CONCEPT OF OPERATIONS

This document describes the Concept of Operations for the Signal of Opportunity CubeSat Ranging and Timing Experiment System (SOCRATES) cube satellite. This document draws heavily from the Mission Overview and Requirements Verification Matrix (RVM) to outline the on-orbit operations and timeline SOCRATES will undergo during its mission in low Earth orbit.
I. Introduction

SOCRATES’s primary mission is to measure astrophysical sources with high-precision time tagging that will allow the instrument to test and demonstrate the concept of spacecraft relative ranging using gamma ray burst timing. SOCRATES will demonstrate the ability to act as a network of satellites that can communicate amongst themselves to determine the relative position from each other. This will be done by comparing the time of arrival for the photons that radiated from the astrophysical sources to determine the relative distance between two satellites. There need to be corrections for the clock offset of the different satellites involved, which is included by our project scientist.

The secondary mission of SOCRATES is to measure the time of arrival of photons, at a known spacecraft position, from astrophysical sources (e.g. solar flares, gamma ray bursts, pulsars) to demonstrate new spacecraft ranging techniques. The secondary mission of SOCRATES is to measure X-ray photon energies and arrival times to study electron acceleration in solar flares, which are the drivers of space weather events (e.g. coronal mass ejections).

SOCRATES will investigate the energy release processes and particle acceleration mechanisms that drive solar flares and coronal mass ejections (CMEs), events that are collectively referred to as solar eruptive events and are the sources of the most extreme space weather. The plasma and solar energetic particles (SEPs) ejected in these events, when Earth-directed, pose radiation risks to spacecraft and astronauts, and in extreme cases could endanger the Earth’s power grid. While the effects of these solar eruptive events are starting
to be well characterized, the acceleration mechanisms for the high-energy particles produced are not yet understood.

SOCRATES will measure flare-accelerated electrons from flares in the declining phase of Solar Cycle 24 and will serve as a pathfinder for a set of simple, inexpensive, hard X-ray flare monitors. These monitors will operate in a similar fashion to the soft X-ray monitor on the NOAA GOES (Geostationary Operational Environmental Satellite) set of spacecrafts, providing continuous monitoring of high-energy flares and better predictive capabilities for extreme space weather events by aiding assessment of CME accelerations and the probability of strong SEP events. With serendipitous co-observations, SOCRATES can also perform groundbreaking new science in the study of solar flare-accelerated electrons, helping to understand the basic generation of the powerful solar sources of space weather. While full investigation of flare particle acceleration will require advanced, expensive instruments with hard X-ray (HXR) imaging capabilities, there is still a useful role for a series of HXR flare spectrometers that can monitor flares to amass statistics on flare-accelerated electrons and serve as an early warning to the presence of SEPs at 1 AU. Consistent, unbroken solar HXR monitoring can be achieved using replicated CubeSats that are developed and replaced throughout the years, much like the larger, more complex NOAA GOES satellites that include instruments to measure lower-energy, “soft” X-rays from the Sun.
Therefore, it is a dual-use sensor that can serve as a relative position, navigation and timing instrument for GPS/GNSS-denied satellite operations, independent of its solar purposes.

Additional details regarding the SOCRATES mission can be found in the Mission Overview document which describes mission objectives, success criteria, and requirements to meet these objectives. For details regarding the RVM referenced requirements, either see the RVM itself or Appendix B of this document.

II. Mission Development

SOCRATES is a science-focused, 3U cube satellite which aims to collect significant amounts of data from astrophysical sources and events, primarily our Sun. It is imperative that SOCRATES’s detector remain sun-oriented for at least 50% of the mission (RVM, OO2-7) and maintain pointing within 25 degrees (RVM, ADCS-4). The orbit requirements come from the requirement to be in range of our ground stations (RVM, FR-10). This corresponds to an inclination at or above 35 degrees. To maintain a sun-facing detector, SOCRATES shall be 3-axis controlled and 2-axis stabilized (RVM, ADCS-1, ADCS-2). In addition, since there is only 1U area of the satellite body facing the sun, deployable solar panels are essential to meet power requirements (RVM, EPS-5). Additional information regarding the power draws and generation can be found in the SOCRATES power budget.

