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Space Technology 5 (ST5) Spacecraft Modeling

A Major Qualifying Project Report: submitted to the Faculty of
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Abstract

The purpose of this project was to update an existing ST5 Simulink Radio Frequency (RF) Communications model to provide more accurate line of sight predictions for long term mission planning. The ST5 satellite consists of two omni directional antennas mounted in opposition to each other resulting in an area of overlapping antenna patterns. The composite RF transmission pattern exhibits a marked zone of destructive phase interference. This area is referred to as the Zone of Interference (ZOI) which results in insufficient link margins for reception. The modified RF model calculates the link margin taking into account the radiative pattern as a function of satellite attitude and a new orbit propagator will be utilized which is capable of maintaining accuracy for up to three months to aid in long-term mission planning. Finally, the output of the Simulink model provides black out dates and times for long term planning.

1 Project Introduction

1.1 Introduction

Space Technology 5 (ST5) is a division of the New Millennium Program (NMP), a NASA supported program established in 1995 with the ambitious goal of advancing space exploration through the development of advanced technologies. The NMP's primary focus is to conduct space flight validation of advanced instruments, spacecraft systems/subsystems and concepts of flight. The goal of space flight validation of these technologies is to eliminate risks to the user and promote hasty integration of these technologies into future space missions. A secondary focus of the NMP is to conduct earth-science data acquisition missions if the mission budget permits. The NMP has proposed to do the following:

1. Reduce the size/weight of spacecraft, thus reducing costs
2. Help spacecraft become "intelligent," able to think for themselves, to minimize support of a mission operations team
3. Enable significantly improved (a several generation leap) technical capabilities in future missions (NMP, 1995)

The NMP first generation missions were designed to provide a comprehensive, system-level validation of high-priority spacecraft interaction and measurements. NMP first generation missions include Deep Space 1 (DS1), Deep Space 2 (DS2) and Earth Observing 1 (EO1). The NMP second generation missions were designed to make greater use of a constellation of satellites as well as system validations. NMP second generation missions include Earth Observing 3 (EO3) and Space Technology 5 (Crisp, Minning 2000).

1.2 Space Technology 5

The goal of the ST5 program is to create and test new components and technology that will provide breakthroughs in performance, capabilities and applications and that can be applied to future small-satellite missions. Figure 1 shows a diagram of the ST5 small-satellite (small-sat) with dimensions.

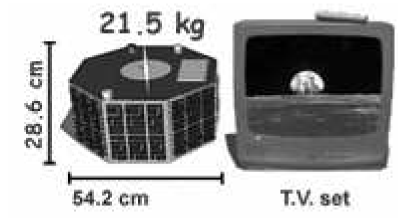

Figure 1: ST5 Satellite
(nmp.jpl.nasa.gov/st5)

One of many reasons for creating smaller satellites is to simplify construction and reduce cost. Smaller satellites are much easier to manage, move, deploy, and also create new possibilities for launching methods. For example a small-sat can fit beneath a larger spacecraft and be launched as a secondary payload, often referred to as "piggybacking".

ST5 consists of three identical satellites weighing approximately 21.5 kg (47lbs), measures 54.2 cm across, and 28.6 cm in height. The new technology components being tested aboard the ST5 satellites include:

- *Autonomous ground station software* for scheduling and orbit determination, specifically designed for constellations of spacecraft
- An *X-Band Transponder* for satellite communications, which requires less than a quarter the voltage and half the power and weighs twelve (12) times less and nine (9) times smaller than proven technology
- *Advance Multifunctional Structures* used for electrical interconnects which will reduce cable mass by saving one (1) kilogram for every one-thousand (1000) connections

- A *Field Programmable Gate Array (FPGA)*, which is a new microelectronic device that eliminates environmental effects, latchup and uses twenty (20) times less power than proven technology
- A new *Variable Emittance Coating (VEC)* which is a thermal coating that emits internally generated heat to cool the spacecraft and in return absorbs heat when the spacecraft is cool
- A *Microelectromechanical Systems (MEMS)* chip, a member of the propulsion systems components that will provide fine attitude adjustments to the spacecraft while using eight and a half (8 ½) times less power and weighing less than half as much as proven technology
- A *Lithium-Ion Power System* that store two to four times more energy than current batteries and has a longer life span than proven technology

In addition to testing the small-sat technologies, ST5 satellites will pursue a scientific data collection mission as well. The ST5 satellites will be test flown through frequent changes in charged particles and magnetic fields in the earth's magnetosphere. While the earth's magnetosphere acts as a protective barrier against the sun's harsh solar rays, some radiative particles do enter the Earth's atmosphere. ST5 satellites will map the intensity and direction of magnetic fields within the inner magnetosphere which will allow scientists to detect the presence of electrical currents carried by energetically charged particles. By studying this region of the magnetosphere scientists will also uncover important information about solar events that disrupt communications, navigation and power systems of the spacecraft. Below in Figure 2 is an artist concept of the earth's magnetosphere.

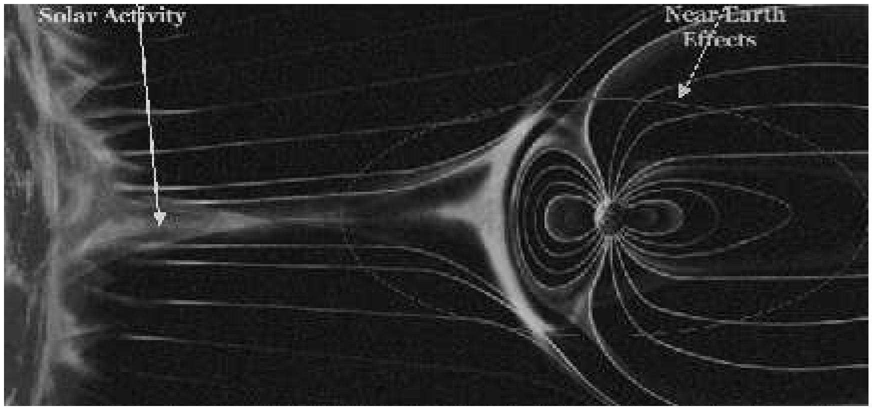

Figure 2: Magnetosphere Surrounding Earth Created by Solar Activity

(st5.gsfc.nasa.gov)

One way to plan for conditions such as the solar events in the Earth's magnetosphere is to provide model-based systems that can simulate multiple conditions of space flight and spacecraft functionality. SimulinkST5 will provide such simulations that can be used for mission planning purposes.

1.3 Problem Description

This project involved working with SimulinkST5 simulations that model the components of the ST5 satellite. The Simulink communications model needs to provide necessary data for the ST5 missions planning to determine communication capabilities. In order to determine the quality of the communications link, link margin, Doppler shift, and look angles must be calculated. These calculations are dependent on the position and velocity of the satellite at a particular time of interest.

The requirements to assess the ST5 communications link for SimulinkST5 can be separated into two sections.

1. Communications Model
2. Orbit Propagator Model

The ST5 satellite consists of two omni antennas mounted in opposition of each other. The composite antenna radiative pattern created by the two antennas creates an area of destructive phase interference. This area is known as the Zone of Interference (ZOI) and was accurately modeled by utilizing receiving antenna gain and line of sight calculations. The communications model was updated to account for spacecraft attitude and the antenna radiative pattern, as well as be able to determine ZOI occurrences and ground station line of sight.

The communications model is highly dependent on accurately knowing the position and velocity vectors of the spacecraft, which are in turn dependant on the accuracy of the orbit propagator model. The propagator must be capable of providing output vectors for long-term mission planning of up to 3-4 weeks. The orbit propagator must then be integrated to the communications model which is used to determine the communication link quality.

1.4 Summary

NASA's New Millennium Program has advanced space exploration by developing advanced technologies and integrating them into future spacecraft missions. Space Technology 5, a second generation spacecraft cluster being developed as part of the NMP, will be validating seven spacecraft technologies as well as the NMP second generation goal of testing the satellite constellation theory. ST5 will also be recording intensity and direction of magnetic fields within the Earth's inner magnetosphere as a secondary science objective to provide scientists with information about space weather that may disrupt communications, navigations and power systems of the spacecraft.

2 Background

2.1 Introduction

In order to fully understand the goals and objectives of the project, a brief overview of NASA, satellite orbits, communication variables including detailed and specific ST5 Communications model information as well as some key background information on link margin and telemetry will be discussed in this chapter. Lastly an overview of the MATLAB/Simulink software needed for SimulinkST5 modification will be discussed.

2.2 National Aeronautical and Space Administration

NASA was created on October 1, 1958, which aided U.S. space exploration. After the Sputnik crisis, NASA inherited the National Advisory Committee for Aeronautics (NACA) and other government agencies and initiated work on space exploration and human space flight. NASA began to conduct space missions within months of its creation and in its forty-five years has made historic achievements in many areas of aeronautics and space research (Garber, 2003).

By conducting cutting-edge aeronautics research on aerodynamics, wind shear, and related topics using wind tunnels, flight testing and computer simulations, NASA has continued to build on what the NACA started. The technical and scientific accomplishments of NASA demonstrate that humans can achieve what they never imagined (Dick, 2003).

NASA's aeronautics research has helped to enhance air transport safety, reliability, efficiency, and speed through such programs as the X-15, lifting bodies, and general aviation. NASA also contributes to Earth science missions, which deal with remote-sensing satellites such as Landsat and meteorological spacecraft. These missions have helped scientists understand the complex interactions between ecological systems on Earth (Garber, 2003).

2.3 NASA Goddard Space Flight Center

NASA Goddard Space Flight Center (GSFC) is located in Greenbelt, Maryland about 6.5 miles from Washington D.C. The GSFC is in charge of many of NASA's earth observations, astronomy and space physics missions. Most of the missions include developing and operating unmanned scientific spacecraft. GSFC also owns other properties outside of Greenbelt, with the most recognized site being the Wallops Flight Facility located near Chincoteague, Virginia.

2.4 Satellite Orbit Determination

In early times it was believed that planetary orbits were circular. In the 16th Century scientists began gathering data that dispelled this belief. Then in the 17th century a scientist named Johannes Kepler stated three laws of planetary motion which explained how each planet moves in an elliptical orbit, with the sun at one of its foci.

2.4.1 Kepler's Laws

German astronomer Johannes Kepler formulated three mathematical statements that accurately described the revolutions of the planets around the sun. These laws can also be applied to a satellite orbiting the earth. Kepler's Laws paved the way for the application of the laws of physics to the motions of heavenly bodies.

Kepler's first law, which is also known as the Law of Elliptical Orbits states,

"*Each planet moves in an elliptic orbit around the Sun, with the Sun occupying one of the two foci of the ellipse*."

An ellipse is a circle with the opposite ends of the diameter pulled outward, making it appear as an oval-shaped figure. The long axis of the ellipse is known as the major axis and perpendicular to the major axis going through the center of the ellipse is the minor axis. There are two points on the major axis called the foci, or focus for singular. The sun occupies one focus and the other is empty.

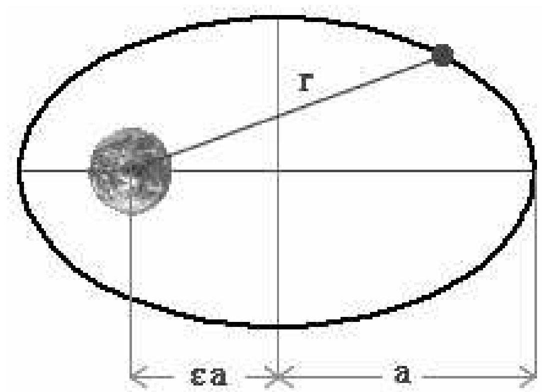

Figure 3: Kepler's First Law

(webphysics.iupui.edu/gpnew/gp2th4.htm)

In Figure 3 the Earth occupies one focus along the major axis. The length "a" represents the semi-major axis, which is half of the major axis. The dot on the ellipse is the satellite or moon orbiting the Earth in an elliptical orbit. The "εa" represents eccentricity. Eccentricity is the ratio between the distances of a focus from the center of the ellipse to the length of the semi-major axis. This determines the shape of an elliptical orbit. For example an eccentricity equal to zero would describe a circular orbit and an eccentricity equal to 1 would describe a highly elliptical orbit. The perigee is the point where the moon or satellite is at its closest point to the earth. The apogee is the point where the moon or satellite is at its farthest point from the earth. Both of these points are located along the major axis.

Kepler's Second Law, which is also known as the Law of Areas states, “*The imaginary line connecting any planet to the Sun sweeps over equal areas of the ellipse in equal intervals of time*.”

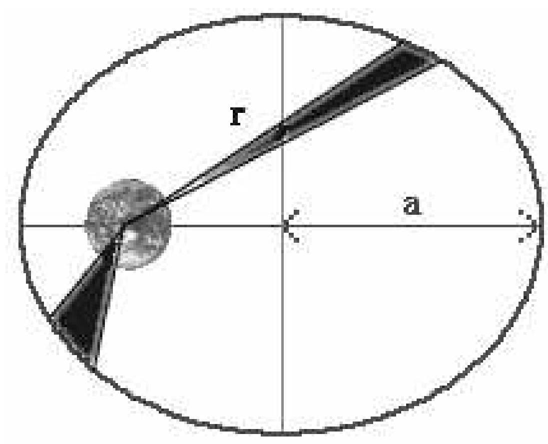

Figure 4: Kepler’s Second Law

(webphysics.iupui.edu/gpnew/gp2th4.htm)

Kepler's statement means, orbital speeds of a planet around the sun vary. The planets move fastest when closest to the sun and slowest when farthest away. This speed change is taken into account in Kepler’s Second Law. The distance covered in proportion to time is shorter when farther away and the distance covered in proportion to time is longer at a closer distance. “The satellite moves around an orbit in such a way that the radius vector sweeps equal areas in equal times” (Gravity and Satellite Orbits, 2004).

Kepler’s Third Law, which is also known as the Harmonic Law states,

“*The square of any planet's orbital period (its sidereal period) is proportional to the cube of its mean distance (the length of the semimajor axis) from the Sun.*”

The orbital period is the time for the planet to move 360° around the sun.

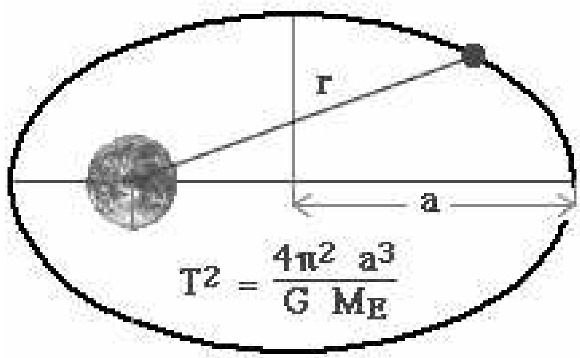

Figure 5: Kepler's Third Law

(webphysics.iupui.edu/gpnew/gp2th4.htm)

This law relates the period of satellite motion to the size of the orbit. T is the orbital period. G is the constant of universal gravitation and M is the mass of the Earth. Or if the sun was at the focus of the ellipse M would be the mass of the sun. To simplify this equation we can write it as,

$$T^2 = Ka^3$$

The constant K replaces all the other variables in the equation that are represented in the figure. This new equation is a literal interpretation of Kepler's third law as the square of the orbital period is directly proportional to the cube of the semi-major axis.

2.4.2 Classical Orbital Elements (Keplerian Elements)

There are six classical orbital elements that are needed to describe an orbit in space and time. This set of elements describes an orbital ellipse around the earth and then orients it three dimensionally and places a satellite along the ellipse in time.

The elements are listed in Table 1:

Name	**Symbol**	**Describes**
1.Semimajor axis	a	a constant defining the size of the orbit (meters)
2. Eccentricity	e	a constant defining the shape of the orbit (0=circular, Less than 1=elliptical)
3. Inclination	i	the angle between the equator and the orbit plane
4. Longitude of ascending node or Right Ascension of the Ascending Node	Ω	the angle between vernal equinox and the point where the orbit crosses the equatorial plane (going north)
5. Argument of perigee	ω	the angle between the ascending node and the orbit's point of closest approach to the earth (perigee)
6. True anomaly	v	the angle between perigee and the vehicle (in the orbit plane)

Table 1: The Classical Orbital Elements

(http://liftoff.msfc.nasa.gov/academy/rocket_sci/orbmech/state/class.html; http://www.stk.com/resources/help/help/stk43/primer/coordsys-18.htm)

Two elements not mentioned in this Classical set that are of some importance are the orbital period T, and the time of periastron passage τ. The periastron is the point of closest approach when orbiting around the Earth, which can also be referred to as the perigee.

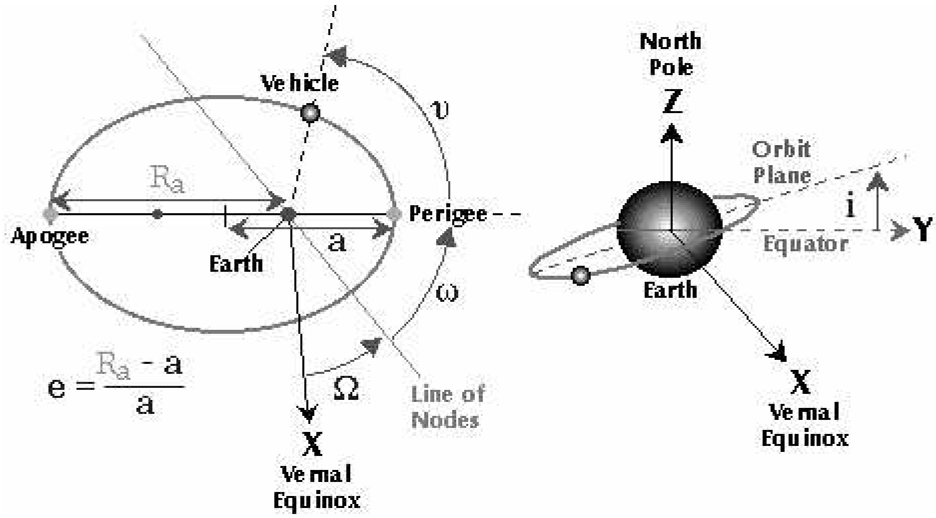

Figure 6: A figure of the classical orbital elements

(liftoff.msfc.nasa.gov/academy/rocket_sci/orbmech/state/class.html)

Figure 6 is a graphical representation of all the classical orbital elements. R_a is the distance from the center of the Earth to the apogee point. The Line of Nodes is the point where the satellites cross the equator.

2.4.3 Earth Orbits

There are an infinite number of possible orbits for an earth satellite. There are three general orbits that have become commonly used in today's space applications: geostationary orbits, sun-synchronous orbits, and polar orbits.

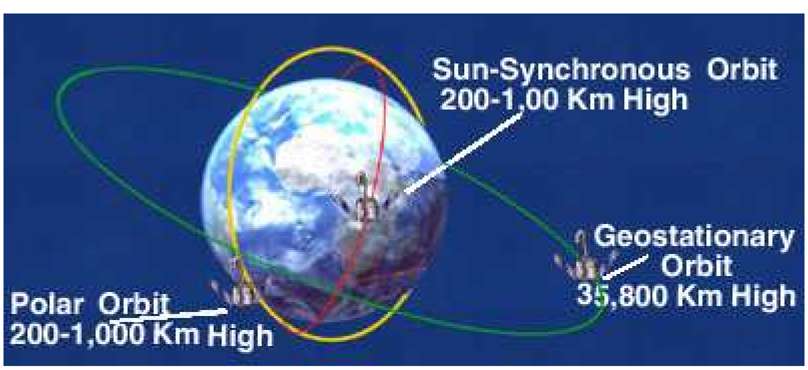

Figure 7: Common Earth Orbits

(www.geo-orbit.org)

Figure 7 above shows a polar orbit, a geostationary orbit, and a sun-synchronous orbit. Sun-synchronous and polar orbits are categorized to be low earth orbits (LEO).

A satellite in the low earth orbit is generally 250 – 300 km above the earth's surface and can go as high as 1000 km on average. This orbit is very close to the earth and is commonly used for global mobile telephone systems and weather satellites. A satellite in this orbit can usually travel around the earth in about 90 minutes.

The sun-synchronous orbit is a special type of low-earth polar orbit, which is mostly used for weather forecast machines and earth observation machines. Sun-synchronous orbits rotate as to maintain constant view of the sun. Satellites that depend highly on solar power need to be in constant view of the sun to maintain operational power.

Satellites in a geostationary orbit have an orbital height of 35,786 km. A satellite in this orbit will appear fixed from the earth's surface because the satellite is traveling at such a velocity that matches the Earth's rotational speed. This orbit is especially good for communications and continent wide weather monitoring. A satellite in this orbit can complete one orbital period in exactly the same time it takes the Earth to rotate 360°, or almost exactly 24 hours.

A satellite in the polar orbit is also said to be in a low earth orbit, the only difference being that satellites in polar orbits travel around the earth from North Pole to South Pole instead of an east to west direction. The ST5 satellites will fly in a polar orbit with an apogee of 4500 km and a perigee of 300 km.

2.4.4 Orbit Propagator

An orbit propagator is a computer simulation that computes the position and velocity of an earth orbiting satellite. The orbit propagator begins with an initial position and velocity vector. Using orbital mechanics the propagator then calculates new vectors with the passing of time. The orbit propagator used for SimulinkST5 is a two body propagator consisting of the earth and the sun. This propagator does not take into account the non-spherical shape of the earth or atmospheric effects or gravitational effects of the moon. For SimulinkST5 we incorporated an existing orbit propagator, and modified it to incorporate Improved Inter Range Vectors for more accuracy, into our newly designed communication model.

Improved Inter Range Vector (IIRV)

An IIRV is a standard message developed by GSFC, which contains six lines of ASCII code describing multiple satellite parameters. The parameters we will use in this project are a position vector, a velocity vector, and the current time. These parameters will be integrated into our orbit propagator to produce a highly accurate simulated orbit of the ST5 satellites.

