NavGold – An On-Orbit Test Bed for Experiments in Formation Flight

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Abstract. An initiative has been undertaken at the University of Colorado at Colorado Springs (CU-Colorado Springs) involving the design, development, launch, and operation of two nanosatellites. Objectives of the project, NavGold, include establishment of an on-orbit test bed for research, testing, and validation of critical nanosat technologies enabling formation flying and constellations. Additional educational objectives provide team members leadership opportunities in a multidisciplinary team with real hardware, budget, and scheduling challenges. Two identical nanosats will be built and flown. The spacecraft design includes a maneuvering propulsion system and three-axis active attitude determination and control. Absolute and precision relative navigation will be accomplished using augmented Global Positioning System information. Maximum use of commercial off-the-shelf hardware and software will be employed, to minimize cost and schedule risk. The aggressive schedule calls for spacecraft delivery in early 2001. The current baseline orbit assumes a shuttle launch, implying a 50+ degree inclined low earth orbit. Spacecraft power and communication designs draw heavily from CU-Colorado Springs experience on the Falcon Gold spacecraft launched in October, 1997, a joint project with the U. S. Air Force Academy. Additionally, the CU-Colorado Springs ground station, communications system, part of the power sub-function, the flight computer, and the in-house developed tri-axial magnetometer and digital sun sensor have been successfully tested on high-altitude balloon flights.

Introduction

The concept of a spacecraft constellation has become very popular recently, where a constellation is defined a group of two or more spacecraft interacting in some manner to meet common mission objectives. Many constellation mission concepts involve the use of small, micro, and nanosatellites. Constellations are viewed as an attractive means of achieving greater mission performance, with increased reliability, and perhaps less cost, when compared to other mission-design approaches. Studies have indicated great promise for this approach.

But, to achieve these mission-performance objectives, several technical challenges must be overcome. Significant technical advancements in command and control techniques, space navigation, spacecraft autonomy and maneuverability, and communications architectures are required. For example, in dense constellations (e.g., inter-spacecraft ranges of < 1 km), relative spacecraft position must be known far more accurately than may be determined through currently available ground-based techniques. Other issues include communication, command and control, and guidance and navigation techniques for formation flight, which involve autonomous relative repositioning of spacecraft in a constellation. Therefore, engineering research must be focused in several critical areas to achieve the goals of high-performance constellations.

A research team at the University of Colorado at Colorado Springs (CU-Colorado Springs), in collaboration with the Air Force Research Laboratory – Phillips and several industrial consultants, has undertaken the task of addressing some of these research issues through the establishment of an on-orbit test bed. The test bed will use two or more nanosatellites with sufficient capability to allow a variety of experiments, focusing on several of the critical enabling technologies noted above. A late 2001 launch into low earth orbit is planned. These experiments may be exploratory in nature or on-orbit validation of new concepts and/or advanced hardware. This paper presents a description of the test-bed nanosats, along with the ground station and some of the on-orbit experiments planned. Finally, it should be noted that other organizations interested in exploring the conduct of experiments on this test bed are encouraged to make their interest known to the CU-Colorado Springs research team in a timely fashion, to determine if their interests may be accommodated in the design.

Program Approach

The project is being coordinated through the Flight Dynamics and Control Laboratory, in the College of Engineering and Applied Science at CU-Colorado Springs. The program objectives include both research and education. The research and design team consists of faculty, current and former students, plus a select group of industrial advisors. The Air Force is helping to provide the launch. Additional funding beyond the current university cost sharing and industrial contributions is still being sought.

The team is relatively small (<25) and interdisciplinary. Rather than having a rigid organizational structure, the team is quite fluid, with significant cross-disciplinary interaction and activity. The fact that the project is focusing on nanosats, rather than larger and more complex spacecraft, allows for team members to keep the "big picture" more clearly, while still being able to focus on their individual tasks. Much of the project activity is conducted via theses, and special-project and design courses.

One challenge common for a university team is the rotation of students through the project, and subsequent loss of the group's "knowledge base". Hence, frequent full-team meetings, workshops, and documentation are critical.

The conceptual approach adopted for spacecraft bus design is to view the spacecraft as a whole, rather than a collection of subsystems. Accordingly, the team is not strictly organized according to subsystems. The focus is on functions rather than subsystems, as discussed in [4], for example. One advantage of this approach is that integrating more than one function into a hardware component is facilitated.

Given that low-cost is a major design constraint, maximum utilization of commercial off-the-shelf (COTS) components will be made, plus many components (hardware and software) are being designed and developed in house. A common approach is to have a student designer be responsible for a component from design through fabrication and test, whenever possible.

The mission objectives are the establishment and operation of an on-orbit test bed for performing experimental engineering research in several of the key fundamental technologies that enable constellations and formation flying. These key technologies include, but may not be limited to, precision relative navigation, relative-positioning and station-keeping guidance, relative attitude determination, and multi-spacecraft command and control structures. Given the engineering-research mission, rather than a specific space-science mission, the spacecraft design requirements include "flexibility," rather than rigid, pre-defined, space-science derived requirements. This flexibility is defined here in terms of experiment flexibility after the test bed is operational. That is, we expect to modify techniques, algorithms, etc. on orbit, based on results from experiments.

Mission Objectives and Constraints

Primary Mission Objective

To establish an on-orbit test bed for development and validation of autonomous relative-navigation techniques and guidance algorithms aimed at meeting the critical relative-navigation, maneuvering, and station-keeping requirements of autonomous constellation operations.