The data SOCRATES collects are the energy and arrival time of each individual photon. Since events from the Sun happen sporadically, SOCRATES’s data collection will occur in spurts; receiving and storing large amounts of data (100-1000 photons per second) for tens of
minutes at a time, then remaining quiet, collecting only background counts, for long periods while awaiting flares or gamma ray bursts. These factors combined with further requirements outlined in the RVM shape the ConOps for SOCRATES which are described next.
III. Concept of Operations

As shown in Figure 1, the SOCRATES ConOps consists of there are 6 modes. Each mode composed of several smaller sub-modes which are referred to as states. Following ejection from the P-POD, the words “Undeployed” and “Deployment” are used to refer to the deployable solar panels. Before SOCRATES can begin any data collection, the solar panels
must be deployed to generate the power required to run the detector and supporting electronics. Details of powered components in each mode/state can be found in Appendix A.

On the highest level, the aim of the ConOps is to ensure the satellite goes through the proper steps to reach the Mission mode. This requires that all systems supporting the payload operate efficiently to move the satellite’s operational focus towards the Mission mode swiftly. The modes are defined either by hardware entry and exit criteria or operational environment changes whereas the states are software defined and occur within modes.

IV. Mode/State Definitions

In the definitions of each mode and state, the following will be described: The name; entrance and exit criteria; and function details including which components are powered on. Table A1 in Appendix A gives a summary of SOCRATES’s components and terminology used in this ConOps. Moving forward, modes will be denoted by **bold** text and states will be shown with *italics*.

1. Launch Mode

   **Launch** encompasses the stowed satellite being carried into orbit via the launch vehicle. Mode exit occurs when the satellite is ejected out of the launch vehicle deployer and the inhibit on the structure is tripped to begin powering on.
2. **Startup Mode**

   **Startup** can be entered in one of two ways: The first is immediately after launch vehicle ejection and the second from **Safe Mode** (defined below). The post-ejection entry criteria will be described first followed by entry from the **Safe Mode**.

   After the inhibit on the structure is triggered, the satellite will enter the **Safe Mode** in the *Power On state*. On-Orbit Operational Requirements UNP9-66 through UNP9-69 given in the UNP User’s Guide state a Radio-Silence Period and No Maneuverability Period of 45 and 15 minutes, respectively. Since SOCRATES does not require any attitude maneuvers until later in the mission, the exit criteria for *Power On* is simply being in orbit for more than 45 minutes. During this state, attitude determination sensors will be powered on (gyro, magnetometers, GPS) along with the flight computer. Throughout this mode, the sensors calibration process will begin in preparation for calculating the attitude solution before the attitude controller can be brought online. After 45 minutes passes the satellite exits *Power On* and moves to *Sensor Calibration*. In this state, the radios will be turned on and 2-way communication with the ground commences.

3. The second way in which SOCRATES can enter **Startup** is in the process of recovering from an entry into the **Safe Mode**. SOCRATES can enter the **SAFE MODE** as a result of various system anomalies. SOCRATES can enter this mode on its own via *Error Resolves* when onboard system detect anomalies. It can also enter **SAFE MODE** as a result of explicit uplink commands or watchdog triggers. If in this
mode, the satellite will downlink a status packet every 5 minutes indicating **Safe Mode** is initiated. The satellite will also be listening for any uplink commands, e.g. to reprogram the flight software. In *Mission Beacon*, sensors will self-calibrate and downlink status messages every 5 minutes regardless of a communication connection. To exit *Mission Beacon* requires a message from ground to re enter *Mission*. The reason for a response from ground is to ensure the downlinked packets contain the expected information before proceeding.