IIRV satellite parameters are obtained by a NASA based satellite tracking system called Tracking and Data Relay Satellite System, or TDRSS. White Sands Complex (WSC) located in Las Cruces, New Mexico, extracts tracking data from the satellite's downlinked telemetry. This telemetry is then sent in TDRSS format to GSFC's Flight Dynamics Facility (FDF). FDF then temporarily stores this information as ephemeris files, or tables giving the coordinates of a celestial body at a number of specific times during a given period. Ephemeris files can then be transformed into acquisition data in an IIRV format. IIRV's are then sent to ground stations several times a day, centered on every four hours, to provide accurate position and velocity vectors of orbiting satellites (Tracking, 2005).

2.5 Spacecraft Attitude

In order to accurately model the communications link, spacecraft attitude must be known. Attitude is the physical orientation of a satellite with respect to a defined spacecraft axis system. The ST5 spacecraft axis system utilizes the Earth Centered Inertial coordinate system. Attitude error, which is the spacecraft misalignment from the target position, will be provided as a right ascension and declination with respect to the ECI coordinate system shown in Figure 8. Right Ascension is the rotation angle about the x-y plane with its reference point being along the y-axis. Declination is the angle above the x-y plane with its reference point being zero degrees parallel to the x-y plane. The attitude data is gathered from a combination of the Miniature Spinning Sun Sensor and the Magnetometer aboard the ST5 spacecraft.

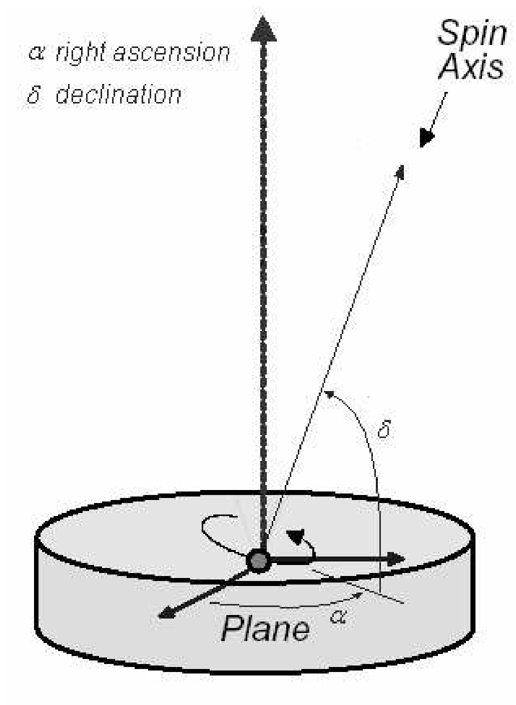

Figure 8: Attitude

The Attitude will change slightly each day in orbit due to atmospheric conditions and gravitational forces. To correct the attitude, ST5 is equipped with a cold gas micro-thruster, which will correctly orient the spacecraft when needed and will maintain 3σ (98.9%) precision of the spin axis.

2.6 Telemetry

Telemetry is the relaying of information from scientific instruments aboard a satellite to a ground station. The information relayed is mostly health and safety information, or spacecraft housekeeping information. This may include battery voltages, solar panel currents and internal temperatures at certain points on the satellite. This information is used to maintain the health and safety of the spacecraft.

When the spacecraft compiles the telemetry data, it is sent in a binary format. This binary code is then converted to something that we can understand, such as degrees or volts, by the receiving equipment at the ground station.

The use of international standards for formatting spacecraft data such as telemetry is growing. The Consultative Committee for Space Data Systems, or CCSDS, has produced a set of standards that are used in most spacecraft missions today. Space Technology 5 is one of the 250 missions that are using these CCSDS standards. The benefits of these standards are reduced cost, reduced risk and development time, and enhanced interoperability and cross-support. CCSDS is standardizing spacecraft platforms and space-qualified hardware components to ground support hardware and software (CCSDS 2004).

Telemetry data has two parts, a header field and an application data field. The header contains the routing information such as the where and when part, and the application data field contains the context of the telemetry information.

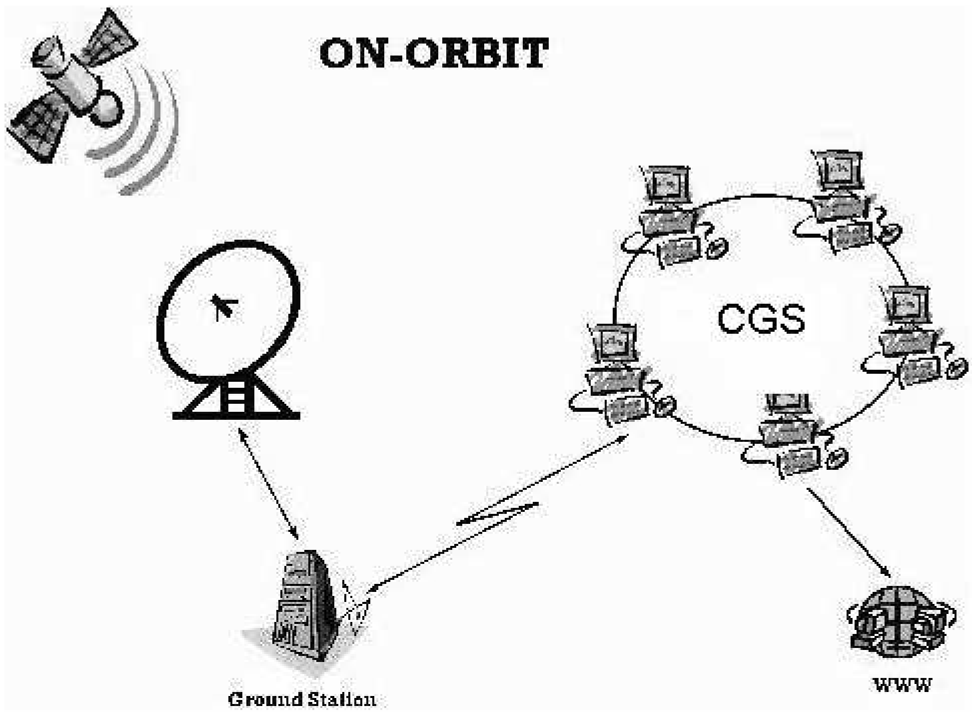

Figure 9: Telemetry

(cslp.gsfc.nasa.gov/assignments/assign4.html)

Figure 9 shows Telemetry in a symbolic manner. The satellite in orbit and in range downlinks the telemetry data to a ground receiver, which is connected to a ground station. The ground station then relays this information to the mission operations control center, or MOC, which is represented above as five computers in a circle. The MOC houses the Combined Ground System, or CGS, hardware and software which takes the binary code and translates it into readable information. This translated information is then relayed through the MOC network document for mission analysts to observe.

2.6.1 Institute for Scientific Research

The Institute for Scientific Research (ISR) is a company that specializes in world class projects in scientific research and advanced development and provides leading edge technology solutions to their clients. ISR researchers work in association with NASA Goddard Space Flight Center and many other federal agencies and private companies. ISR is an independent non-profit corporation located in West Virginia (ISR, 2003).

ISR developed a satellite interface utilizing the GMSEC bus that provides access to the ST5 satellite telemetry. GMSEC stands for Goddard Space Flight Center Mission Services Evolution Center. The GMSEC bus creates a standard way of transmitting data from one application to another. Multiple applications will be accessing the bus at any time and must be capable of communicating with each other.

2.7 ST5

For the ST5 model, specific components need to be identified to ensure proper function of the satellite. The satellite can be broken down into eight subsystems.

- The *structural/mechanical* subsystem describes the satellite itself. The satellite structure and all moving parts, instruments, and other systems are components of the structural/mechanical subsystem.
- The *thermal subsystem* regulates the thermal characteristics of the satellites and its multiple components. The ST5 thermal subsystem consists of blankets, coatings and the Variable Emittance Coating (VEC) technology, which will be flight validated.
- The *power/electrical subsystem* supplies the other subsystems with the necessary power. The power is generated using solar panels that surround the ST5 satellite, as well as a Li-Ion battery that can be recharged via the solar panels.
- The *radio frequency (RF) communications system* uses an X-band transponder, low noise amplifier (LNA), high power amplifier (HPA), and two antennas mounted on the top and bottom of the satellite. This system allows the satellite to communicate with the ground stations.
- The *guidance, navigation, and control subsystem* uses a Miniature Spinning Sun Sensor (MSSS) and a magnetometer to gather data which can be used to determine the attitude of the satellite and ensure spin-stabilized control (Frisbee like rotation).

- The *propulsion subsystem* utilizes cold gas micro-thrusters. These thrusters are fired in pulses to conserve energy yet keep the satellites in orbit as well as a proper distance from earth and each other.
- The *Command and Data Handling (C&DH) subsystem* monitors and records the data gathered by the subsystem components, and magnetometer.

The SimulinkST5 behavioral model consists of the electrical power subsystem, the data recorder, and the communication subsystem. For our goal we needed to familiarize ourselves with the communication subsystem as well as the Simulink model developed by the 2002 WPI MQP team.

2.7.1 ST5 Communication Subsystem

The ST5 communications subsystem utilizes the Deep Space Network (DSN) and Ground Network (GN) ground stations. This subsystem consists of an X-band-transponder, a diplexer, two antennas, a low noise amplifier, and a high-power amplifier for transmission purposes. Figure 10 shows a block diagram of the ST5 Communication subsystem.

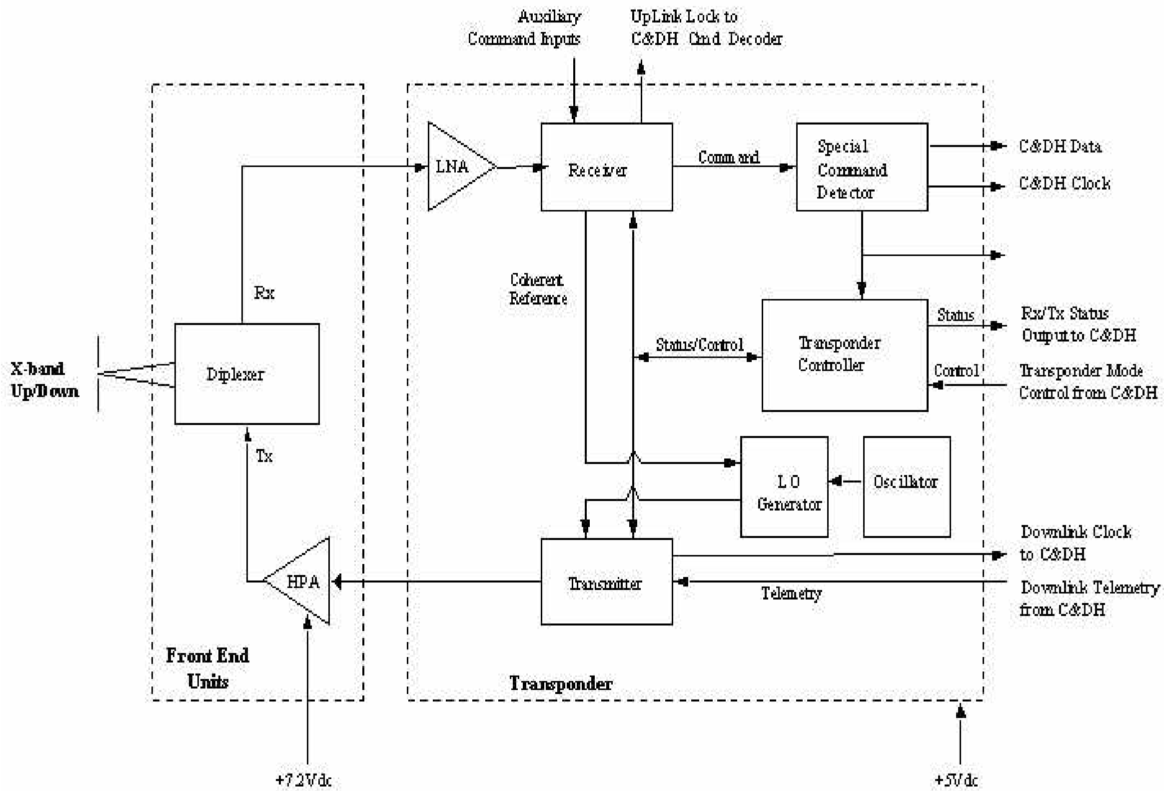

Figure 10: ST5 Communication subsystem

(ST5 MQP 2002)

The X-band transponder is a NMP technology that will be tested on the ST5 mission. The transponder has the following characteristics:

- Uplink data rate of 1Kbps
- Downlink data rate of 100Kbps
- Downlink data rate of 200Kbps
- Emergency real-time downlink rates of 10Kbps or 1 Kbps
- Transmission frequency approximately 8.5 GHz (transmission once per orbit)

Using the DSN and GN networks for ground station communications requires some extra planning because each is a shared system. The time and duration of transmission must be scheduled ahead of time to coincide with the projected orbit of ST5 satellites and the availability (schedule) of DSN and GN stations. Data stored on the data recorder as well as real-time housekeeping data is transmitted over a time period of approximately 6 minutes per satellite, per orbit.

2.7.2 ST5 Antenna

Recent changes to the ST5 orbit from a geosynchronous orbit to a polar orbit have affected the communication capabilities of the current ST5 antenna. The previous ST5 antenna known as the QHA (Quadrafilar Helix Antenna) shown in Figure 11, was designed for long distance communications for the geosynchronous orbit which had an apogee and perigee of 36,000 km and 300 km respectively. The new low earth orbit is now one eighth the size of the old orbit with an apogee and perigee of 4500 km and 300 km. The QHA will meet the requirements of the new orbit but will not be as effective.

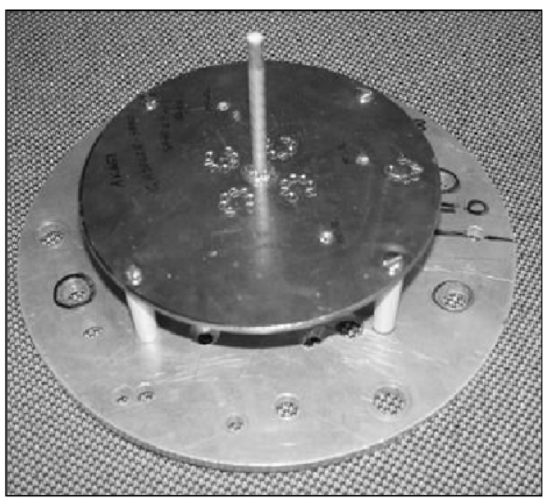

Figure 11: Quadrafilar Helix Antenna

(ST5 ΔCDR)

New Technology Evolved Antennas, shown in Figure 12, developed by NASA Ames Research Center may be available for ST5 which would increase the communication capabilities in LEO’s. This new antenna is called the Evolved Antenna or EA.

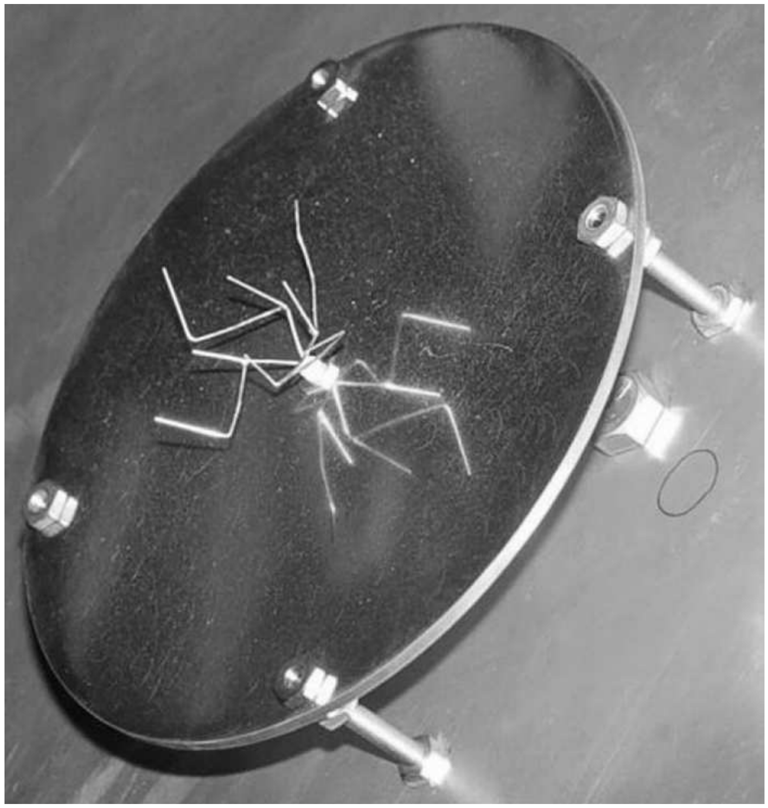

Figure 12: Evolved Antenna
(ST5 ΔCDR)

The wire form is designed using tree-structured computer algorithms. Each prong is mathematically placed to provide maximum capabilities. This tree structure creates a potential for high gain over a wider range of radiative pattern angles. The EA produces a greater gain and can be manufactured at a lower cost than the QHA due to its limited number of parts. The EA is still in test phases and the technical specifications were not available. Predicted EA parameters are currently used by the Communications model.

2.8 Link Margin

To evaluate the communication systems performance we had to account for the Link Margin (LM). The link margin is the difference between the required signal to noise ratio and the actual signal to noise ratio.

$$LM = \frac{E_b}{N_0} - \frac{E_b}{N_0}(reqd) \tag{2-1}$$

If the link margin is not sufficient enough, it will result in an undesired bit error rate at the receiver.

Link margin calculations begin with signal-to-noise power ratio (SNR). The SNR is the ratio of the signal power over the noise power as shown in Equation (2-2) below.

$$SNR = \frac{signal(power)}{Noise(power)} \tag{2-2}$$

A decrease in the signal power results in the SNR decreasing (loss). An increase in noise power, as well as increases of interfering signal power (noise), will have the same effect. Losses can also occur from the signal being absorbed, reflected, or scattered, before the signal reaches the receiver.

There are four primary sources of noise when dealing with satellite transmission.

1. Thermal noise generated within the link
2. Atmospheric noise
3. System nonlinearities
4. Interference signals from other users

SNR is the quantity of greatest interest for this analysis because our system evaluation is based on our ability to detect the signal, with an acceptable error probability (in the presence of noise). The calculated signal to noise ratio takes into account the antenna power and gain, the data rate, the frequency, and the various losses that would affect the signal (free space loss, atmospheric losses, and passive losses). Equation (2-3) shows the equation for signal to noise ratio. Link Margin (2-4) is simply Equation (2-3) minus a required signal to noise ratio.

$$\frac{E_b}{N_0} = EIRP + L_s + L_a + G_r - T_s + \kappa - R \tag{2-3}$$

$$LM = EIRP + G_r - R - \kappa - T - L_s - L_o - \frac{E_b}{N_0}(reqd) \tag{2-4}$$

Equation (2-4) has eight variables that need to be considered. L_s is the space loss, L_o is all other losses including the ones previously mentioned above, G_r is the receiving antenna gain (which includes diameter of antenna and half-length beamwidth), κ is Boltzmann's Constant which is a constant equal to 228.6, R is the data rate, T is the system temperature in Kelvin, and $\frac{E_b}{N_0}(reqd)$ is the required signal to noise ratio which is a constant requirement defined by ST5 parameters. The signal to noise ratio requirements changes for uplink and downlink. *EIRP* is the Effective Isotropic Radiated Power, which takes into account the transmitting power and gain, shown in Equation (2-5).

$$EIRP = P_t + G_t \tag{2-5}$$

G_t is the transmitting antenna gain. P_t is the transmitting antenna power. The required signal to noise ratio for any downlink to either ground network is 4.45 dB. The required signal to noise ratio for any uplink from either ground network is 9.6 dB. These variables can all be found in the Radio Frequency Interface Control Document (RFICD) provided to us by Victor Sank (ST5 RF/Communications, vsank@pop500.gsfc.nasa.gov).

2.9 MATLAB/Simulink Software

MATLAB is a high performance language for technical computing allowing for integrating computation, visualization, and programming in an easy-to-use environment. MATLAB problems and solutions are expressed in familiar mathematical notation, whose basic data element is an array that does not require dimensioning. This software solves many technical computing problems in a fraction of time than it would take to write a program in a scalar non-interactive language such as C or FORTRAN (Learning MATLAB, 2002).

Simulink is an interactive tool for modeling, simulating, and analyzing dynamic, multi-domain systems. It is part of the MATLAB software package distributed by The MathWorks, Inc. Simulink integrates seamlessly with MATLAB, providing immediate access to an extensive range of analysis and design tools. Simulink allows the user to build a block diagram, simulate the system's behavior, evaluate its performance, and refine the design. The software also allows for the modeling of linear and non-linear systems in continuous time, sampled time (single-rate or multi-rate), or both. These benefits make Simulink the tool of choice for control system design, digital signal processing (DSP) design, communications system design, and other simulation applications (Learning MATLAB, 2002).

Simulink models are hierarchical so that the user can use either a top-down or bottom-up approach, allowing for comprehensive organization of systems, subsystems, and components at different levels. This also provides the user with insight into how a model is organized and how the different parts interact. Once a model is designed it is not difficult to simulate it by using the simulation menus that Simulink provides, or by using commands in MATLAB's command prompt. The simulation menu is useful for interactive simulations, while the command line is applicable for batches of simulations (Dabney & Harman, 2001).

2.10 Summary

ST5 is composed of three small-satellites to be launched into a low earth orbit and flown for three months in a "string of pearls" formation. The mission of ST5 is to verify new technologies in space flight. Our project deals with link margin analysis, basic satellite orbits, telemetry, antenna's, attitude, and MATLAB/Simulink software. The critical system for verifying the new technology is the communications subsystem. In order to ensure successful completion of the mission, a behavioral model needs to be created focusing on the communication model using MATLAB/Simulink simulation software.

