Multi Channel Transceiver (http://www.radiometrix.com/node/187) [2015-03-26].

[2] AFS2-436 Radiometrix Amplifier (http://www.radiometrix.com/content/afs2-0) [2015-03-26].


[5] Li-ion 18650 3.7 5200mAh Rechargeable PCB Protected Battery Pack Tenergy (http://www.tenergy.com/31001) [2015-03-27].


**Appendix A**

**Result plots**

**A.1 Molniya orbit results**

Fig. A.1: Altitude (above Earth) and speed for Molniya orbit as a function of time.

Fig. A.2: Apogee and perigee for Molniya orbit as a function of time.

Fig. A.3: Change of RAAN and inclination for Molniya orbit as a function of time.

**A.2 Sun synchronous results**

Fig. A.4: Altitude (above Earth) and speed for sun synchronous orbit as a function of time.

Fig. A.5: Apogee and perigee for sun synchronous orbit as a function of time.

Fig. A.6: RAAN and inclination for sun synchronous orbit as a function of time.
A.3 Ardusat-1 results

**Fig. A.7**: Altitude and speed for Ardusat-1 with atmosphere set to low intensity.

**Fig. A.8**: Altitude and speed for Ardusat-1 with atmosphere set to medium intensity.

**Fig. A.9**: Altitude and speed for Ardusat-1 with atmosphere set to high intensity.

A.4 F-1 results

**Fig. A.10**: Altitude and speed for F-1 with atmosphere set to low intensity.

**Fig. A.11**: Altitude and speed for F-1 with atmosphere set to medium intensity.

**Fig. A.12**: Altitude and speed for F-1 with atmosphere set to high intensity.
A.5 Orbital life time

Fig. A.13: Altitude and speed for all three atmospheric settings (low, medium and heavy intensity) with a satellite mass of 0.65 [kg].

Fig. A.14: Altitude and speed for all atmospheric settings (low, medium and heavy intensity) with a satellite mass of 1 [kg].

A.6 Available power

Fig. A.15: Power, 6 am, Q1 & Q2, RUFS-1 m= 0.65 & 1 [kg].

Fig. A.16: Power, 8 am, Q1 & Q2, RUFS-1 m= 0.65 & 1 [kg].

Fig. A.17: Power, 10 am, Q1 & Q2, RUFS-1 m= 0.65 & 1 [kg].

Fig. A.18: Power, 12 am, Q1 & Q2, RUFS-1 m= 0.65 & 1 [kg].
A.7 Temperature Variation

Fig. A.19: Temperature, 6 am, Q1 & Q2, RUFS-1 m = 0.65 & 1 [kg].

Fig. A.20: Temperature, 8 am, Q1 & Q2, RUFS-1 m = 0.65 & 1 [kg].

Fig. A.21: Temperature, 10 am, Q1 & Q2, RUFS-1 m = 0.65 & 1 [kg].

Fig. A.22: Temperature, 12 am, Q1 & Q2, RUFS-1 m = 0.65 & 1 [kg].
## Appendix B

### Link Budget

<table>
<thead>
<tr>
<th>Table B.1: Down-link</th>
<th>Symbol</th>
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