4. **System Check Mode**

**System Check** is defined based on whether the solar panels have been deployed or not. In an undeployed state, the panels remain retracted to the sides of the structure. The radio antenna is a patch antenna on the opposite end from the detector, so even whilst undeployed, the satellite has the capability of closing a link.

Operations during **System Check** begin with **System Calibration**. In this state, the satellite’s radio is powered on for the first time and downlinks status messages every 5 minutes. In addition, the attitude determination sensors that were calibrated send their measurements to the flight computer which, in turn, runs the attitude determination algorithm. There are three ways to exit **System Calibration**. First, when in **System Check**, if errors are occurring in the system, the satellite enters **System Error Resolve** where the components may be power cycled. The radio is still listening and sending status packets every 5 minutes. If the satellite’s power drops
below 25%, System Power Save will be initiated and the radio will cease further transmission and only be in a listening state.

Transition to the next mode from System Calibration requires that a message from the ground has been received to initiate solar panel deployment. The actual deployment does not occur in this mode, but a specific message must be received before entering the next mode.

5. Initialization Mode

Initialization is entered when the satellite is preparing itself to enter Mission. To exit this mode, the solar panels must be fully deployed and detector window facing the sun. To achieve this, three states are required within Initialization: Initialization Deployment, Initialization Maneuver, and Initialization Error Resolve.

After receiving a message from the ground in System Calibration, the satellite routes current to the hot wires which hold the solar panels parallel to the structure. After deployment, the panels are now perpendicular to the structure’s longest sides and in the same plane with the detector window. Throughout this process, the radio is still downlinking status packets every 5 minutes. Exiting this state occurs after current has been given to all hot wires and solar panels deployed. Verification of deployment can be done in one (or multiple) of three ways: 1. Touch sensors could be installed and output differently depending on whether the solar panel is in a deployed position or undeployed; 2. We can monitor the current coming from the panels and compare them to each other; 3. Prior to deployment, the satellite sends a rotation about the
z-axis and measures the current required to do so. After *Initialization Deployment*, the command to rotate about the z-axis is given again and current measured. These values are compared and we can calculate how many panels successfully deployed. None of these methods have been explored thoroughly yet.

*Initialization Maneuver* begins actuating the magnetorquers to orient the detector face towards the sun. Status messages continue downlinking every 5 minutes during this mode. Exit criteria for this mode is a sun pointing angle measured within the requirement (RVM, ADCS-4). Once that condition is met, the data collection may begin in **Mission** mode.

*Initialization Error Resolve* is a state entered when an error occurs during either panel deployment or attitude maneuvering. For example, consider the hypothetical scenario during deployment when a hot wire is not cut. Then *Initialization Error Resolve* would be initiated and either on-board software would be required to adjust the attitude solution, or the satellite may enter Safe Mode and require reprogramming from the ground. Otherwise, *Initialization Error Resolve* operates similarly to *System Error Resolve*.

6. **Mission Mode**

*Mission Default* is the default state for most of the mission. Without any response from the ground, the entrance criteria for *Mission Default* is that ADCS-4 is met. This is the first time during the mission where the Detector and FPGA are powered on as the satellite will begin sensing events from astrophysical sources. The radios will not
transmit during this state. The reason for this is RF noise concerns when the detector is collecting data. Data transmitting over the antenna may produce false detector triggers.

Entry to Mission Downlink can occur in a few ways. First, if the on-board storage is approaching its capacity, the flight computer determines based on currently commanded settings whether data collection must stop or if data can be overwritten. If collection must stop, then the satellite begins taking full advantage of any link that occurs with the ground. Alternately, this can happen autonomously if onboard algorithms (watching detector rates) determine that no meaningful data are being collected by the detector. In this situation if a link occurs between the ground station and satellite, data will be transmitted. This transmission would only continue if that specific link is available. The last way of entering this mode is via uplink from the ground. Based on solar conditions, predictions can be made for the probability of upcoming solar activity. If it is determined by the project scientist that the satellite will not likely see major events in a given time window, then an uplink command can be given to tell the satellite to focus on downlinking data without fear of interfering with data collection. The exit criterion is a ground command instructing the detector to continue operation and return to regular mode transitions. Otherwise, if data storage reaches acceptable levels, the satellite will autonomously return to Mission Default and continue gathering data.