3 Project Statement

3.1 Introduction

The purpose of this project was to accurately determine ST5 satellite communication capabilities using Simulink. In order to upgrade the communications model to provide accuracy for long-term mission planning, new components were added to the orbit propagator model and communications model. Below, we provide a detailed description of our project goals, objectives and tasks.

3.2 Project Goals

The primary goal of this project is to accurately model the ST5 communications link for long-term mission planning. Determining communication capabilities is dependant on the location of the satellite at a specific time of interest; therefore the communications model will need accurate position and velocity vectors describing the satellite orbit.

The orbit propagator outputs a position and velocity vector that is then interfaced with the communications model. The previous communications model relied on a simple two body propagator that provided accurate vectors for 3-4 hours. Since SimulinkST5 will be running simulations for a period of 3 weeks, the propagator was upgraded for more accuracy. The new orbit propagator utilizes Improved Inter Range Vectors (IIRV's) which provide discrete points describing the satellite location. The IIRV's are incorporated to the two body propagator to provide realignment of the satellite orbit, correcting any propagation errors. Upgrading the orbit propagator model consisted of the following primary objectives and tasks.

1) Orbit Propagator Model
    a) Incorporate IIRV files

        i) Understand Orbit Propagator Algorithm
        ii) Interpret IIRV files
        iii) Understand Coordinate Transformations
    b) Interface Outputs to Communications Model

Using the orbit propagator, the ST5 communications model was updated to determine communications capabilities. The updated model accounts for the ST5 antenna radiative pattern that provides an antenna gain to be used for link margin calculations. The antenna radiative pattern was also modeled to help determine the occurrence of the Zone of Interference. Determining ZOI required the line of sight calculation and ZOI to be upgraded as a function of spacecraft orbital geometry and attitude.

The communication model will asses the communications link for each ground station that ST5 uses. A new DN ground station, McMurdo, will be used to help support the low earth orbit of ST5. The location of McMurdo (Antarctica) provides the most frequent communication possibilities for polar orbits. For each ground station link margin, Doppler shift, and look angles are all calculated to determine communication capability. The communications model can be broken down into several objectives and tasks.

1) Communications model
    a) Incorporate GN ground station, McMurdo
        i) Research GN antenna parameters
    b) Model Antenna Radiative Pattern
        i) Gather EA pattern data
    c) Calculate Line of Sight
        i) Understand Coordinate Transformations
    d) Determine Zone of Interference
        i) Understand Orbital Geometry

The communications model provides multiple output graphs of link margin, Doppler Shift, ZOI, and line of sight. The model also provides output files of date and times corresponding to line of sight and ZOI.

3.3 Summary

The primary goal of this project was to update the existing communications model to support long-term mission planning to predict communication capabilities. To accomplish this, the orbit propagator was upgraded to provide accurate positions and velocity vectors that are then interface to the communications model. The communications model was upgraded to model the antenna radiative pattern as a function of attitude which is used to determine line of sight, and ZOI. The final output of the model provides the necessary data to predict communications link capabilities.

4 Model Structure

4.1 Introduction

The communications model is designed to accurately determine the ST5 communication link quality. Since the communications model is a function of the orbit propagator outputs, the communications model can only be as accurate as the orbit propagator. Therefore the orbit propagator is designed to provide better accuracy for long-term mission planning. In this section we outline the structures of both the communications model and orbit propagator model.

4.2 Communications Model

The purpose of the ST5 Communications model, shown in Figure 13, is to accurately determine the link quality between ground stations and the satellite. Link quality is dependent on ground station position, satellite position, and antenna parameters. The model is used to calculate look angles, link margin, Zone of Interference and Doppler Shift for each ground station. There are four inputs necessary for the Communications model, date/time information, latitude/longitude/altitude of each ground station, satellite ECI position and velocity vectors, and the spacecraft attitude.

4.2.1 Inputs/Outputs

The communication model is dependent on ECI position and velocity vectors generated from the orbit propagator. The ECI vectors are structured with time and stored in an array that the model accesses through the Satellite ECI subsystem. The model requires the ground station latitude/longitude/altitude (LLA) to determine the ground stations position as a function of time. Ground Station LLA Coordinates subsystem takes user input from an initialization file, which must be executed before the simulations can run. The date and time information is also initialized before the simulation is run and then sent to the Date/Time subsystem. These three input subsystems are inputs to the S-Function, SFunGStations, to determine look angles for satellite visibility and link margin. There are two SNR input subsystems for GN and DSN stations, which input antenna parameters to calculate the link margin.

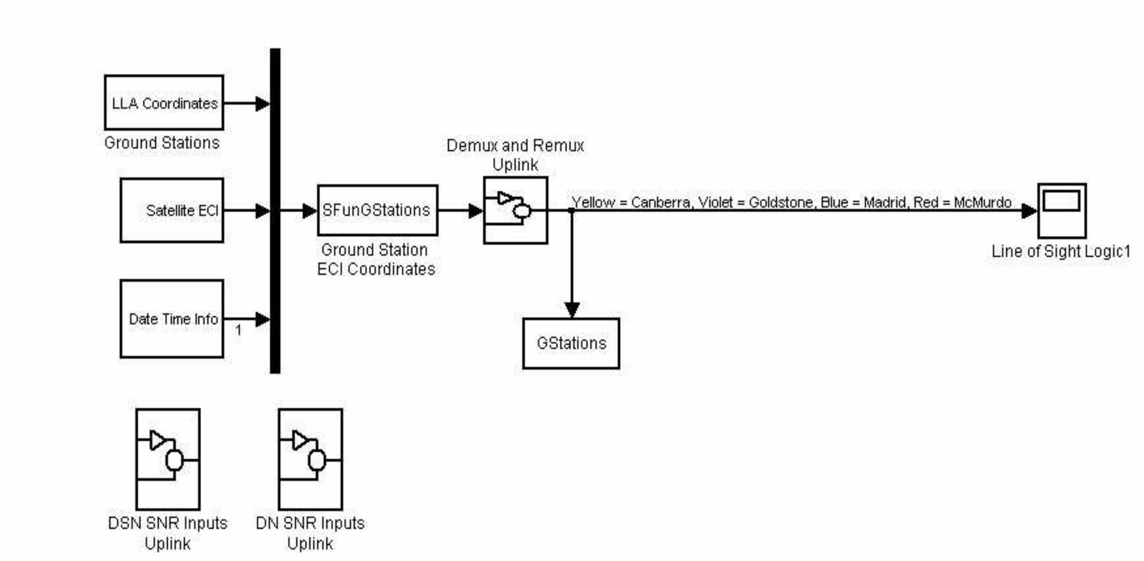

Figure 13: Top Level Communications Simulink Model

With these inputs the model outputs a line of sight logic graph describing visibility for each ground station. The graph shows logic 1 (visible) or logic 0 (not visible) as a function of time. The model outputs a file listing dates and times of black outs for visibility that can be compared with the logic graph. The model will also provide a file listing dates and times of ZOI that occur within the line of sight. Link margin and Doppler shift calculations are also graphed versus time.

4.2.2 Ground Station Visibility

The ability of the ground station to see the satellite depends on the location of the satellite in orbit and the maneuverability of the ground station antenna. Coordinates called Look

Angles give angular measurements of where the ground station antenna needs to be pointed at a given time in order to communicate with the satellite.

For the Communications model, the look angles are calculated within the S-Function, SFunGstations. The calculation can be broken up into four parts: the Julian Date calculation, the Greenwich Sidereal Time calculation, the Latitude/Longitude/Altitude (LLA) to Earth Centered Inertial Coordinates (ECI) conversion, and finally the ECI to Look Angles conversion.

Julian Date Calculation

A Julian date (JD) is the interval of time measured in days from the epoch January 1, 4713 B.C., 12:00 pm. A Julian date is a single number that corresponds to the Gregorian calendar. The time of day for a particular JD can be represented as a decimal fraction appended to the JD number. A Julian day is measured from noon to noon so a decimal fraction of 0 would indicate noon (Vallado, 2001).

The JD for noon, November 24, -4713 is 0. More recent JDs are used as references for counting JDs, and it is common to reference back to the first day of January of a given decade to simplify JD calculations. The equation used for the Julian Date calculation is taken from Astronomical Algorithms by Jean Meeus shown in Equation (4-1).

$$J_{date} = fix(365.25*(year+4716)) + fix(30.6*(month+1)) + day + 2 - fix(year/100) + fix(year/400) - 1524 \tag{4-1}$$

This Equation inputs a year, month and day to calculate a Julian date (Sklar, 1998).

Greenwich Sidereal Time (GST)

With a Julian date calculated, the next step is to calculate GST. Greenwich Sidereal Time is the angle between the Prime Meridian and the Vernal Equinox. The GST can then be used to calculate the sidereal time which is a measure of time defined by the motion of the vernal equinox in hour angles. For a given place (latitude) and instant in time the sidereal time gives the hour angle of that equinox which is necessary to calculate the ECI coordinates for a given point on the earth's surface (Meeus, 98).

The GST calculator used in the model, created by the 2002 ST5 MQP, takes a Julian Date and time of day (hour, minute, second) input and converts it into a single number. The conversion into a single number is done by first subtracting 0.5 from the JD (so that the interval measured is from midnight to midnight, not noon to noon), and then adding as fractions the exact time of day. The model is continuously changing the time of day input to the model which in turns causes the current Julian date, GST, and ECI point to continuously change. The Equation used to obtain the single number representation of the JD and time is:

$$Current = JD - 0.5 + hr * 0.0417 + \min * 6.94e^{-4} + \sec * 1.156e^{-5} \tag{4-2}$$

With the current date, subtraction is used to determine the amount of time passed since a certain reference date (JD 2440952.5 or midnight, January 1, 1971). The GST in radians, is known for this reference date and can be calculated for any known date and time using the Equation:

$$\Theta_g = \Theta_g 0 + 1.0027379093 * 2 * \Theta * D \tag{4-3}$$

Where Θ_g is the GST in radians; $\Theta_g 0$ is the GST in radians of the reference date; 1.0027379093 is the number of rotations the earth completes in one solar day; and D is the difference between the reference date and the current date (Bate, 1971).

Ground Station to ECI Coordinate Conversion

The conversion from Ground Station position to ECI coordinates (Figure 14 and Figure 15) uses the LLA of the ground station and incorporates the rotation of the earth. Since every point on the earth is rotating and ECI is a fixed axis system, the ECI position of the ground station is a function of time. For this reason the conversion from LLA to ECI coordinates is a necessary intermediate calculation to determine the visibility of each ground station.

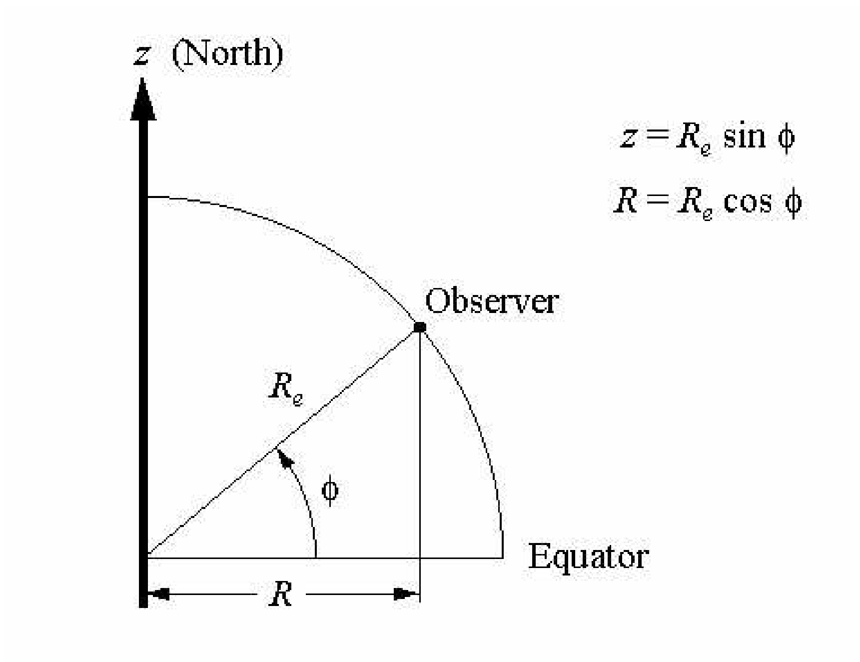

Figure 14: Latitude to ECI Conversion

(celestrak.com/columns/v02n01)

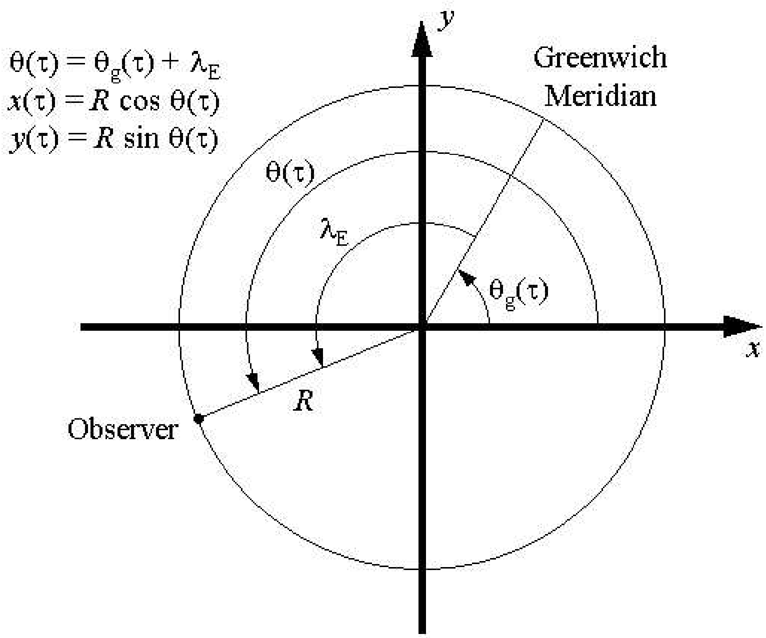

Figure 15: Longitude to ECI conversion

(celestrak.com/columns/v02n01)

The calculation method used in this model is the same calculation that the 2002 ST5 MQP used that was based on the FalconSat project. The first step is the computation of the local sidereal time (LST), the angle between the Vernal Equinox and the local longitude. The local sidereal time is calculated by adding the Greenwich Sidereal Time to the local longitude.

$$\text{LST} = \text{GST} + \text{longitude} \tag{4-4}$$

The second step is the computation of two geodetic constants c and d, which account for the flattening of the Earth, b.

$$\begin{aligned} b &= \sqrt{\left(1 - \left(flat(2 - flat) * \sin(latitude)^2\right)\right)} \\ c &= R_{eq} / b + 0.001 * altitude \\ d &= R_{eq} * (1 - flat)^2 / b + 0.001 * altitude \end{aligned} \tag{4-5}$$

Where *flat* is the flattening of the Earth and R_{eq} is the radius of the Earth at the equator in meters.

The final step is the use of the geodetic constants and the latitude and longitude to calculate the ECI coordinates at a given time.

The ECI coordinates are calculated as:

$$\begin{aligned} X &= c * \cos(latitude) * \cos(LST) \\ Y &= c * \cos(latitude) * \sin(LST) \\ Z &= d * \sin(latitude) \end{aligned} \tag{4-6}$$

ECI to Look Angles Conversion

Now that the ground station LLA has been converted to ECI coordinates we have two ECI positions, Satellite ECI (from orbit propagator) and Ground Station ECI. We can now calculate the angles pointing between the two, known as look angles, which describe the vector needed for the ground station to point to the satellite. Look angles consist of azimuth, elevation, and range. Azimuth is the horizontal direction of the antenna, expressed as the angular distance between the antenna and the satellite. The azimuth is measured clockwise from north (0°), through the east (90°), south (180°), and west (270°). Elevation is the angle from the horizon of the earth to the satellite, describing the vertical tilt of the ground station antenna.

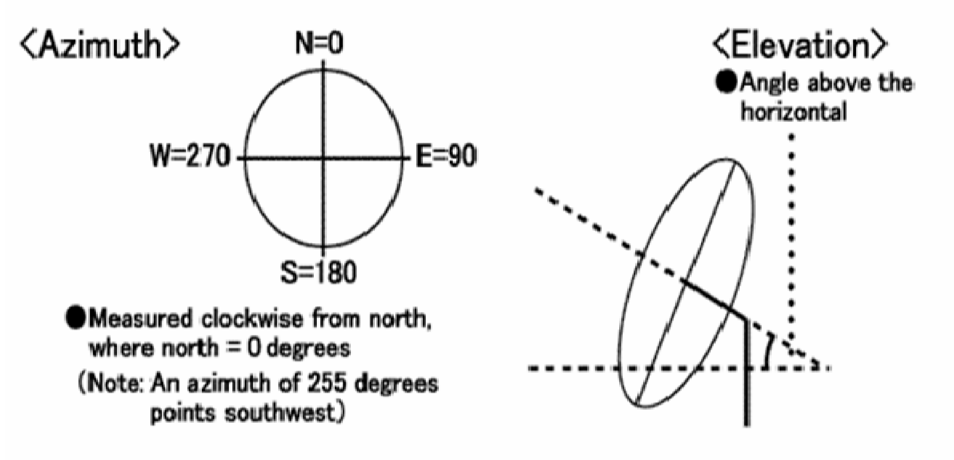

Figure 16: Azimuth and Elevation
http://www.nhk-jn.co.jp/wp/img/antenna_e.gif

The first step in determining the look angles is calculating the range vector, the distance between the ground station and satellite. The range vector is calculated by taking the difference between the satellite's ECI coordinates and the ground station's ECI coordinates.

$$[r_x, r_y, r_z] = [x_s - x_g, y_s - y_g, z_s - z_g] \tag{4-7}$$

Where $[r_x, r_y, r_z]$ is the range vector, $[x_s, y_s, z_s]$ are the satellite's ECI coordinates, and $[x_g, y_g, z_g]$ are the ground station's ECI coordinates.

The range vector must be converted to the Topocentric-Horizon (SEZ) Coordinate system, shown in Figure 17, in order to determine visibility of the satellite. SEZ coordinates takes into account that the earth is not flat and that the azimuth and elevation angles are with respect to the curvature of the earth not the ECI coordinate system. The SEZ system rotates with the site and the local horizon forms the fundamental plane. The transformation to SEZ is done by rotating through the local sidereal time about the Z-axis (the Earth's rotation axis), then rotating through the ground station's latitude about the Y-axis (Vallado 2001). The Equations used for these calculations are:

$$r_s = \sin(\varphi)\cos(\theta) * r_x + \sin(\varphi)\sin(\theta) * r_y - \sin(\varphi) * r_z \tag{4-8}$$

$$r_E = -\sin(\theta) * r_x + \cos(\theta) * r_y \tag{4-9}$$

$$r_Z = \cos(\varphi)\cos(\theta) * r_x + \cos(\varphi)\sin(\theta) * r_y + \sin(\varphi) * r_z \tag{4-10}$$

Where r_S, r_E, and r_Z are the range vector expressed in SEZ coordinates, θ is the latitude and φ is the local sidereal time (Vallado 2001).

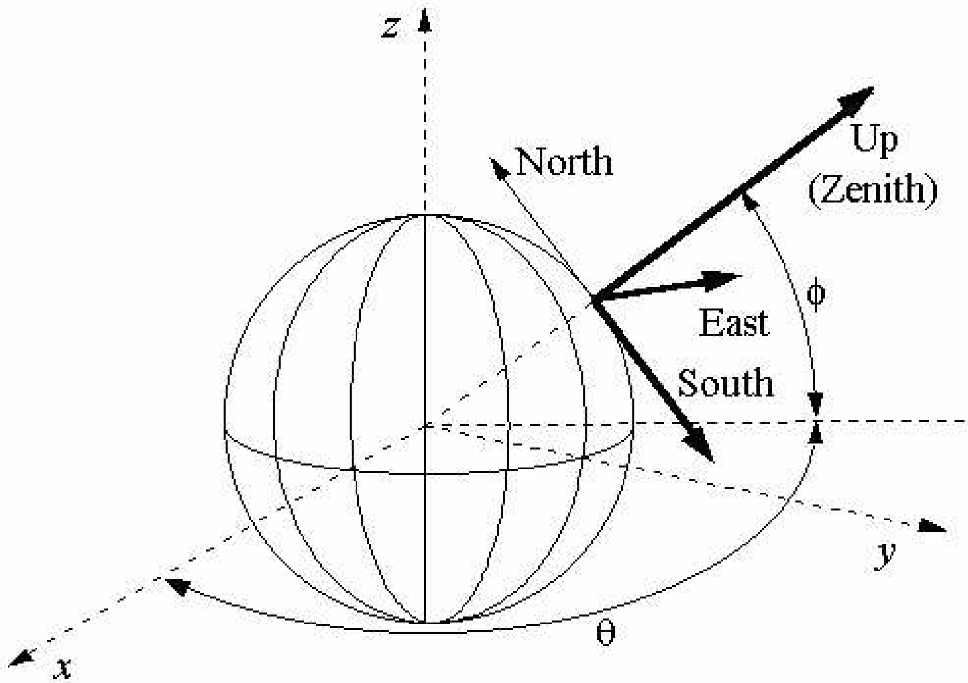

Figure 17: Topocentric-Horizon Coordinate System

(http://celestrak.com/columns/v02n02)

The look angles can now be calculated using the SEZ coordinates with the following equations.

$$range = \sqrt{(r_S^2 + r_E^2 + r_Z^2)} \tag{4-11}$$

$$Elevation = \sin^{-1}(r_Z / range) \tag{4-12}$$

$$Azimuth = \tan^{-1}(-r_E / r_S) \tag{4-13}$$

The specifications for antenna visibility are dependant on the specific antenna in use. The previous Communications model used a required 10° elevation and assumed no restriction on the azimuth angle. So any time the elevation is between 10° and 170°, the satellite is visible. Our communication model will use the same requirements.

4.2.3 Link Margin Calculation

The link margin is the difference between the desired minimum signal to noise ratio and the actual signal to noise ratio in dB. Link margins that are less than the desired minimum can result in unacceptable bit error rates in the received signal. Figure 18 shows the Simulink model that calculates link margin.

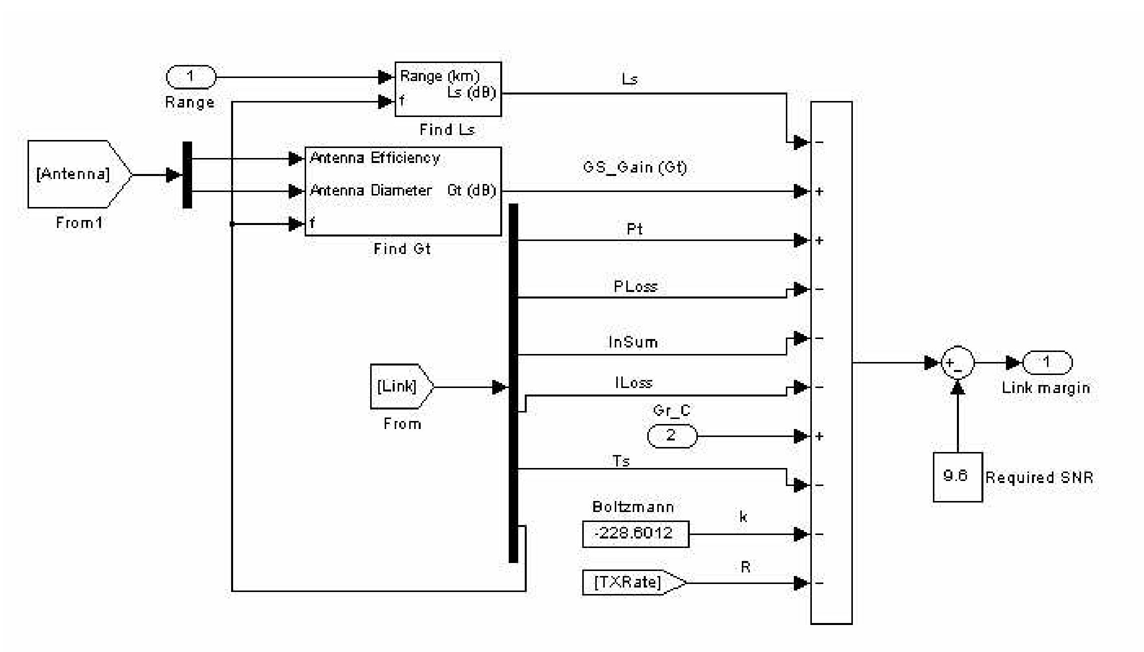

Figure 18: Link Margin Calculator

The link margin calculator takes as constant inputs all variables from Equation (2-3), except the EIRP, L_s, and G_r. The gain of the receiver (G_r) is dependant on the antenna radiative pattern and attitude of the spacecraft. G_r is determined in the Zone of Interference subsystem and inputted into the link margin calculator. Space loss, L_s, is dependent on the system frequency and the range calculated from look angles, shown in Equation (4-14).

$$L_s = -\left(20\log\left(\frac{c}{f4\pi}\right) - 20\log(1000\times Range)\right) \tag{4-14}$$

where L_s is equal to space loss, *Range* is the distance between the ground station and the satellite, c is equal to the speed of light and f is equal to the system frequency.

Figure 19 shows the subsystem that calculates space loss.

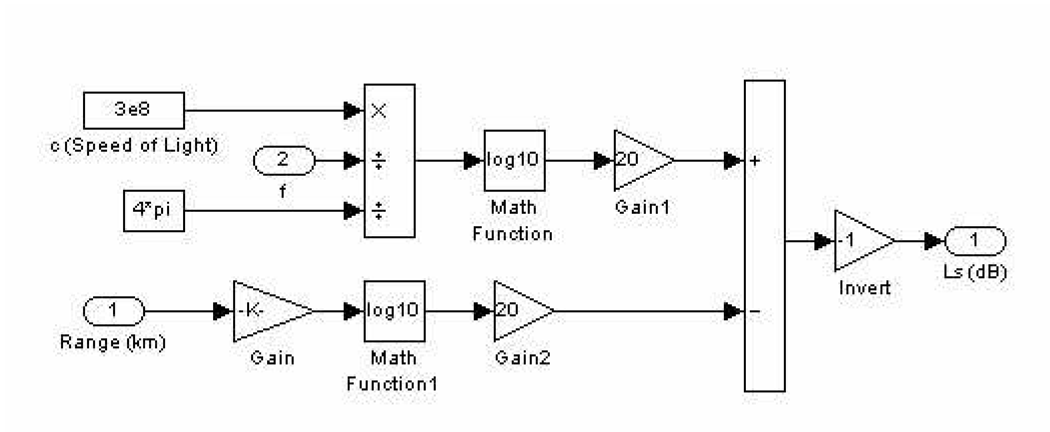

Figure 19: Space Loss Calculator

EIRP takes into account transmission gain and transmission power. Transmission gain is calculated by using Equation (4-15) and the calculation subsystem is shown in Figure 20 with inputs antenna diameter, antenna efficiency and frequency. Transmission power is a constant specific to the ground station.

$$Gt = 10\log\left(\frac{\left|\frac{Ad\times f\times\pi}{c}\right|^2 \times Ae}{100}\right) \tag{4-15}$$

where G_t is equal to the transmission gain, A_d is equal to the antenna diameter, A_e is equal to the antenna efficiency, f is equal to the system frequency and c is equal to the speed of light.

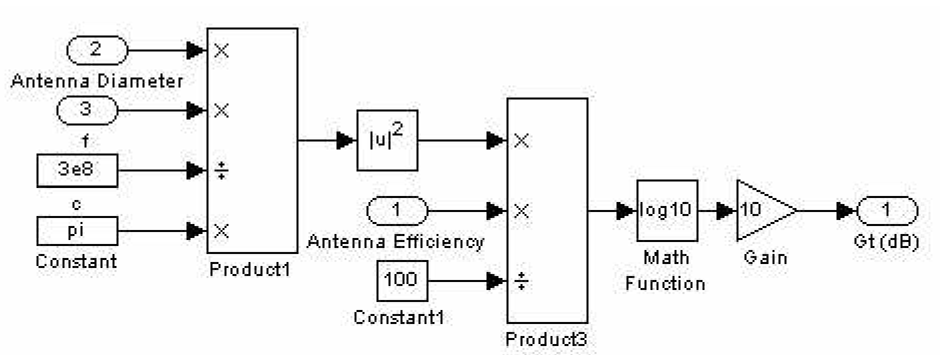

Figure 20: Ground Station Transmission Gain Calculator

Doppler Shift Calculation

Our model uses the same Doppler shift calculation as the 2002 ST5 MQP. The ST5 satellites transmitting frequency is 8.74 GHz and the frequency received by the ground station antennas will be 8.74 GHz plus/minus the Doppler frequency shift. The model calculates the Doppler shift as follows:

$$\Delta f = \frac{v_{rel} * f}{c} \tag{4-16}$$

where Δf is the change in frequency, v_{rel} is the velocity of the spacecraft relative to the velocity of the ground station, f is the transmitted frequency, and c is the speed of light.

In Figure 21, a range input from the ground station s-function (SFunGStations) calculates the continuously changing range as the satellite moves in orbit which is then used to calculate relative velocity.

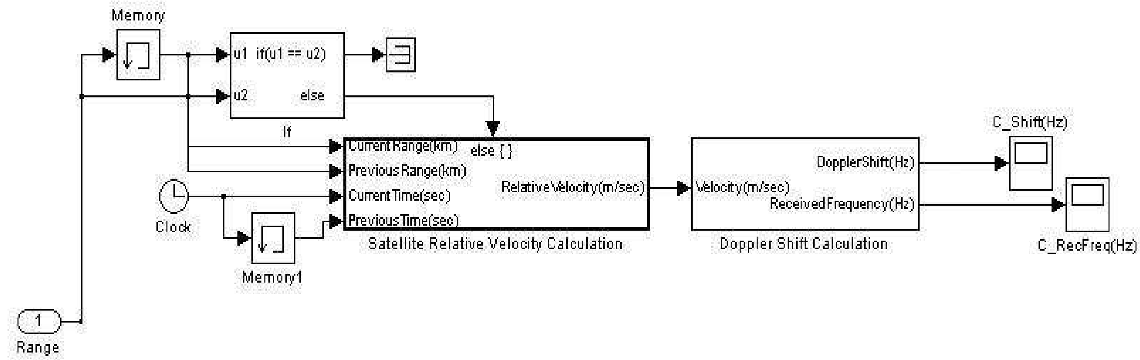

Figure 21: High-Level Doppler Calculation Subsystem

Figure 22 calculates the relative velocity by subtracting the previous range from the current range then dividing by the amount of time the satellite is required to move from the previous range to the current range.

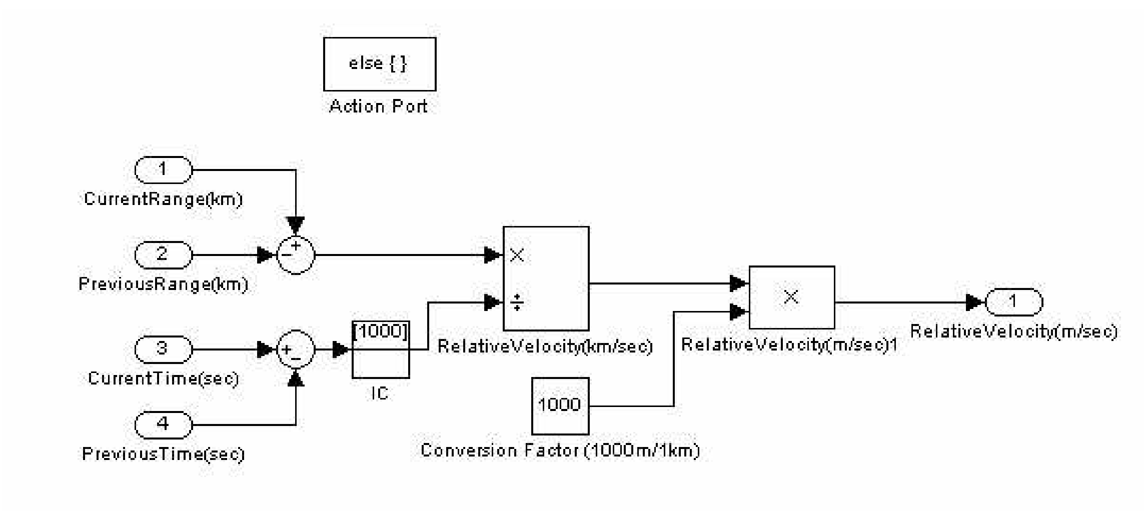

Figure 22: Satellite Relative Velocity Calculation Subsystem

The relative velocity is then inputted into the Doppler Calculation subsystem, Figure 23, which uses Equation (4-16) to calculate the Doppler shift.

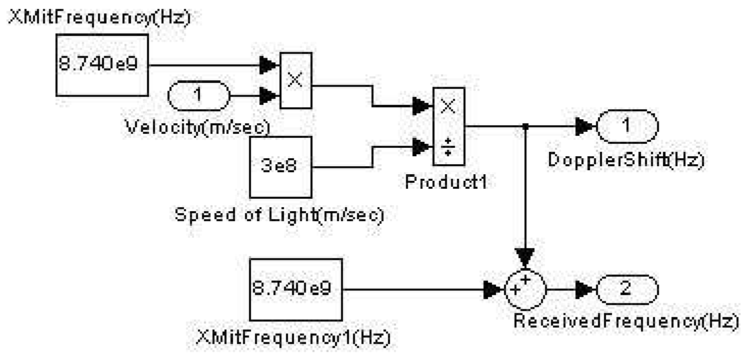

Figure 23: Doppler Shift Calculation Subsystem

4.2.4 Zone of Interference (ZOI) Calculation

The ST5 satellite is using a new technology, known as the Evolved Antenna which provides a much better antenna gain for the ST5 orbit. The satellite consists of two antennas (Top and Bottom) with each antenna providing a radiative pattern that describes the gain of antenna at a given point surrounding the satellite (360 degrees). The ZOI is caused by the two antenna patterns overlapping and creating destructive phase interference (Figure 24), which will result in a 3-4 minute (maximum 10 minutes) transmission loss.

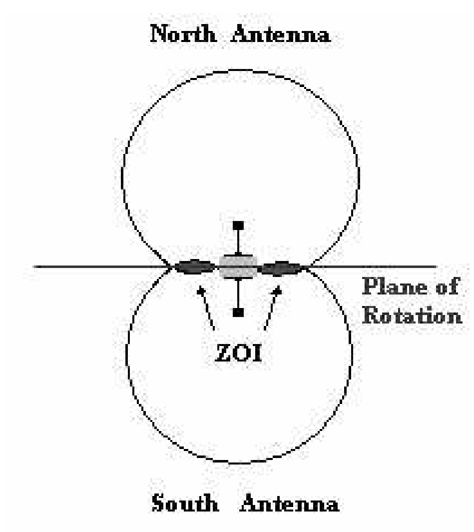

Figure 24: Zone of Interference

The Communications model is designed to determine the gain of the satellite antenna (G_r) based on the radiative pattern and the attitude of the spacecraft. The ZOI subsystem will determine when the ground station line of sight intersects the destructive pattern as well as output an accurate antenna gain for link margin calculation. In order to determine the satellite gain and ZOI, the angle that the line of sight intersects the satellite's spin axis must be determined. We will refer to this angle as the Radiative Pattern Angle (RPA). The antenna gain is a function of the attitude because the radiative pattern is fixed on the spin axis of the satellite. If the satellite attitude changes the radiative pattern moves as well. Determining the antenna gain and ZOI can be broken down into 3 steps: Topocentric Equatorial Conversion, Angle Calculation, and Radiative Pattern Modeling.

Inputs/Outputs

The ZOI calculator, shown in Figure 25, is dependent on two vectors: the vector pointing from the ground station to the satellite ($\vec{\rho}_{IJK}$), and the vector from the earth (ECI origin) to the ground station ($\vec{r}_{siteIJK}$). These two vectors are inputs to the Topocentric Equatorial Coordinates (TEC) conversion subsystem. The two attitude parameters, satellite declination (SatDec) and satellite right ascension (SatRA), are input into the radiative pattern angle calculation subsystem as an array structured with time. The attitude parameters describe the tilt of the spacecraft spin axis.

Since the ST5 satellites will be able to communicate with any of the four ground stations at a given time, the ZOI subsystem will determine the antenna gain for each ground station. The ZOI subsystem will then output the antenna gain which will be used for link margin calculations and zone of interference start dates and end dates.

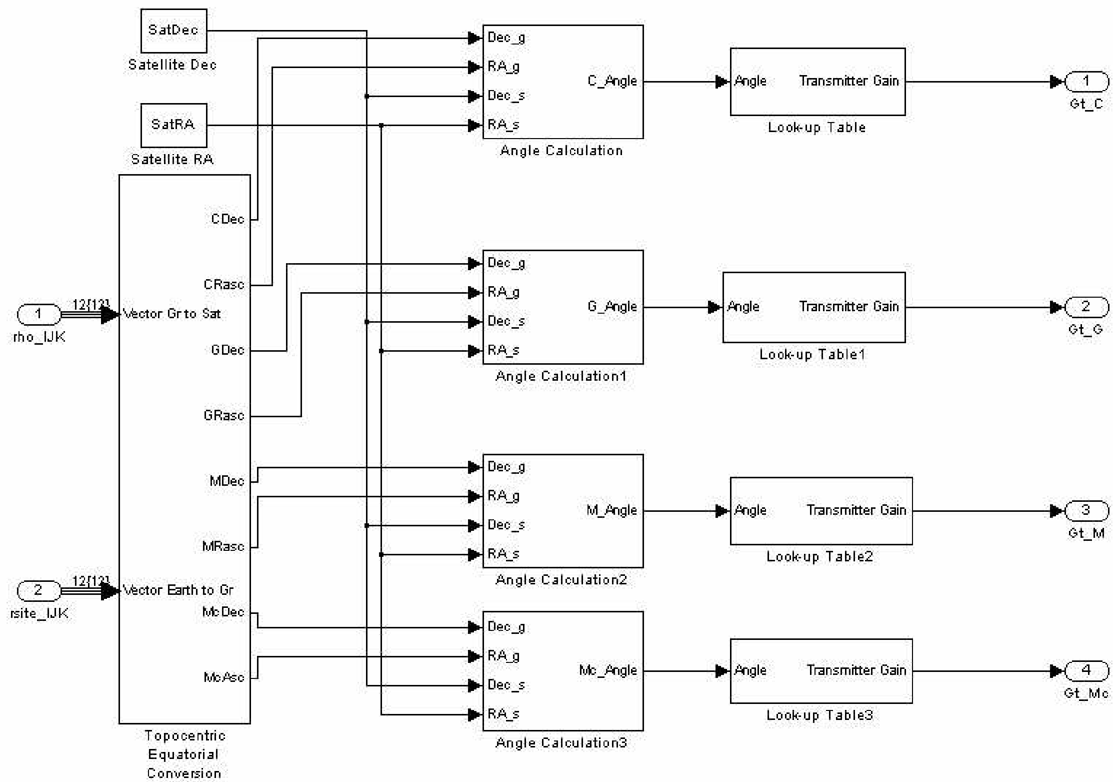

Figure 25: ZOI Calculator

Topocentric equatorial conversion (TEC)

The problem of determining the angle that the ground stations transmission intersects the radiative pattern is dependent on the ground station look angles and the satellite spin axis position. To determine the line of sight, the s-function, SFunGStations, calculates look angles from the SEZ coordinate system. The satellite attitude is provided as a declination and right ascension in ECI coordinates. For the RPA calculation to work, both the look angle and satellite spin axis must be in the same coordinate system. Therefore the model must convert the look angles from SEZ to TEC.

Topocentric Equatorial Coordinates, shown in Figure 26, are the same coordinates as ECI except the origin of the coordinate system is moved from the center of the Earth to a specific point on the Earth (ground station), keeping all axis parallel to each other. Given our two vector inputs we can determine our right ascension and declination in TEC as follows:

$$\vec{\rho}_{IJK} = \vec{r}_{IJK} + \vec{r}_{siteIJK}$$

$$\rho = |\vec{\rho}|$$

$$\sin(\delta_t) = \frac{\rho_k}{\rho}$$

IF $\sqrt{\rho_I^{\ 2} + \rho_J^{\ 2}} \neq 0$

$$\sin(\delta_t) = \frac{\rho_J}{\sqrt{\rho_I^{\ 2} + \rho_J^{\ 2}}} \quad \text{or} \quad \cos(\alpha_t) = \frac{\rho_I}{\sqrt{\rho_I^{\ 2} + \rho_J^{\ 2}}}$$

ELSE

$$\sin(\delta_t) = \frac{\dot{\rho}_J}{\sqrt{\dot{\rho}_I^{\ 2} + \dot{\rho}_J^{\ 2}}} \quad \text{or} \quad \cos(\alpha_t) = \frac{\dot{\rho}_I}{\sqrt{\dot{\rho}_I^{\ 2} + \dot{\rho}_J^{\ 2}}}$$

$$v_{SiteIJK} = \varpi_{\oplus} \times \vec{r}_{SiteIJK}$$

$$\dot{\vec{\rho}}_{IJK} = \vec{v}_{IJK} - \vec{v}_{SiteIJK}$$

(Vallado, 2001)

Where $\vec{r}_{IJK}$ is the vector pointing from the earths center to the satellite and $\vec{r}_{siteIJK}$ is the vector pointing from the earths center to the ground station. $\vec{\rho}_{IJK}$ can then be calculated as the vector from the ground station to the satellite. With the magnitude of $\vec{\rho}_{IJK}$, the right ascension (α_t) and declination (δ_t) of the satellite can be calculated. The ELSE statement takes care of the case when the satellite is pointed directly over the spin axis of the earth (z axis). For this case we need to know the velocity vector of the satellite which is taken from the orbit propagator. $\varpi_{\oplus}$ is the earth's rotational velocity and is often assumed to be constant.

$$\varpi_{\oplus} = 7.292115 \times 10^{-5} \pm 1.5 \times 10^{-12}$$

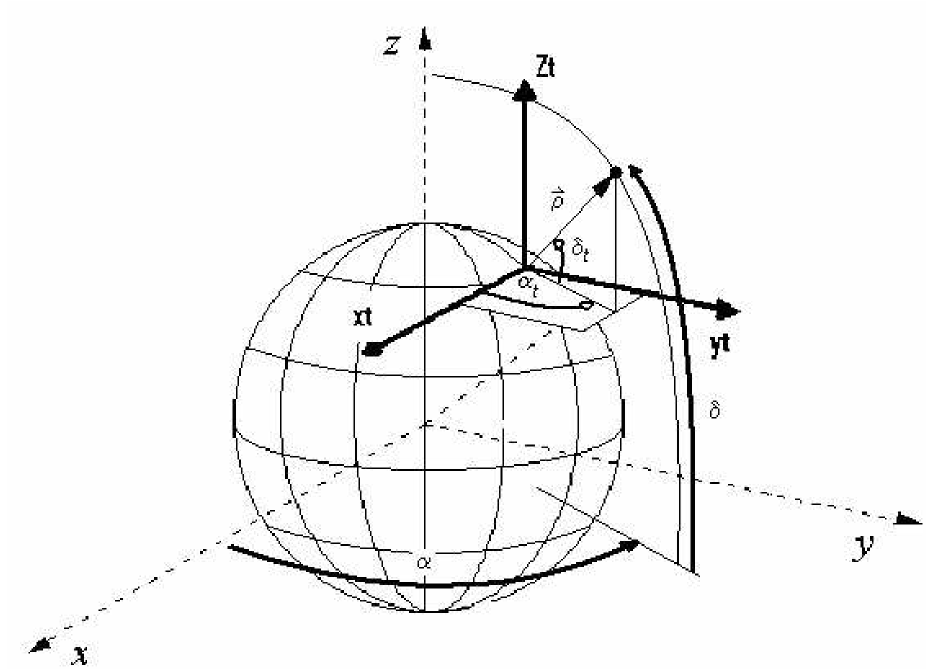

Figure 26: Topocentric Equatorial Coordinate System

These calculations are done using a MATLAB function called Topocentric.m. This function takes the two position vectors for each ground station calculation (12 total vectors) and the velocity vector as inputs from the s-function, SFunGStations. Topocentric.m outputs a right ascension and declination for each ground station (8 outputs).