We have an agreement to use Aerospace Corporation Ground Station Equipment to “send… and/or receive communication messages from satellites in Earth orbit”
(see Ground Support Design documentation, Satellite License Agreement, section 1.4). This agreement allows us to downlink to any Aerospace ground station that is available to receive data. A new Aerospace ground station is currently being constructed on University of Minnesota property in Ash River, Minnesota, which enabled this agreement with Aerospace Corporation (see pictures 1 and 2 below).

![Figure 1: Recent photo of the new Aerospace Ground Station in Ash River](image1)

![Figure 2: Receiver at Ash River](image2)

*Mission Power Save* state can be entered at any point and would include turning the Detector, FPGA, and data transmission off to allow the charging of the batteries. Power levels of 25% necessitate entrance to this mode and charge levels above 80% constitute the exit criteria.

Satellite attitude drift will occur over time and be corrected in *Mission Maneuver*. With pointing requirements on the SOCRATES satellite coupled with noise concerns to the detector, attitude control will be conducted intermittently. Based on sensor data and an attitude determination algorithm, a solution will be made. If ADCS-4 is not met, then this mode will be entered requiring the detector to switch off for corrections to SOCRATES’s attitude be made to sun-faced pointing. When the sensors calculate
the attitude and deem it is within requirements, the satellite will change modes accordingly.

*Mission Error Resolve* operates very similarly to *Undeployed Error Resolve* and is a potential entry point for **Safe Mode** in the event of hardware malfunction or a forced response from the ground is needed.

7. **End of Life**

Since SOCRATES is a science mission, the longer the satellite remains operational, the more data we can obtain. Due to the relatively large surface area of SOCRATES, the satellite’s orbit will, naturally, decay before the deorbit requirement provided in the UNP User’s Guide. As a result, this is our End of Life mode.

Upon reentry, SOCRATES does not have any volatile, hazardous, or dense enough substances that would survive reentry. It is assumed that all material aboard SOCRATES will incinerate in the upper atmosphere and not reach the ground.

V. **Timeline**

Time frames were not specified while defining entrance and exit criteria. In this example, only the nominal path in Figure 1 will be followed. The amount of time spent in the varying modes and states will be assigned based on the path depicted below in Figure 2.
Figure 2. Simplified ConOps showing estimated time spent in each mode.

The transition between **Startup** and **System Check** is a time-based requirement given in the UNP User’s Guide. Moving from **System Check** to **Initialization** depends on allowing enough time for the attitude determination sensors to calibrate. Once those sensors have finished, which is estimated to take approximately 1-2 complete orbits, the satellite will await a response from the ground before proceeding to **Initialization**.

Upon entering **Initialization**, the panels will be deployed and magnetorquers initiated to orient the satellite’s detector face towards the Sun. The time it takes for the transition to **Mission** depends on the rotation rates the satellite must overcome to stabilize its pointing. It is estimated that this will take between 7 and 10 orbits which equates to around 10 – 16 hours assuming similar orbital speeds to the International Space Station.

As mentioned above with SOCRATES being a data gathering mission, the more time in orbit with the sensor, the better. However, to justify minimum orbit
requirements and establish the time necessary to accomplish our maximum success criteria (RVM, M1C-1), analysis was conducted. This analysis consisted of expected solar flares to occur during a window in which SOCRATES may launch and collect data, based on flare rates in the present declining solar cycle as well as the same phase of the previous solar cycle. Results are shown below in Figure 3. Taking the worst-case scenario for a launch around April 2019, we arrive at the conclusion: in order to accomplish our maximum success criteria of characterizing 26 solar flares, SOCRATES shall be designed to collect data for at least 16 months. Launch opportunities must take this mission length into consideration.
Figure 3: Three different plots depicting mission length required to meet minimum and maximum success criteria as shown in RVM, M1C-1. X-axis notation given as month, year. For example, Apr 19 indicates a launch date in April of 2019. Predicted rates are based on flare rates at a similar phase of the previous solar cycle and are scaled to the relatively lower intensity of the present solar cycle. Because data are based on actual observations of the previous cycle, month-to-month variations (e.g. changes between March and May 2018) are not significant as they are based on highly variable activity, but year-to-year changes (i.e. long-term trends) are significant.
## Appendix A