Radiative Pattern Angle (RPA)

The RPA calculator (Figure 28) calculates the angle that the ground station line of sight intersects the radiative pattern. Figure 27 illustrates the RPA concept.

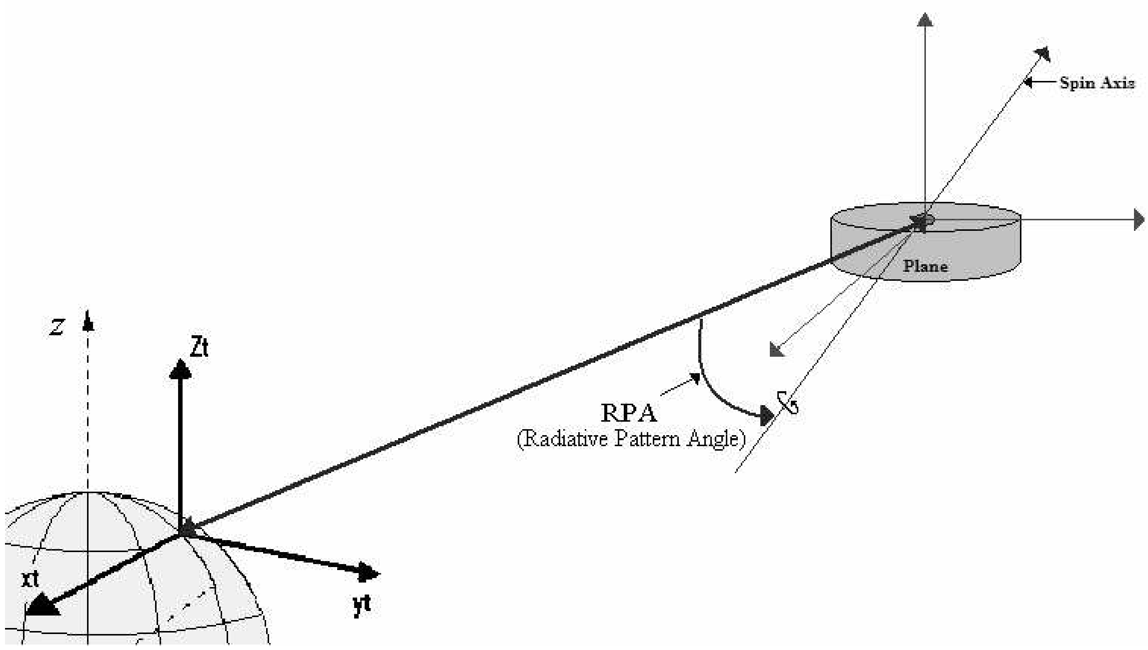

Figure 27: Radiative Pattern Angle

This calculation is dependant on the satellite attitude given as right ascension (RA_s) and declination (Dec_s). Also needed are the look angles in TEC, given as a right ascension (RA_g) and declination (Dec_g). The input angles give one vector describing the spin axis of the satellite and another vector pointing from the ground station to the satellite. The antenna radiative pattern is centered on the spacecraft spin axis; by calculating the angle between these two vectors we can determine the radiative pattern angle. Given these two vectors this angle can easily be determined using vector math. The first step is to take the right ascension and declination of each vector and convert to an x, y, and z Cartesian coordinate position.

$$x = d\cos\delta\cos\alpha \tag{4-17}$$

$$y = d\cos\delta\sin\alpha \tag{4-18}$$

$$z = d\sin\delta \tag{4-19}$$

Where α is equal to the right ascension, δ is equal to the declination and *d* is equal to the range. The range in this case is equal to 1.

By using Equations (4-17, 18, and 19) we now have a satellite spin axis vector, Sat_{xyz}, and a ground station look angle vector, LA_{xyz}. The radiative pattern angle can now be determined by solving for the angle between these two vectors using the following equation:

$$\theta = a\cos\frac{LA_{xyz} \bullet Sat_{xyz}}{\left|LA_{xyz}\right|\left|Sat_{xyz}\right|} \tag{4-20}$$

The Simulink model of Equation (4-20) is shown in Figure 28. The model inputs the necessary angles and gives the radiative pattern angle as an output. The equation will work only for situations when Dec_g and Dec_s are both less than zero. In any other scenario, Equation (4-20) will give the opposite angle. For these scenarios the RPA is easily corrected by adding 180º. Figure 29 shows the system that takes care of the two scenarios of the equation.

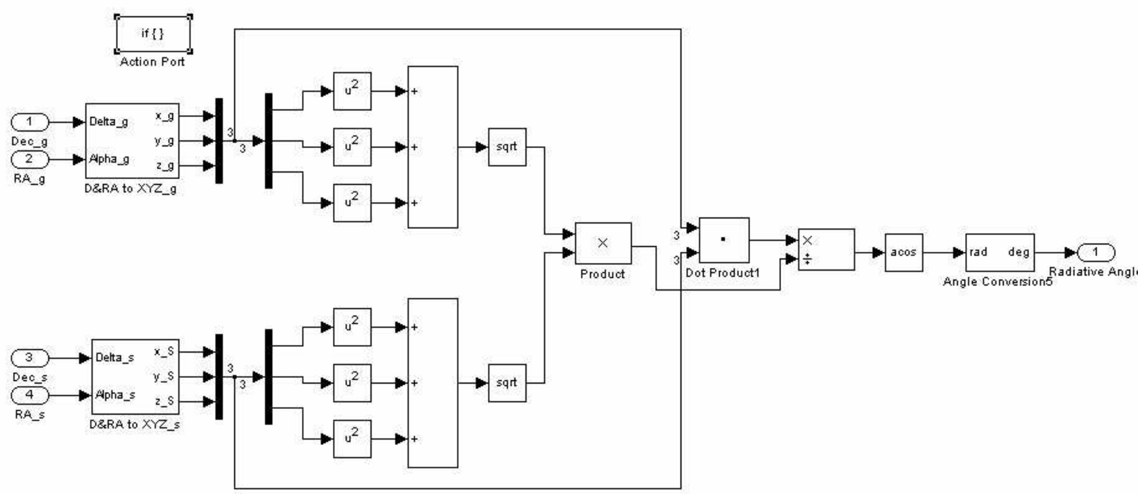

Figure 28: RPA model

Within the “IF” statement of Figure 29 is the RPA calculation subsystem. The “ELSE” statement simply adds 180° to the same calculation.

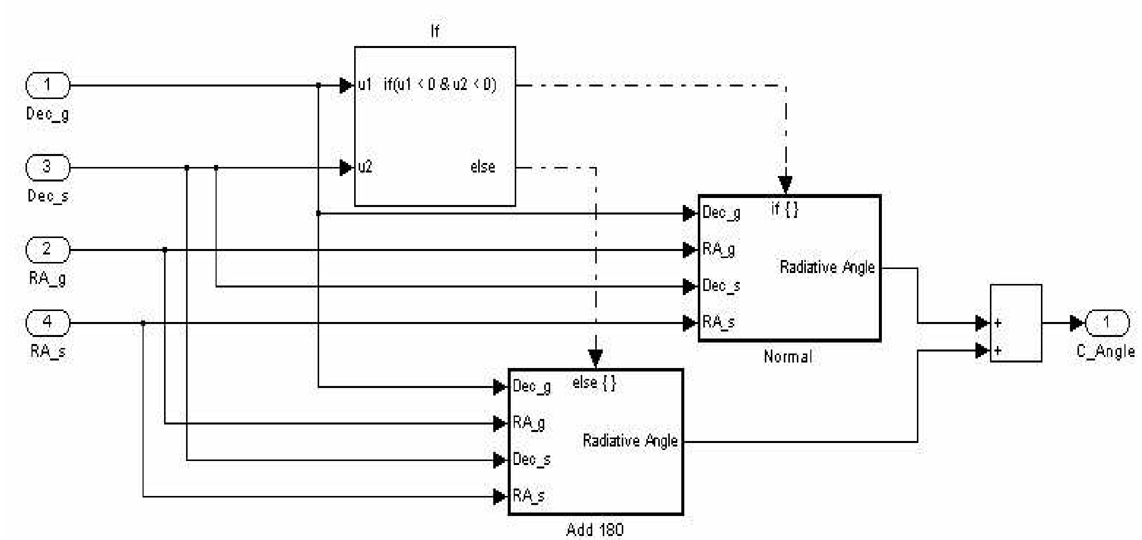

Figure 29: RPA Subsystem Calculator

Radiative Pattern Modeling

The EA antenna ST5 will be validating has not yet been fully tested and the results are not attainable. For our project we approximated gain values of the EA by examining the EA radiative pattern versus the QHA radiative pattern (Figure 30) given to us from the ST5 Critical Design Review files. Figure 31 shows an approximated sample EA patterns versus the old QHA patterns.

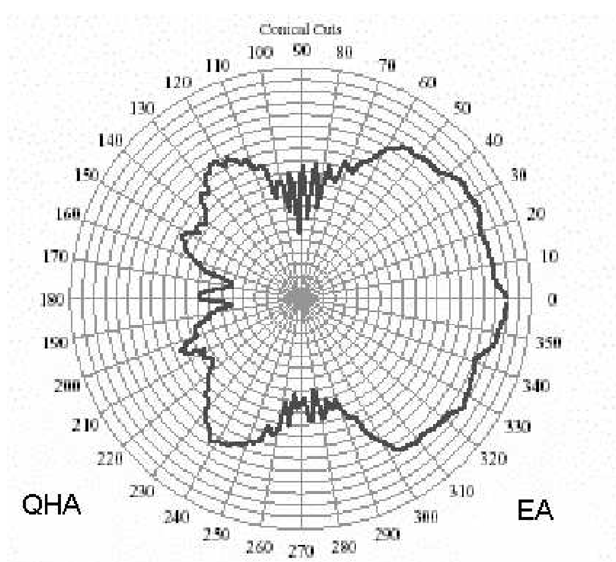

Figure 30: QHA Patterns vs. EA Patterns

(ST5 ΔCDR)

The EA has much better gain capabilities compared to the QHA in LEO. From this figure we were able to construct our approximation of the EA radiative pattern.

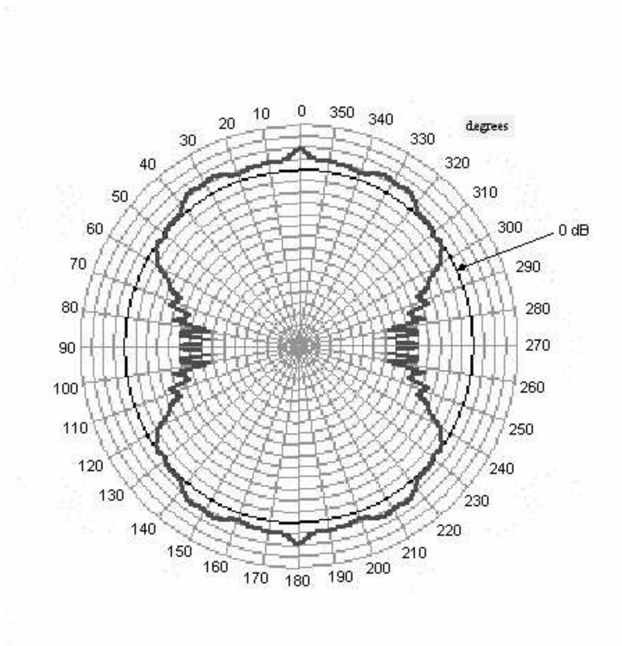

Figure 31: EA Radiative Pattern

The pattern is now rotated to show the ZOI along the x-axis centered along the 90-270 degree axis. The rotational axis of the spacecraft, or z-axis, is aligned with 0-180 degrees. Due to the constant rotation of the spacecraft about the z-axis we will be unable to determine where the radiative pattern angle will intersect the three-hundred sixty degree radiative pattern. For example, we can not distinguish the difference between 20 degrees and 370 degrees. To solve this problem we created a 180 degree pattern and then mirrored this pattern. We compared the mirrored angles of the 360 degree pattern and recorded the worst case scenario.

From this 180 degree radiative pattern we can create a look up table to correspond a certain angle to a certain transmission gain. This look up table was created using an excel spreadsheet called “Look_up_EA”, and imported into our Simulink model using the

MATLAB function xlsread.m. The excel spreadsheet allows for easy modifications due to future updated antenna patterns.

The previously calculated radiative pattern angle from our "Angle Calculation" subsystem is inputted into our look up table. From the look up table a transmission gain is outputted. Figure 32 shows the Simulink subsystems needed for this computation. The Look-Up Table subsystem is an already existing Simulink function. The excel spreadsheet we created is called from inside the Look-Up Table function.

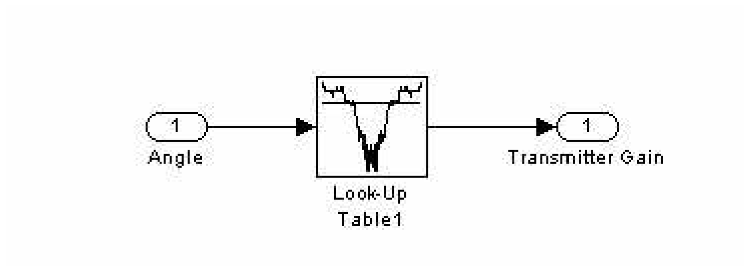

Figure 32: Look-Up Table

The output gain is then used to determine line of sight logic, link margin, Doppler shift and Zone of Interference dates.

4.2.5 Generating Output files

The Communications model will output eight separate excel spreadsheets, along with two separate graphs, of the each ground stations visibilities dates and each ground stations ZOI visibility dates. The excel spreadsheets can easily be viewed after the model is run, and the graphs can be viewed by using the scopes placed in the top-level diagram of the Communications model.

Date/Time Information

In order to produce exact date/time information, a calendar/time function was added to the model in the Date/Time input subsystem.

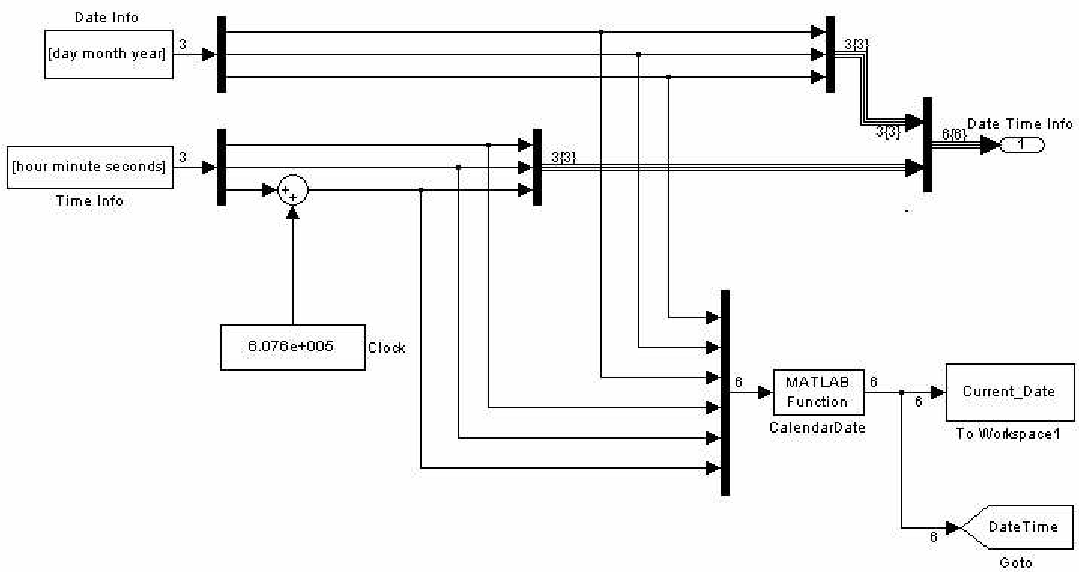

Figure 33: Date/Time Information

Figure 33 is the Simulink model of the Date/Time Info subsystem. There are six inputs to this model. Day, month, year, hour, minute and seconds are all initialized in the first received IIRV file. The clock is continuously running and added to the seconds input to provide updated passing of time for the simulation. The MATLAB function CalendarDate.m takes in the year, month, day, hour, minute, and seconds as a 1 x 6 array and examines the amount of seconds passed and converts it to a real date. This function will increment the year, month, day, hour, and minute depending on how many seconds have passed. The function accounts for each month having a different amount of days and also accounts for leap years. The newly generated date is then is written to the workspace and outputted as a global flag used to determine the line of sight dates, Zone of Interference dates.

Line of Sight Dates

Line of sight dates are needed for future communications planning. The line of sight visibility graph shows logic 1 (visible) and logic 0 (not visible). It does not give specific visibility dates. Using the line of sight logic graph and the date/time information the dates of visibility can be determined.

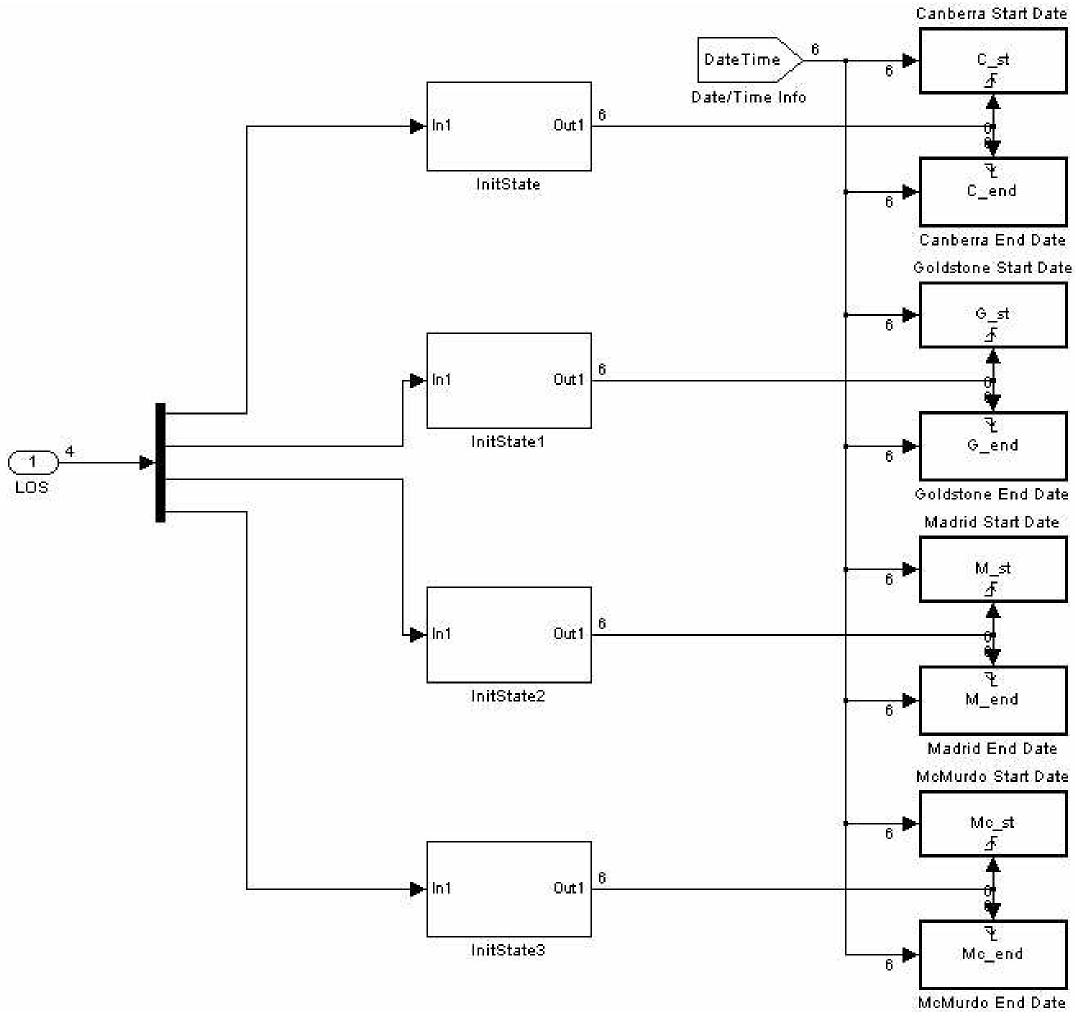

Figure 34: Start/End Dates for Line of Sight

Figure 34 shows the line of sight start and end date calculation subsystem. The start and end dates are measured by using a sample and hold system. The sample and hold system has two inputs: the date/time information and the line of sight logic. The system monitors the line of sight logic and associates a start date with a rising edge and an end date with a falling edge. Eight separate work spaces are created using this method. A MATLAB function called "output_times" is used to combine the start and end dates of a single ground station and then print it as an excel spreadsheet for viewing. The "InitState" subsystems account for the possibility of a logic 1 as an initial position. If an initial position of logic 1 is present then the initial time is outputted as a start date.

Zone of Interference Graph and Dates

In order to determine the Zone of Interference dates and graphs, the elevation of the ground station in SEZ and the radiative pattern gain are desired. Elevation is required because it is not necessary to determine Zone of Interference dates and graphs if the satellite is not visible by the ground station. The radiative pattern gain is required because if the satellite is visible by the ground station, then the radiative pattern gain must be sufficient enough to establish communication. If the radiative pattern gain is insufficient then the line of sight falls in the Zone of Interference.

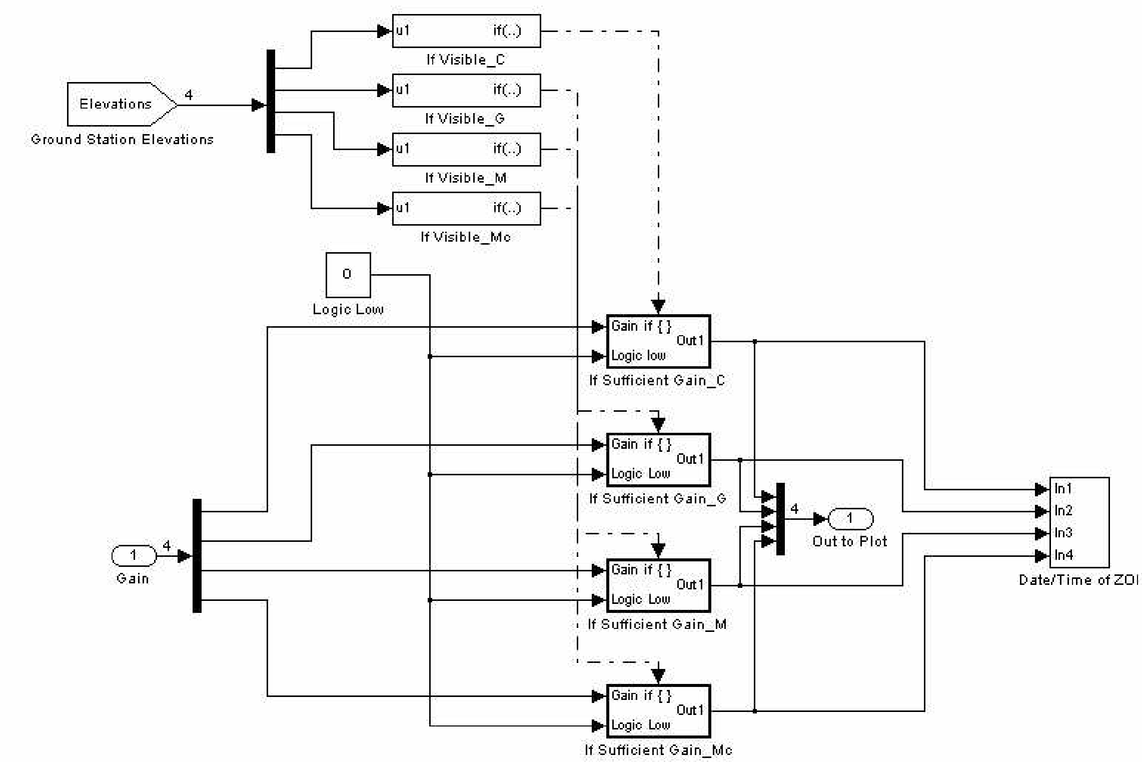

Figure 35: Zone of Interference Plot and Date Calculation

The elevations of each ground station are inputted from the Elevations global flag. If the satellite is visible by the ground station the model assesses the gain and provides logic 1 for a failed communication and logic 0 for a successful communication. If the gain is less than or equal to -10 dB then it is a failed communication and the line of sight falls in the zone of interference. The “Date/Time of ZOI” subsystem is another series of sample and hold systems that supply the dates for the start and end of the Zone of Interference. These dates are also written to a workspace and outputted as excel spreadsheets.

4.2.6 Communications Model Verification

To ensure the Communications model capabilities, each section was tested individually before being integrated into the final model. By validating the model it will limit errors that may happen when using the final product, since the final model can not be tested without flight validation.

When verifying the look angle calculations we applied the same methodology that the 2002 ST5 MQP did. They used a scenario given in Orbital Coordinate Systems, Part III, written by Dr T.S. Kelso, where the date was given as 18 Nov 1995, the time 12:46 pm, ground site latitude was 45° N and longitude was 93° W, and the satellite ECI position was [-4400.594, 1932.870, 4760.712]. With these given values, Dr. Kelso computed the Azimuth as 100.36° and the Elevation as 81.62°. The communications subsystem, using the same given input values, calculates the Azimuth as 102.92° and the Elevation as 83.11°. Since we used the same method as the 2002 ST5 MQP, our results are the same. The Azimuth calculation error is 2.55% and in the Elevation calculation error is 1.83%. This error includes any error generated by the latitude/longitude to ECI coordinate conversion.

Similarly we also used the same method for testing the ground station latitude/longitude to ECI coordinate conversion as the 2002 ST5 MQP. This was verified using another scenario given by Dr. Kelso, where the date was 01 Oct 1995, the time was 9:00 am, and the latitude and longitude were 40° N and 75° W, respectively. The ECI coordinates given by Dr Kelso were [1703.295, 4586.650, 4077.984]. The ECI coordinates calculated by the Communications model were [1694.65, 4589.85, 4077.99]. The error in the ECI coordinates was [.51%, 0.07%, 0.00%] and the total position error was 9.22 km.

To test the Link Margin Calculator we compared the results to the ST5 RFICD provided to us by Victor Sank. The RFICD was last revised on April 2, 2004. We first calculated the available signal to noise ratio and then subtracted the required signal to noise ratio to find the link margin. Space loss, antenna gain, transmitter power, passive loss, atmospheric loss, implementation loss, receiver antenna gain, system temperature, Boltzmann's constant, and the data rate are all needed to calculate the available signal to noise ratio. The calculated available signal to noise ratio from the RFICD is equal to 65.04 dB and the Link Margin is equal to 55.44 dB. The calculated available signal to noise ratio from our model is equal to 64.91 dB and the Link Margin is equal to 55.31 dB. The total error between calculations was 0.2%.

To test the "Topocentric Equatorial Conversion" function we used Fundamentals of Astrodynamics and Applications written by David A. Vallado. Chapter 4, example (4-1) shows the conversion from SEZ coordinates to TEC. Our MATLAB function has three inputs which are the vector from the satellite to the ground station ($\vec{\rho}_{IJK}$), the vector from the ground station to the satellite ($\vec{r}_{siteIJK}$), and the velocity vector ($\vec{v}_{IJK}$), also known as the position and velocity vectors. From the Vallado book we are given $\vec{r}_{IJK}$, $\vec{r}_{siteIJK}$, and $\vec{v}_{IJK}$ as test values. $\vec{\rho}_{IJK}$ can be determined using this equation:

$$\vec{\rho}_{IJK} = \vec{r}_{IJK} + \vec{r}_{siteIJK}$$

From this we can now enter the three inputs into the MATLAB function Topocentric.m. For this test we will be using Vallado's vectors and angles as test values which are shown in Table 2:

SEZ (Topocentric Horizon Coordinate System)		TEC (Topocentric Equatorial Coordinate System)	
ρ_{ijk} (km)	[1,752,242,148.28 -3,759,560,585.46 -1,577,572,099.3]	Right Ascension α (degrees)	294.989 111 5
$r_{site(ijk)}$ (km)	[4,066.716 -2,874.545 3,994.302]	Declination δ (degrees)	-20.823 562 3
v_{ijk} (km/s)	[-18.324 18.332 7.777]		

Table 2: Test vectors and angles

Given the SEZ coordinates we can now test our model. When executed the model outputs a right ascension of -65.01 degrees and a declination of -20.82 degrees. The only difference is the right ascension where 294.99 degrees and -65.01 degrees are not the same number but are equal to the same angle on the x-y plane. The MATLAB function only holds two decimal places compared to the seven decimal places that Vallado gives. This error will amount to an error of less than 1% in both cases.

The RPA Calculation was tested by using sample numbers for the ground station and satellite right ascension and declinations. For this test the ground stations right ascension and declination was 180° and -45° respectively and the satellite right ascension and declination was 90° and -87° respectively. These numbers were chosen because they are simple to solve and can be solved by using a geometric approach as well as a calculated approach. When using the geometric approach the approximated RPA was 48°. The hand calculated RPA which was determined by using Equations (4-17) through (4-20), was exactly the same as the Simulink models answer of 45.08°. The equations used to calculate the RPA were directly implemented into the Simulink subsystem, which provides accurate solutions.

The look-up table was tested by entering values for the RPA input. Transmission gain values of the look-up table output were hand checked against the excel spreadsheet that was created. All input values corresponded to the appropriate output values.

4.3 Orbit Propagator

The orbit propagator provides position and velocity vectors describing the satellites orbit which are then used by the Communications model to determine look angles and line of sight logic. The orbit propagator was created by Jim Morrissey (ST5 Attitude Control System Analyst, jmorriss@pop500.gsfc.nasa.gov) and is the same propagator used in the previous Communications models. The model uses some basic physics with an initial position and velocity vector to determine the satellite orbit. The algorithm is based on two body orbital mechanics, earth and satellite. The orbit propagator was designed to provide short term propagation, of approximately 3-4 hours of accuracy. Newly integrated Improved Inter Range Vector files will be received every four hours updating the orbit propagator by providing new position and velocity vectors. These vector files are received from a NASA generated satellite tracking system.

4.3.1 Algorithm

Newton's 2nd law states that the rate of change of momentum is proportional to the forces impressed and is in the same direction as that force. We know the force impressed on the satellite due to our two-body model is the gravitational force from the earth. Newton's Law of Universal Gravitation states that any two bodies attract one another with a force proportional to the product of their masses and inversely proportional to the square of the distance between them. Newton's 2^{nd} law also states that the gravitational force from the earth would cause the satellite to accelerate toward the center of the earth.

Given Newton's Laws we are able to calculate acceleration given an initial satellite position vector. Newton's Universal Gravitational law is a function of mass and distance. Therefore given the mass of the earth and the satellite, as well as an initial position, the acceleration of the satellite can be calculated. Calculation of the acceleration requires some vector math with a position vector input. Figure 36 shows the necessary vector math used to calculate acceleration vector due to gravity, based on a position vector.

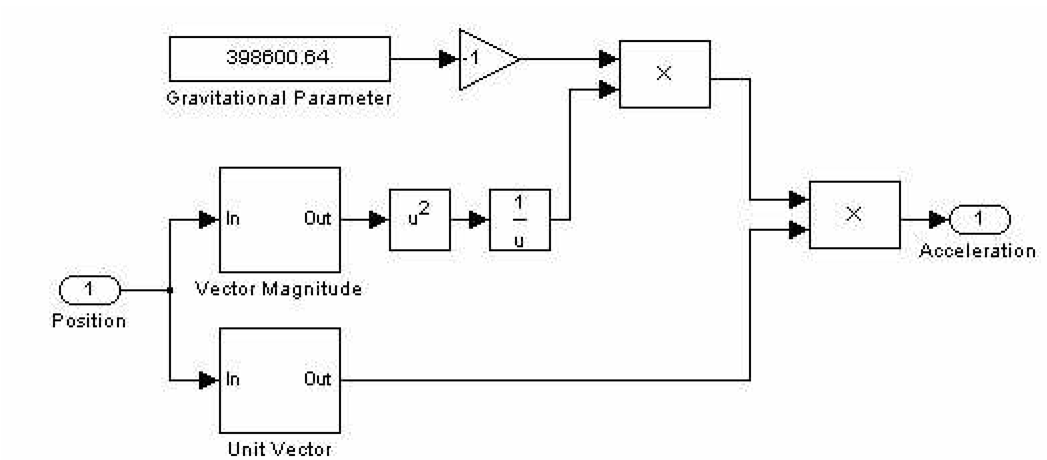

Figure 36: Acceleration Vector Calculation

By integrating the given acceleration vector a velocity vector derived. We can then integrate the velocity vector to receive the next position vector. The integration steps are shown in Figure 37.

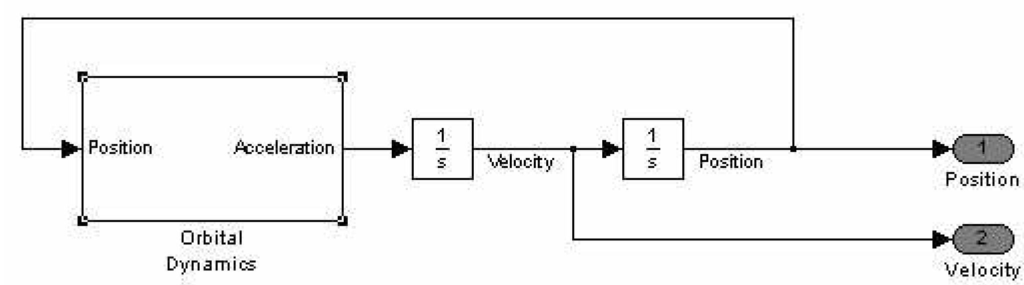

Figure 37: Position Vector Calculation

Once the new position vector is calculated it is fed back into the "Orbital Dynamics" subsystem to recalculate a new acceleration vector and the process continues. If the orbit propagator is run for a long enough time, the model will create enough position vectors to propagate a whole orbit.

When the position and velocity vectors are created we need a way to access the information to be used for the Communications model. To do this separate Simulink subsystem was created that will take both vectors and structure them with time. Figure 38 shows the satellite ECI coordinates created by the orbit propagator stored in an array with the simulation time that corresponds to each vector. The array is then sent to the MATLAB workspace where it can be used by other models.

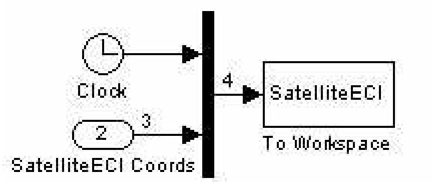

Figure 38: Satellite Coordinates to Workspace

4.3.2 Improved Inter Range Vector (IIRV)

The Communications model is designed to provide long-term mission planning, therefore we require an orbit propagator that provides accuracy beyond a 1-2 hour time frame. One way to improve the orbit propagator accuracy is to incorporate IIRV's. An IIRV is a message file provided by the ground station that tracks the satellite. The IIRV contains six lines of ASCII code that provide multiple satellite parameters. The IIRV's are sent centered around every 4 hours that we will use to provide a satellite position and velocity vector for a given time in orbit. For long term mission planning future IIRV's are predicted and propagated to provide orbit position and velocity vectors for the entire mission duration.

The model takes a set of IIRV's and interfaces them with MATLAB. The MATLAB function St5_mod.m locates the set of IIRV's and parses the data into variables that can be used for the orbit propagator. The set of IIRV's provide discrete points in the orbit describing position and velocity. The orbit propagator can now be used to create position and velocity vectors for points between the IIRV inputs. This was accomplished by creating a MATLAB function IIRV.m that examines the date of each individual IIRV and propagates all points until the time of the next IIRV is received. Figure 39 helps illustrate this concept.

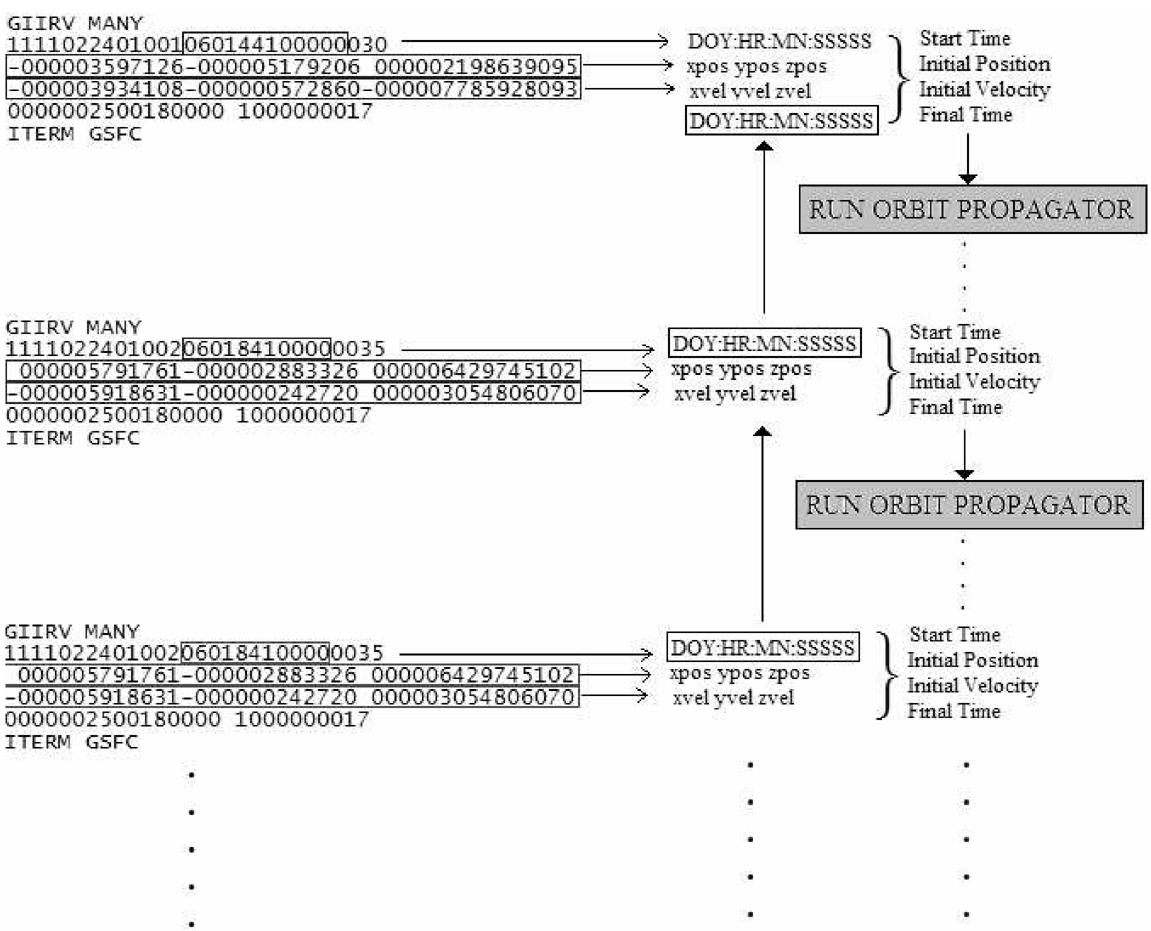

Figure 39: IIRV.m Algorithm

The IIRV data on the left from Figure 39 is stored in the MATLAB workspace created by St5_mod.m. The orbit propagator requires 4 inputs:

1. Starting Time
2. Initial Position Vector
3. Initial Velocity Vector
4. Final Time

The function IIRV.m reads the first IIRV (Blue) which provides a Starting Time, Initial Position and Initial Velocity for the orbit propagator. The Final Time is recorded when the next IIRV occurs. In order to accomplish this, the function imports the start time of the next IIRV (Red) and uses it as a Final Time. The second column from Figure 39 shows the format the inputs are received in.

1. Start Time - Day of year (DOY): Hour (HR): Minute (MN): Seconds (ss.sss)
2. Position Vector – [xpos ypos zpos]
3. Velocity Vector – [xvel yvel zvel]
4. Final Time - Day of year (DOY): Hour (HR): Minute (MN): Seconds (ss.sss)

The orbit propagator can now create position and velocity vectors for the 3-4 hour period between the IIRV inputs. The orbit propagator stops when it reaches the final time and then reads in the next IIRV (red). The Final Time of the previous simulations becomes the new Start Time while the next IIRV (green) will provide the new Final Time. This process continues until there are no more available IIRV's.

It should be noted that for the real-time state health assessment mode the IIRV's are not all received at once due to real-time. The orbit propagator would receive one IIRV and propagate based on that single input. The Simulink model would then run until the next IIRV is ready to be received.

4.3.3 Earth Centered Fixed (ECF) to ECI Conversion

An intermediate step exists before the IIRV's can be used. The IIRV position and velocity vectors are provided in the Earth Centered Fixed (ECF) coordinate system. Our

Communications model is dependant on the orbit propagator inputs being in ECI coordinates. Therefore before IIRV.m can be run the IIRV input vectors must be converted from ECF to ECI.

The transformation of an ECF position vector $\mathbf{r}_{ecf}$ to an ECI position vector $\mathbf{r}_{eci}$ is given by the following vector-matrix operation.

$$\mathbf{r}_{eci} = [\mathbf{T}]^T \mathbf{r}_{ecf} \tag{4-21}$$

The transformation matrix **[T]** is dependant on the Greenwich Sidereal Time at the given moment of interest. The elements of the transformation matrix **T** are given by

$$[\mathbf{T}] = \begin{bmatrix} \cos\theta & \sin\theta & 0 \\ -\sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{bmatrix} \tag{4-22}$$

Where θ is the Greenwich Sidereal Time (GST). GST is defined as,

$$\theta = \theta_{g0} + \omega_e t \tag{4-23}$$

where θ_{g0} is the Greenwich sidereal time at 0 hours UT, ω_e is the inertial rotation rate of the Earth, and t is the elapsed time since 0 hours UT. The ECI velocity vector is determined by differentiating the following expression.

$$\mathbf{v}_{eci} = [\mathbf{T}]^T \dot{\mathbf{r}}_{ecf} + [\dot{\mathbf{T}}]^T \mathbf{r}_{ecf} = [\mathbf{T}]^T \mathbf{v}_{ecf} + [\dot{\mathbf{T}}]^T \mathbf{r}_{ecf} \tag{4-24}$$

The elements of the matrix $[\dot{\mathrm{T}}]$ are as follows:

$$\left[\dot{\mathrm{T}}\right]=\begin{bmatrix} -\omega_e \sin\theta & \omega_e \cos\theta & 0 \\ -\omega_e \cos\theta & -\omega_e \sin\theta & 0 \\ 0 & 0 & 0 \end{bmatrix} \tag{4-25}$$

To convert from ECF to ECI we utilized the MATLAB orbital mechanics toolbox provided by Marco Concha (ST5 Guidance and Navigation Control, Marco.Concha@nasa.gov). From this toolbox we created the function ecf2eci.m. Inputs to this function are a date/time array (year month day hour minutes seconds), ECF position vector, and an ECF velocity vector. The ECF input vectors are used directly by Equations (4-21) and (4-24). The date/time array is input to the function juliandate.m to calculate a Julian date. The Julian date is then inputted into the function GAST.m which calculates the apparent Greenwich Sidereal Time that is used for the matrix transformations (Equations 4-22, 4-25).