<table>
<thead>
<tr>
<th>Component</th>
<th>System</th>
<th>Summary</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>Detector</td>
<td>This is the sun-facing detector that is encased within its own structure and requires specific mounting. This system measures an analog signal to be read by the FPGA.</td>
</tr>
<tr>
<td>Field Programmable Gate Array (FPGA)</td>
<td>FPGA</td>
<td>Processes a digitized version of the detector’s pulse output. The FPGA takes a series of 0.2µs samples and compresses the pulse data into 3 parameters which are transferred to the flight computer for downlink.</td>
</tr>
<tr>
<td>Flight Computer</td>
<td>CDH</td>
<td>Takes data from the FPGA and ADCS components. ADCS data is used to determine an attitude solution and output necessary commands for attitude correction. Flight computer assembles data packet to be sent to ground via Radio.</td>
</tr>
<tr>
<td>Gyroscope</td>
<td>ADCS</td>
<td>One of four components used to compute attitude solution.</td>
</tr>
<tr>
<td>Magnetometers</td>
<td>ADCS</td>
<td>Second of four components for attitude solution.</td>
</tr>
<tr>
<td>Sun Sensors</td>
<td>ADCS</td>
<td>Third of four components used for attitude solution. The sun sensors are used as a determination validation that the detector is sun-pointing.</td>
</tr>
<tr>
<td>Magnetorqueurs</td>
<td>ADCS</td>
<td>Attitude control component.</td>
</tr>
<tr>
<td>GPS</td>
<td>ADCS</td>
<td>The last component used for the attitude solution. The pulse per second from the GPS is also vital to the mission as the time of arrival of incoming photons must be precise.</td>
</tr>
</tbody>
</table>
Radio COM Communications radio with both uplink and downlink capabilities. Operates at 915 MHz.

Power Manager EPS Includes both the batteries and management board. This system takes in the solar panel generated power and converts it to 3.3V and 5V over the CAN bus rail.

Variable Power Supply EPS This supply takes the output from the power manager and converts to various voltages necessary to operate the FPGA and Detector systems.

<table>
<thead>
<tr>
<th>RVM Reference</th>
<th>Requirement</th>
</tr>
</thead>
<tbody>
<tr>
<td>OO2-7</td>
<td>The orbit shall allow the spacecraft to be in sunlight for 50% or its time in orbit.</td>
</tr>
<tr>
<td>ADCS-4</td>
<td>SOCRATES shall be able to control its orientation using magnetorquers.</td>
</tr>
<tr>
<td>FR-10</td>
<td>The CubeSat shall be able to receive and transmit from and to the ground station</td>
</tr>
<tr>
<td>ADCS-1</td>
<td>SOCRATES shall be three-axis controlled</td>
</tr>
<tr>
<td>ADCS-2</td>
<td>SOCRATES shall be two-axis stabilized with controlled spin about sun-face</td>
</tr>
<tr>
<td>EPS-5</td>
<td>The EPS shall deliver voltage and current to all subsystems.</td>
</tr>
<tr>
<td>M1C-1</td>
<td>Minimum Success Criteria: SOCRATES shall produce an energy spectrum between 10 keV to 100 keV with a resolution of 25 keV for 1 solar flare</td>
</tr>
</tbody>
</table>
Maximum Success Criteria: SOCRATES shall produce an energy spectrum between 10 keV to 100 keV with a resolution of 10 keV for 26 solar flares

Table B1: Referenced requirements from SOCRATES’s RVM as they appear in order in the ConOps.