4.3.4 Orbit Propagator Model Verification

To verify the functionality of the orbit propagator we plotted the satellite position vectors versus time.

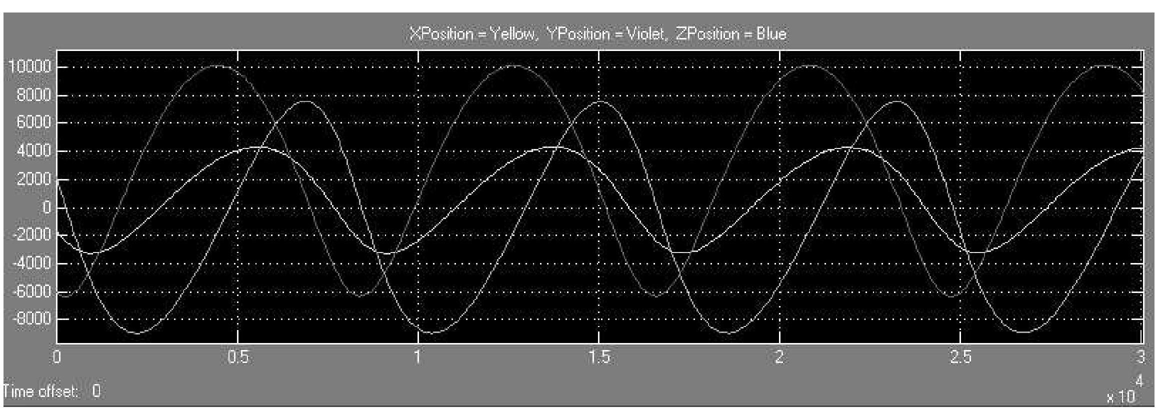

Figure 40: Satellite x-y-z position plot

Figure 40 shows each component of the position vector plotted versus time. From the graph it can be determined that the orbit propagator is working properly due to the sinusoidal motion of the position. The period of each sinusoid shows one full orbit of the satellite. Since the satellite completes one orbit in 136 minutes, the period of each sinusoid from Figure 40 should be 136 minutes (8160 seconds). The first peak of the x-position occurs at 5560 seconds and the second peak occurs at 13750 seconds. The elapsed time between the two peaks gives a period of 8160 seconds which is equal to the orbital period of the spacecraft of 136 minutes.

4.4 Model integration to Simulink ST5

ISR has developed the Simulink ST5 model which consists of the solid state data recorder subsystem and the command and data handling subsystem. The C&DH subsystem contains all the necessary telemetry data needed for the ST5 communications model. They have also provided the necessary steps that are needed to add Simulink Models to their existing SimulinkST5. All that is needed is the proper Simulink model information that can be placed in an xml configuration file (ST5config.xml). This information consists of the GMSEC profile directive related to the model, model identity, model location, file, name, and optional initialization files. The next step is to add the subsystems initial conditions and telemetry mnemonics to the ST5config.xml file. This allows our model to use spacecraft and/or ground station telemetry as well as set up necessary initial conditions. Once our model has been executed, the output information is sent back to the bus for use by other applications.

4.5 Summary

The original SimulinkST5 model was modified by upgrading the existing orbit propagator to provide mission planning for up to three months. The existing Communications model was also improved to include Zone of Interference calculations, antenna radiative pattern calculations, spacecraft attitude adjustments, simulated calendar information, and dates of communication blackouts. The new functions were incorporated into the model by adding subsystems and MATLAB functions into the existing SimulinkST5 setup.

The Communications model is now able to calculate when the ST5 satellite is in the Zone of Interference. This is done by examining ground station elevations and satellite antenna gains determined from the antenna radiative pattern. The spacecraft attitude is also used in the radiative pattern calculations. Calendar dates of communication blackouts are given in year, month, date, hour, minute and seconds which are printed out as excel spreadsheets for monitoring.

The restructured orbit propagator allows for updated satellite position and velocity vectors every four hours from IIRV files. This new method replaces the old orbit propagator which was only accurate for 2-3 days of propagation which did not meet long term planning requirements.

5. Results

5.1 Introduction

The SimulinkST5 Communications model was created to supply data to support short term and long term modeling of the ST5 communications link. The Communications model consists of two major subsystems, the orbit propagator and the Communications model. The model generates several outputs that are written to the workspace. Among the outputs is a line of sight graph, a Zone of Interference graph, a link margin graph, a Doppler shift graph, and excel spreadsheets providing the exact visibility dates of the line of sight and Zone of Interference. The generated outputs will provide the user with the ability to plan for future communication links with the satellite.

The orbit propagator model was modified and tested in order to provide an accurate position and velocity of the satellite to be used by the communications model to determine the link margin, Doppler shift, line of sight, and Zone of Interference. The orbit propagator must be run before the Communications model to provide ECI position vectors for look angle calculations. The orbit propagator was tested by plotting the orbit in MATLAB, and by plotting the x, y, and z position of the satellite.

The Communications model output was verified by comparing the actual visibility dates, created by Marco Concha using STK software, with the Simulink produced visibility dates. The Zone of Interference graph and dates can not be tested because there is no data to compare it to.

5.2 Communications Model

Line of Sight

When verifying the line of sight we consider the ground station look angles and link margin as the only factor of visibility. Marco Concha's line of sight spreadsheets are based on when the ground station is able to detect the satellite based only on look angles. In order to view the ground station line of sight, the logic was written to a workspace and then plotted using a MATLAB script file. Figure 41 shows the line of sight output obtained from the Communications model.

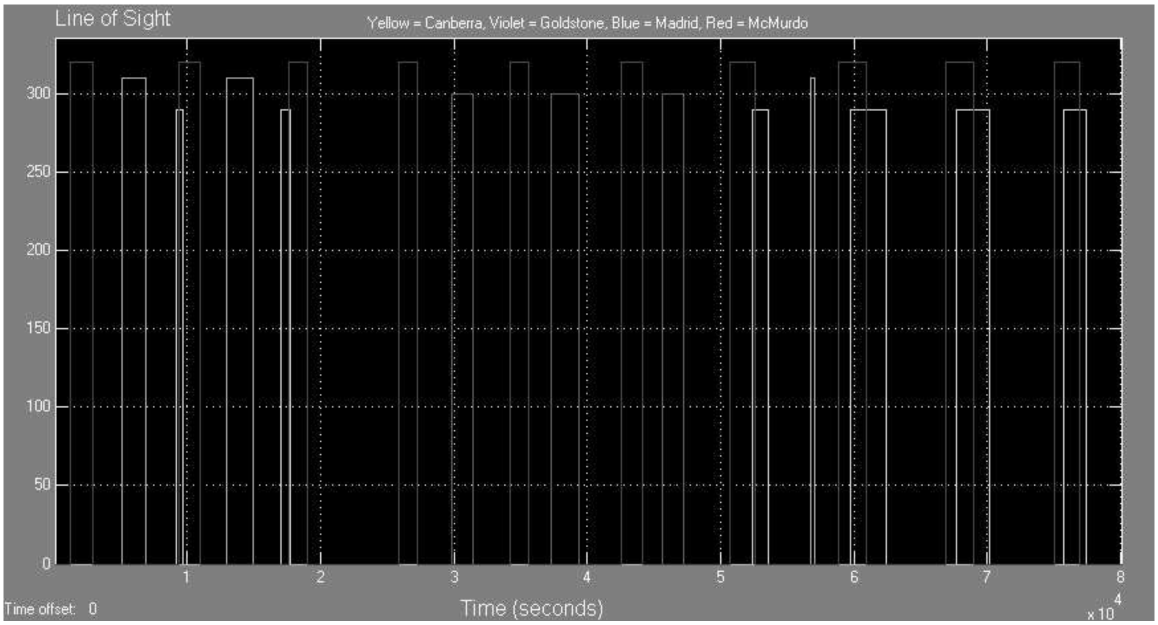

Figure 41: Ground Station Line of Sight

The elapsed time is measured along the x-axis in units of second. The y-axis reads a logic high for visible and a logic low for not visible. We used arbitrary numbers as logic high for each ground station to distinguish between them. Yellow corresponds to the Canberra ground station, violet corresponds to Goldstone, blue corresponds to Madrid, and red corresponds to McMurdo. McMurdo is visible most frequently because of its location with respect to the apogee of the spacecraft orbit. McMurdo is available for approximately 25 minutes each visibility pass.

From the line of sight graph we were able to create an output file that listed the start and end dates of each ground station visibility pass. To do this the Simulink model tracks start and end dates of each visibility pass and stores the data to the MATLAB workspace. Once the information is in the workspace a MATLAB script can be used to write the workspace into an excel spreadsheet with a desired format that we specified. The

Simulink excel spreadsheet can then be compared with the excel spreadsheet given to us by Marco Concha. Figure 41 shows the excel spreadsheet given to us by Marco. Figure 43 shows the excel spreadsheet generated by SimulinkST5.

	A	B	C	D	E	F	G
1	ST5 View Period 120 days from Launch						
2	Antenna Mask ON, Slew rate limits ON, ST5 North Antenna ON, Elevation Limit OFF						
3							
4	1	3/1/06 15:05	3/1/06 15:33	28.442	McMurdo		
5	2	3/1/06 16:02	3/1/06 16:17	14.226	Madrid-DS65		
6	3	3/1/06 17:13	3/1/06 17:24	10.542	Canberra-DS45		
7	4	3/1/06 17:20	3/1/06 17:46	26.136	McMurdo		
8	5	3/1/06 18:12	3/1/06 18:31	19.018	Madrid-DS65		
9	6	3/1/06 19:27	3/1/06 19:37	9.475	Canberra-DS45		
10	7	3/1/06 19:36	3/1/06 20:02	25.189	McMurdo		
11	8	3/1/06 20:38	3/1/06 20:46	7.621	Madrid-DS65		
12	9	3/1/06 21:53	3/1/06 22:18	25.056	McMurdo		
13	10	3/1/06 22:54	3/1/06 23:04	10.595	Goldstone-DS15		
14	11	3/2/06 0:11	3/2/06 0:32	21.066	McMurdo		
15	12	3/2/06 1:01	3/2/06 1:19	18.443	Goldstone-DS15		
16	13	3/2/06 2:28	3/2/06 2:56	28.004	McMurdo		
17	14	3/2/06 3:17	3/2/06 3:34	16.574	Goldstone-DS15		
18	15	3/2/06 4:45	3/2/06 5:18	32.71	McMurdo		

Figure 42: Excel Spreadsheet provided by Marco Concha

The highlighted rows of Figure 42 are the McMurdo start and end dates for ground station line of sights. The start dates are listed in column B and the end dates are listed in column C. Column D lists the duration of visibility.

	A	B	C	D	E	F	G	H	I	J	K	L	M
1							McMurdo Ground Station						
2			Start Time							End Time			
3													
4	Year	Month	Day	Hour	Minute	Second		Year	Month	Day	Hour	Minute	Second
5													
6	2006	3	1	15	1	0		2006	3	1	15	29	20
7	2006	3	1	17	17	40		2006	3	1	17	42	40
8	2006	3	1	19	34	20		2006	3	1	19	57	40
9	2006	3	1	21	52	40		2006	3	1	22	14	20
10	2006	3	2	0	11	0		2006	3	2	0	34	20
11	2006	3	2	2	29	20		2006	3	2	2	56	0
12	2006	3	2	4	46	0		2006	3	2	5	17	40
13	2006	3	2	7	1	0		2006	3	2	7	36	0
14	2006	3	2	9	16	0		2006	3	2	9	51	0
15	2006	3	2	11	31	0		2006	3	2	12	2	40
16	2006	3	2	13	44	20		2006	3	2	14	14	20
17	2006	3	2	16	1	0		2006	3	2	16	27	40
18	2006	3	2	18	16	0		2006	3	2	18	41	0
19	2006	3	2	20	34	20		2006	3	2	20	56	0
20	2006	3	2	22	52	40		2006	3	2	23	14	20

Figure 43: Simulink ST5 Generated Excel Spreadsheet

The SimulinkST5 generated output shows only the McMurdo visibility dates. Each ground station was separated into its own individual spreadsheet unlike the spreadsheet that was given to us. The SimulinkST5 spreadsheet is divided into two major columns, a start date and an end date. The dates are listed in year, month, day, hour, minute, and seconds. To create this spreadsheet, for this particular example, the user will execute a MATLAB script file by typing McMurdo.m in the command prompt. Other ground stations can be accessed by typing these commands:

- Canberra.m
- Goldstone.m
- Madrid.m

When analyzing the start and end dates from Figure 42 and 43 it should be noted that there is a small difference in times. The difference between the two will range anywhere from one to four minutes. These dates may be different due to the fact that Marco Concha calculated the times using worse case scenarios which would limit the accuracy in his model.

Zone of Interference

When calculating the Zone of Interference the model accounts for the ground station elevation, the satellite antenna gain, and satellite attitude. The elevation is taken from the ground station look angles that are calculated in the S-Function. The satellite antenna gain is calculated using the radiative pattern angle and look-up table. The attitude is a text file input to the MATLAB workspace. When the ground station elevation is greater than 10 degrees and less than 170 degrees the model will inspect the receiver gain. If the gain is less than or equal to -10 dB then the communication link is in the Zone of Interference. Figure 44 is a plot of the ground station line of sight and the Zone of Interference line of sight.

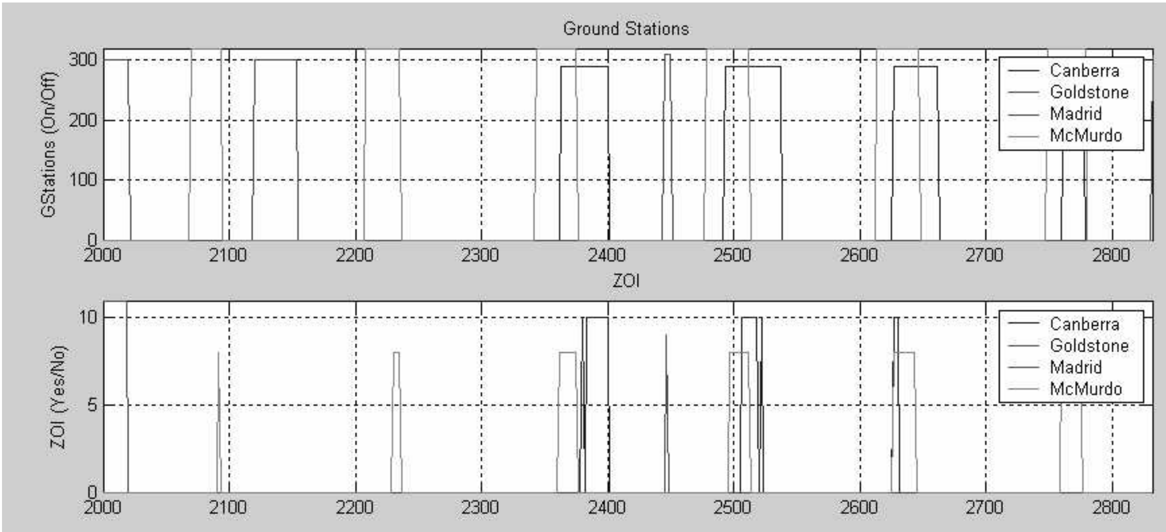

Figure 44: Ground Station Line of Sight and Zone of Interference Line of Sight

Similar to the ground station line of sight graph the Zone of Interference graph shows logic high for when transmission occurs in the Zone of Interference and logic low for not in the Zone of Interference. The Zone of Interference always resides inside ground station line of sight times.

In order to view the start and end dates of the Zone of Interference an excel spreadsheet was created. The start and end dates are written to a workspace and then a MATLAB script file writes the data to an excel file. Figure 45 shows the excel spreadsheet of start and end dates for the Zone of Interference as well as the ground station visibility dates.

	A	B	C	D	E	F	G	H	I	J	K	L	M
1	McMurdo Ground Station (ZOI)												
2	Start Time							End Time					
3													
4	Year	Month	Day	Hour	Minute	Second		Year	Month	Day	Hour	Minute	Second
5													
6	2006	3	1	15	17	40		2006	3	1	15	19	20
7	2006	3	1	15	21	0		2006	3	1	15	24	20
8	2006	3	1	15	27	40		2006	3	1	15	29	20
9	2006	3	1	17	34	20		2006	3	1	17	39	20
10	2006	3	1	17	41	0		2006	3	1	17	42	40
11	2006	3	1	19	54	20		2006	3	1	19	56	0
12	2006	3	2	2	52	40		2006	3	2	2	54	20

	A	B	C	D	E	F	G	H	I	J	K	L	M
1	McMurdo Ground Station												
2	Start Time							End Time					
3													
4	Year	Month	Day	Hour	Minute	Second		Year	Month	Day	Hour	Minute	Second
5													
6	2006	3	1	15	1	0		2006	3	1	15	29	20
7	2006	3	1	17	17	40		2006	3	1	17	42	40
8	2006	3	1	19	34	20		2006	3	1	19	57	40
9	2006	3	1	21	52	40		2006	3	1	22	14	20
10	2006	3	2	0	11	0		2006	3	2	0	34	20
11	2006	3	2	2	29	20		2006	3	2	2	56	0
12	2006	3	2	4	46	0		2006	3	2	5	17	40
13	2006	3	2	7	1	0		2006	3	2	7	36	0

Figure 45: Simulink ST5 ZOI Dates

To create this the Zone of Interference spreadsheet the user will type McMurdoZOI.m in the MATLAB command prompt. Other ground station Zone of Interference dates are accessed by typing these commands:

- CanberraZOI.m
- GoldtsoneZOI.m
- MadridZOI.m

Link Margin and Doppler Shift Results

Figure 46 shows the plots of ground station line of sight, Doppler shift, and link margin. For the times of visibility the link margin and Doppler shift can be determined. The ground stations will need to adjust their transmission frequencies to account for the effects of Doppler shift. The link margin must be at least 3 dB for a successful transmission. From Figure 46 we can see that the link margin of ST5 is always above 3 dB. The McMurdo ground stations link margin is much lower than the DSN stations but never goes below 8 dB. The lower link margin from McMurdo is caused by a weaker transmission power of the GN antenna.

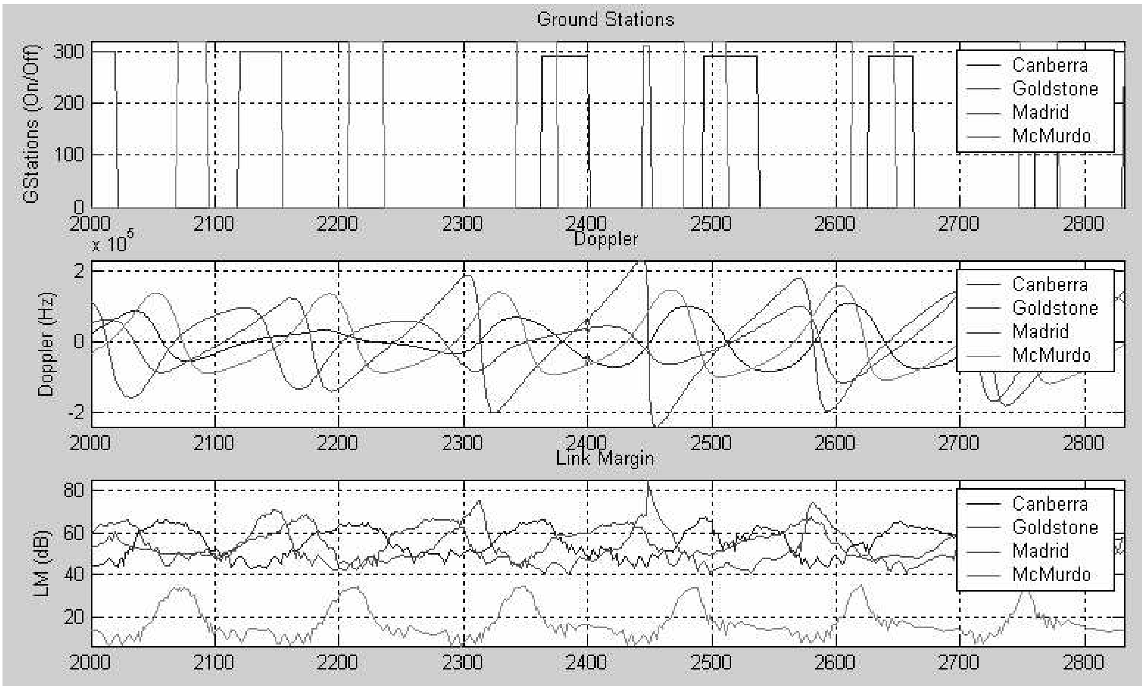

Figure 46: Ground Station Line of Sight, Doppler Shift, and Link Margin

5.3 Orbit Propagator

The orbit propagator model generates the satellite position and velocity state vectors that describe the satellites orbit. The data is stored in arrays (FinalSatelliteECI, FinalSatellitevECI) that the Communications model can use to asses the communication link.

The orbit propagator reads in IIRV's as initial state inputs to the model. The model then propagates until the time of the next IIRV. This method provides hourly corrections to the position and velocity vectors of the satellite, providing a more accurate orbit. The IIRV's are separated by 4 hours; therefore the first IIRV should be used 4 hours into the simulation. Figure 47 is a graph of the x component of satellite position. This shows the first IIRV making the first correction to the propagator. The blue circle is the point where a new position and velocity vector is inputted to the orbit propagator. This point occurs at 14400 seconds which is equivalent to 4 hours.

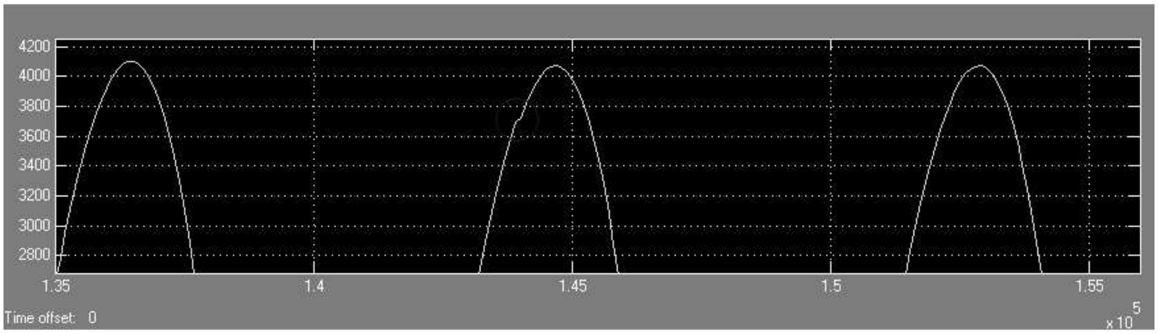

Figure 47: IIRV correction

Once the orbit propagator has been run the orbit can be plotted. To plot the orbit PlotOrbit.m is executed using the positions vector (FinalSatelliteECI) and velocity vector (FinalSatellitevECI). Figure 48 shows the satellite orbit.

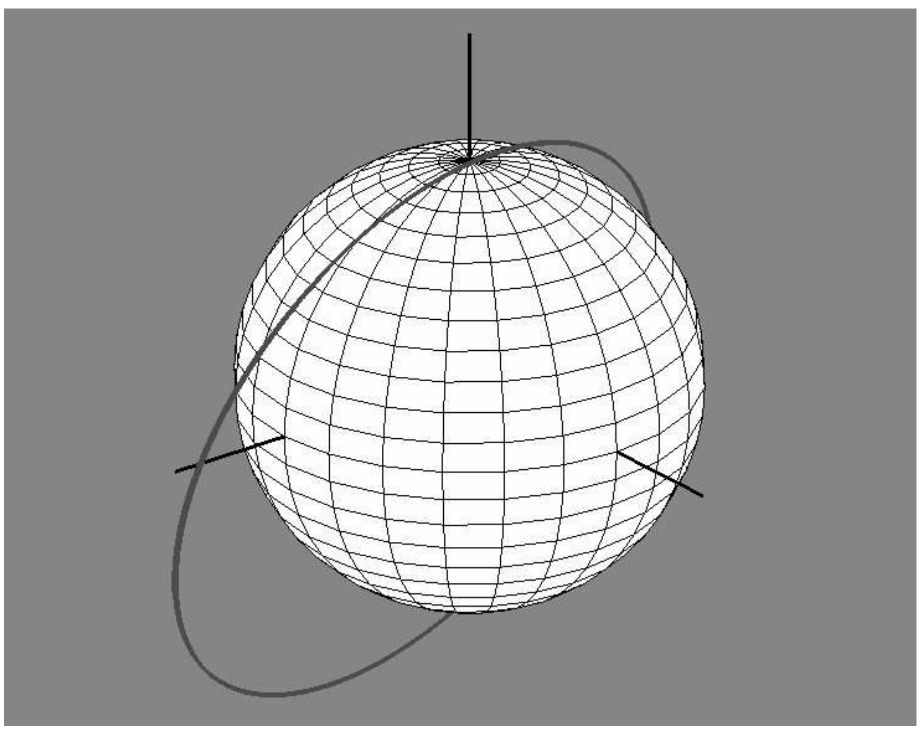

Figure 48: ST5 Orbit

5.4 Summary

The SimulinkST5 model generates several outputs which are generated from the two major subsystems: the orbit propagator and the Communications model. The model outputs are written to the workspace and can be viewed using MATLAB script files. Outputs generated by the model are presented as readable and serviceable documents.

The orbit propagator produces a plot of the IIRV correcting the orbital path of the satellite as well as a three-dimensional plot of the ST5 satellite orbit. The Communications model provides graphs of the ground station line of sight, Zone of Interference line of sight, link margin and Doppler shift. The Communications model also provides excel spreadsheets giving the start and end dates of the line of sight for each ground station as well as the start and end dates of the Zone of Interference occurrences.

6. Conclusions

6.1 Introduction

The Simulink modeling software was used to create a new communication model to assess the ST5 communications link. The ST5 Communications model is a powerful and user-friendly model that provides accurate line of sight logic and Zone of Interference logic. The model provides an output graph and excel file that shows the black out times due to line of sight and ZOI in an easy to view format.

6.2 Model Capabilities

After reviewing the results of the communications model, we can now state that the model fulfills the goal of determining ST5 communication capabilities. The orbit propagator is now able to accurately propagate the ST5 orbit for a 3-4 week period and the communications model is able to accurately determine the link margin due to the new ST5 antenna radiative pattern which will allow us to predict the zone of interference.

The orbit propagator will now provide orbital accuracy of up to 3-4 weeks. The propagator model uses IIRV's which will be received every 4 hours providing new position and velocity vectors of the satellite. The model can then use the IIRV inputs to properly adjust the satellites orbital position and velocity, correcting any error created by the orbit propagator.

Using the IIRV position and velocity vectors the communications model now has the necessary inputs to accurately determine the satellite communication capabilities. The ST5 antenna radiative pattern was accurately modified to improve link margin calculations for the ground station to satellite line of sight logic. The modeled antenna pattern was then used to determine the occurrences of the zone of interference as a function of spacecraft orbital geometry and satellite attitude. The model will output excel spreadsheets that provide start and end dates for the line of sight logic and ZOI occurrences of each ground station.

6.3 Suggested Model Modifications

There are some suggestions for the modification of the model that will help the functionality, which would make it a more valuable tool for the mission.

At this time, our model only provides offline mission planning used for long-term satellite health assesment. The model can be easily changed to incorporate the real-time state health assessment. The orbit propagator model will take the input IIRV's in real time and then propagate for a 4 hour period. The communications can then be run with the new propagation and use real-time satellite attitude. When the next IIRV is ready the orbit propagator and Communications model can be run again in the same manner. This can provide ST5 mission planning with knowledge of the satellite communications capabilities every 4 hours during the mission.

Another modification would be to use a more accurate propagator algorithm to provide accurate propagation if the IIRV's should come to an end. Currently, the orbit propagator takes in the IIRV's and adjusts the path of the orbit every four hours but if the IIRV inputs should run out the orbit propagator would only be as accurate as the current two body model. The two body model is accurate enough for our purposes because it will be reset every 4 hours due to the IIRV’s.

6.4 Summary

The ST5 Communication model is a very powerful tool, which will aid in the ST5 mission planning. The model is very flexible and is easily updated by the user. The model shows the successful implementation of SimulinkST5 provides an accurate model–

based mission operations tool. The communication model will accurately model ST5 communications link for successful mission planning.

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Appendix 2: Contacts at NASA/GSFC

Robert Shendock – ST5 Flight Operations System Engineer
Shendock@pop700@gsfc.nasa.gov
(301) 286-9398
Bob was the project mentor. Bob met with us everyday to check on progress and answer any questions. Bob set up meetings and conferences with other NASA Goddard engineers to further aid our progress.

Federico Sanidad – ST5 Flight Operations System Engineer
Federico.C.Sanidad.1@gsfc.nasa.gov
(301) 286-9507
Rich acted as the assistant mentor to our project. Rich went over our project description and answered any basic questions we had. He also filled in for Bob when he was unable to attend meetings and presentations.

Glenn Bock – ST5 Flight Operations Ground System Engineer
gbock@pop400.gsfc.nasa.gov
(301) 286-8706
Glenn helped us solve the radiative angle pattern problem. He also supplied us with documents that explained the basics of spaceflight to help us further understand aerospace engineering.

Jim Morrissey – ST5 Attitude Control System Analyst
jmorriss@pop500.gsfc.nasa.gov
(301) 286-0529
Jim is the original creator of the orbit propagator. Jim met with us to help solve the new orbit propagator problem. Jim was supposed to be our project mentor but was transferred to a different mission before we arrived.

Seth Shulman – E-01 Flight Operations Technical Lead
Seth.Shulman@gsfc.nasa.gov
(301) 286-3437
Seth is a MATLAB expert. He provided us with MATLAB functions and gave us some guidance when writing MATLAB functions. Seth also helped solve the IIRV portion of the orbit propagator.

Marco Concha – ST5 Guidance and Navigation Control
Marco.Concha@nasa.gov
(301) 286-6038
Marco supplied us with sample IIRV's to test in our model. He also met with us to solve IIRV portion of our orbit propagator. Marco also supplied us with the necessary MATLAB files to do some coordinate conversions.

Victor Sank – ST5 RF/Communications

vsank@pop500.gsfc.nasa.gov
(301) 286-2645
Victor provided us with the RFICD. The RFICD verified our Link Margin calculations and also provided us with ground station parameters and antenna parameters. We spoke with Victor on the phone to verify our results with the RFICD.

Kevin Blahut - ST5 Mission Operations Engineer
kevin.blahut@gsfc.nasa.gov
(301) 286-5761
Kevin has acted as an assistant project mentor in the past. Kevin gave us a tour of the ST5 building facilities. We were able to see the actual ST5 satellite under construction.

Kenneth J. Witt – ISR; Senior Member Research Staff, Information Systems Branch Supervisor
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(304)368-9300
Kenneth is a member of the ISR team that provided the GMSEC bus that will be interfaced with our model.

Jason W. Stanley – ISR; Member Research Staff, Software Systems Branch
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Jason is a member of the ISR team that provided the GMSEC bus that will be interfaced with our model.

Appendix 3: List of Terms

Apogee: The point where the moon or satellite is at its farthest point from the earth.

Argument of perigee: The angle between the ascending node and the orbit's point of closest approach to the earth (perigee).

Attitude: The orientation of a spacecraft relative to its direction of motion.

Azimuth: Horizontal angular distance from a reference direction, usually the northern point of the horizon, to the point where a vertical circle through a celestial body intersects the horizon, usually measured clockwise period.

Doppler Shift: An offset in frequency as perceived by a receiver, from the nominally transmitted frequency caused by relative motion of the transmitter and receiver.

Eccentricity: A constant defining the shape of the orbit (0=circular, Less than 1=elliptical).

ECF (Earth Centered Fixed): A coordinate system with its origin at the center of the Earth and axes which are fixed in the central body.

ECI (Earth Centered Inertial): A coordinate system with its origin at the center of the Earth and axes which are fixed in inertial space.

Elevation: The height to which something is elevated above a point of reference such as the ground.

Greenwich Sidereal Time: Angle between the Prime Meridian and the Vernal Equinox, at a given time. Measure of time as a star, or sidereal, time rather than as a solar time.

Inclination: The angle between the equator and the orbit plane.

Julian Date: Single number representation of the year, month, day, and time information.

Link Margin: The difference between the required signal to noise ratio and the actual signal to noise ratio.

Local Sidereal Time: Angle between the local Longitude and the Vernal Equinox, at a given time. Can be calculated as the Greenwich Sidereal Time added to the local East Longitude, in radians or degrees.

Longitude of ascending node or Right Ascension of the Ascending Node: the angle between vernal equinox and the point where the orbit crosses the equatorial plane (going north).

Perigee: The point where the moon or satellite is at its closest point to the earth.

Signal to Noise Ratio: average signal power and average noise power.

Telemetry: The relaying of information from the scientific instruments aboard the satellite to the ground station.

The Line of Nodes: The point where the satellites cross the equator.

Topocentric: A coordinate system originating at a point on the Earth. The axes are defined so that the x is in the local north location, y is in the local east direction, and z is along the inward normal to the surface.

True anomaly: The angle between perigee and the vehicle (in the orbit plane).

Semi-major axis: one-half the maximum diameter, or the distance from the center of the ellipse to one of the far ends.

Signal-to-noise power ratio: The ratio of the signal power over the noise power.

Vernal Equinox: The moment at which the sun passes through the point at which the ecliptic intersects the celestial equator, about March 21, marking the beginning of the spring in the northern hemisphere.

Appendix 4: List of Acronyms

ΔCDR	Delta Critical Design Review
ADS	Attitude Determination System
ASCII	American Standard Code for Information Interchange
C&DH	Command and Data Handling
CCSDS	Consultative Committee for Space Data Systems
CGS	Combined Ground System
DS1	Deep Space 1
DS2	Deep Space 2
DSN	Deep Space Network
DSP	Digital Signal Processing
EA	Evolved Antenna
ECF	Earth Centered Fixed
ECI	Earth Centered Inertial
EIRP	Effective Isotropic Radiated Power
EO1	Earth Observing 1
EO3	Earth Observing 3
FDF	Flight Dynamics Facility
FOT	Flight Operations Team
FPGA	Field Programmable Gate Array
GEO	Geosynchronous Earth Orbit
GHz	Giga Hertz
GMSEC	Goddard Space Flight Center Mission Services Evolution Center
GN	Ground Network
GPS	Global Positioning System
GSFC	Goddard Space Flight Center
GST	Greenwich Sidereal Time
GTO	GEO Transfer Orbit
HPA	High Power Amplifier
ISR	Institute for Scientific Research
JD	Julian Date
Kbps	Kilo Bytes per Second
LEO	Low Earth Orbit
LLA	Longitude/Latitude/Altitude
LM	Link Margin
MEMS	Microelectromechanical Systems
LNA	Low Noise Amplifier
MOC	Mission Operations Control Center
MQP	Major Qualifying Project
MSSS	Miniature Spinning Sun Sensor
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
NMP	New Millennium Program
RF	Radio Frequencies
RFICD	Radio Frequency Interface Control Document

RPA	Radiative Pattern Angle
SatDec	Satellite Declination
SatRA	Satellite Right Ascension
SEZ	Topocentric Horizon Coordinate System
SNR	Signal to Noise Ratio
SSR	Solid State Recorder
ST5	Space Technology 5
TDRSS	Tracking and Data Relay Satellite System
TEC	Topocentric Equatorial Coordinate System
UT	Universal Time
VEC	Variable Emittance Coating
WPI	Worcester Polytechnic Institute
WSC	White Sands Complex
WWW	World Wide Web
ZOI	Zone of Interference

Appendix 5: ST5 Model Users' Manual

ST5 Communications Model

This section is a manual providing directions on using the ST5 Communications model. It describes the steps to run the Communications model with the necessary inputs as well as how to edit input parameters. It also provides the methods for viewing the outputs of the model.

Verifying Existence of Necessary Files

Before using the Communications model it is important verify that the current work directory contains all necessary files. This can be done by typing "ls" into the MATLAB command prompt.

>> ls

The following is a list of the files necessary for simulation:

OrbitPlot.m PlotLMDS.m PlotZOI.m GStationsPlot.m	Plotting script files
Init.m St_5mod.m IIRV.m	Files used to initialize constants
ST5_STK_WP.txt Look_up_EA.xls UDAP.txt Attitude.txt	Input Files to MATLAB
Communications2004.mdl OrbitProp.mdl	Models
CalenderDate.m DOY.m ecf2eci.m F_T.m GAST.m NUTATION.m MODULO.m juliandate.m Topocentric.m	MATLAB Functions used by models
Canberra.m Madrid.m Goldstone.m McMurdo.m CanberraZOI.m MadridZOI.m GoldstoneZOI.m McMurodZOI.m	MATLAB Output Functions
SFunGStations.m	S-functions used by models

If all the files are listed, we then can begin setting up the workspace for the Communications model. The commands that are necessary to set up the workspace have been stored in Init.m. To run the init file, type “Init” in the MATLAB command prompt.

>>Init

Note: This command may take several seconds to execute.

Before:

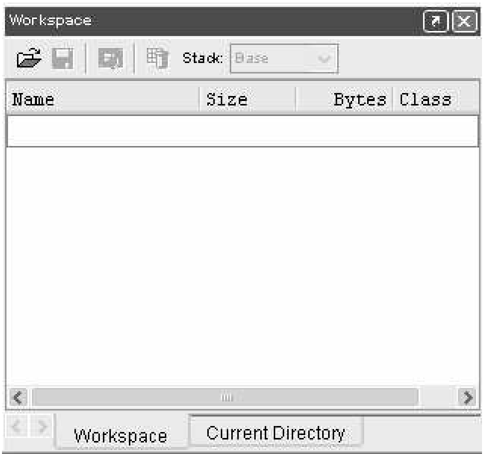


Figure 1 – Workspace before Init.m is run

After:

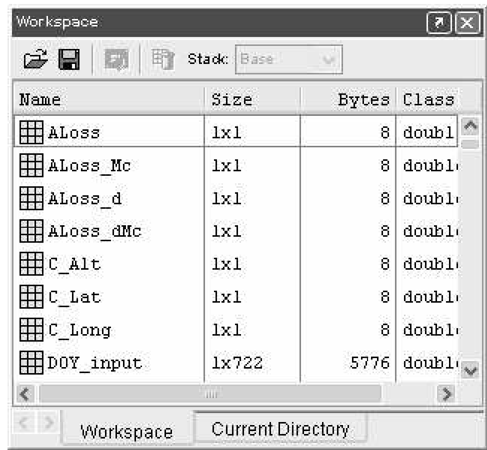


Figure 2 – Workspace after Init.m is run

The file contains ground station link margin parameters, simulations step size, parameters for OrbitPlot.m, and IIRV.m. If input parameters need to be edited, just enter the following command in the command prompt.

>>edit Init

ST5 Orbit Propagator

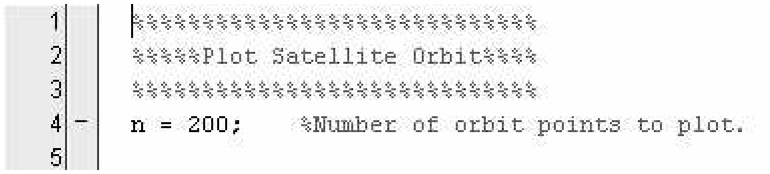
Before running the Communications model the orbit propagator model (OrbitProp.m) must be run. This model will determine the ECI coordinate of the satellite as a function of time. The orbit propagator is run within Init.m, which contains the IIRV.m script file.

This script file reads in the IIRV files stored in the current directory (ST5_STK_WP.txt). The orbit propagator is then run in intervals between the IIRV’s.

To view the orbit propagator output run OrbitPlot.m, by typing “OrbitPlot” in the MATLAB command prompt.

>>OrbitPlot

This script file will output the last 200 points of the Satellite orbit. To edit the number of points, type “edit OrbitPlot” in the command prompt. On the 4^{th} line of the script file the variable n can be changed to edit the number of points plotted.



```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%%%%Plot Satellite Orbit%%%%
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
n = 200;    %Number of orbit points to plot.
```

Communications

Once Init.m has been run the Communications model is ready to be executed. This model will determine the ground stations ability to communicate with the satellite. It does this by calculating the satellite's line of sight, link margin and Doppler.

To run this model, type the name of the model, "Communications2004", into the MATLAB command prompt.

>> Communications2004

When the model opens, press the Simulink play button to run the simulation.

Figure 3 – Locating the start button

The status bar slowly fills as the simulation is run. When the simulation is complete, you can close the model and move on to the next step.

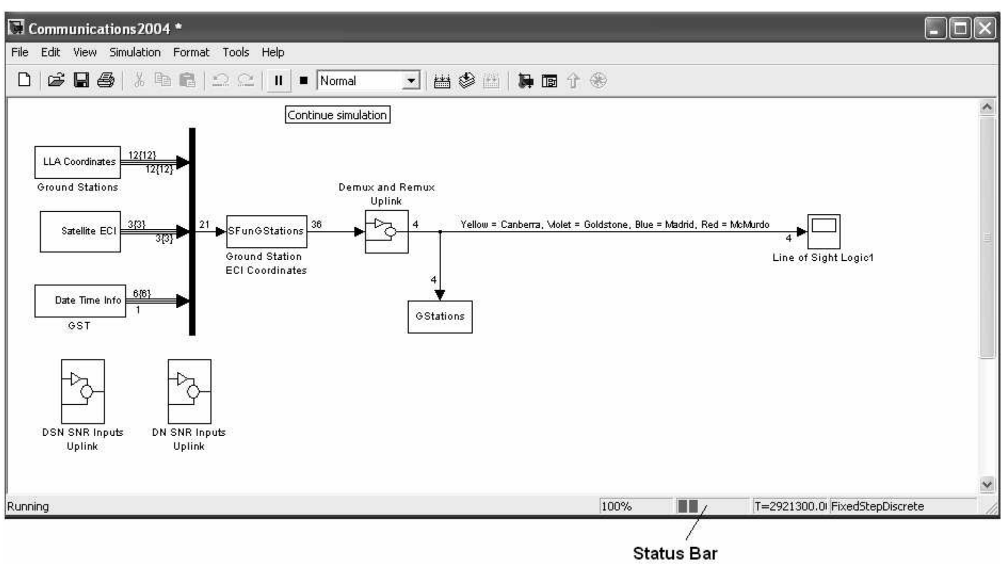


Figure 4 – Waiting for simulation to finish

Plotting Model Outputs

Once the Communications model has finished running the data of interest can be plotted. The model stores link margin, Doppler shift, line of sight, and ZOI, in the workspace as arrays structured with time. The data can then be accessed by the plotting scripts: PlotLMDS.m, PlotZOI.m, and GStationsPlot.m.

Each script plots the following:

PlotLMDS.m	Plots line of sight, link margin, and Doppler shift vs time.
PlotZOI.m	Plots line of sight, and Zone of Interference logic
GStationsPlot.m	Plots line of sight

To view a plot, simply type the name of the corresponding script in the MATLAB command prompt.

Output Files

The model can also produce Calendar Dates and times that correspond to the line of sight, and Zone of Interference logic. Each ground station has its own script file that will save data to an excel spreadsheet for viewing. There is a script file for both line of sight and Zone of Interference times; Canberra.m, CanberraZOI.m, Goldstone.m, GoldstoneZOI.m, Madrid.m, MadridZOI.m, McMurdo.m, McMurdoZOI.m. To view the data simply type the name of the ground station script file in the MATLAB command prompt.

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