Secondary Mission Objectives-

- To demonstrate autonomous relative-navigation accuracies of less than 100m, with a goal of 10 m accuracy or better.
- To demonstrate autonomous relative-position maneuvering and station-keeping capability with relative position accuracy equal or better than 100m.
- To demonstrate autonomous relative-attitude maneuvering and station-keeping capability with relative attitude accuracy equal or better than two degrees.
- To accomplish all primary and secondary objectives at the absolute minimum cost.
- To document critical technological accomplishments in developing autonomous navigation, maneuvering and station-keeping capabilities.
- To educate the student team members in all facets of spacecraft mission development, from initial concept to final launch and on-orbit operations.

These objectives were derived based on the following considerations, with regard to constellations and formation flight.

A spacecraft constellation can be described in terms of at least four constellation descriptors. These descriptors are 1) the degree of autonomy, 2) the geometric configuration, 3) the communications architecture, and 4) the control architecture of the constellation. Regarding the degree of spacecraft autonomy in the constellation, for example, each spacecraft could be completely controlled from the ground, in contrast to total spacecraft autonomy where the constellation spacecraft operate completely independent from ground control. The optimal level of autonomy is mission dependent, but typically will fall between the two extremes cited.

The geometric configuration of the constellation describes the spacecraft's relative locations and relative orientations (or attitude). Further, the geometric configuration can be fixed or variable. For example, if gross on-orbit spatial re-positioning and/or reorientation is required by the mission, the geometric configuration is dynamic. But maneuvering capability of some sort is almost always required, even if the geometric configuration is static. Due to environmental perturbations on the member spacecraft, relative-position station keeping and relative orientation station keeping must be performed. This requires that the relative position of the spacecraft within the constellation as well as the relative orientation of the spacecraft within the constellation be determined, with some degree of autonomy. Incidentally, this defines the relative navigation and relative attitude determination problems, analogous to navigation and attitude determination of a single spacecraft.

The communication architecture of the constellation describes how the spacecraft communicate with the ground and with each other, what information is transferred or crosslinked between the spacecraft, and how often. Some form of communication with the ground will be required for data retrieval, health monitoring, constellation commanding, etc. One potential communication configuration might be that each spacecraft communicates with every other spacecraft, as well as the ground. An alternative communication configuration would involve each spacecraft reporting to a single primary spacecraft that is responsible for all intra-constellation as well as ground communications.

Finally, a constellation can be described by its control architecture that describes how the constellation maintains and possibly changes its geometric configuration and/or communication architecture. One example of control architecture might include a single leader spacecraft, having knowledge of the absolute positions and attitudes of all other spacecraft. This leader might then perform all relative-position and attitude determination, and command any necessary maneuvers by the other spacecraft. This control architecture is often referred to as a "leader-follower" architecture. Another control architecture is a "cooperative" control architecture, in which all spacecraft interact cooperatively in some fashion to maintain the geometric configuration and/or communication architecture. As with the level of autonomy, the optimal control architecture will be mission dependent.

Many open technical issues exist related to the optimal constellation descriptors defined above, as well as the best system design approach leading to a selected constellation level of autonomy, geometric configuration, and communications and control architectures. However, three key areas appear critically in need of fundamental and generic research, and are the primary focus of our efforts. These are the relative navigation, relative position/attitude maneuvering and relative position/attitude station keeping problems. Within these problems, questions arise as to the optimum measurement set and coordinates for relative navigation, as well as the optimal estimation algorithms. These questions apply to both the relative position and the relative attitude determination problems.

With regard to navigation accuracy requirements, [2] and [3] define a close-formation constellation, or geometric configuration, as "…two spacecraft kept to a close separation ranging from tens of meters to less than one km…". If separation distances of this magnitude are required, certainly ground-based orbit determination methods are not appropriate, since these methods can at best deliver position-determination accuracy to about several kilometers [1]. Even space-based determination techniques such as TDRSS can only deliver an accuracy of hundreds of meters in LEO. Hence GPS, enhanced in some fashion perhaps with additional ranging measurements will be necessary. Furthermore, for autonomous constellation operations, autonomous relative navigation with accuracy on the order of tens of meters will be required. Such generic constellation mission requirements lead to our NavGold mission objectives.

Mission Requirements and Constraints

In order to meet these mission objectives, a minimum of two spacecraft will be required on-orbit, and at least one spacecraft must have autonomous navigation and maneuvering capability **-** both position and attitude.

Programmatic constraints have lead to the shuttle being selected as the baseline launch vehicle. As a result, the mission orbit will be constrained to be circular, with a period of approximately 92 min, an altitude of about 400km, and a nominal inclination of 51 deg. With this orbit, ground communications will be limited by an average pass duration of about 5 min, with an average of 5 passes per day, assuming a single ground station located on the CU-Colorado Springs campus.

A nominal launch date has been identified as mid 2001. Near this time, the earth is expected to be experiencing solar-maximum conditions. This fact, coupled with the 400km initial orbit, leads to a predicted mission life of approximately 3 - 5 months before rapid orbit decay. Finally, the nominal launch date coupled with the nominal orbital parameters limits average sunlight exposure time to about 1 hr/orbit (one orbit is approximately 92 min).

Spacecraft Design Requirements and Constraints
Autonomous navigation capability will require one or more GPS receivers and on-board processing. Maneuvering and station keeping will require maneuvering-propulsion capability plus three-axis attitude determination and control. This in turn will require attitude sensors and torque actuation. The launch configuration imposes some design constraints. For example, the total mass per spacecraft is limited to approximately 14 kg, and the total volume per spacecraft is limited to approximately 3900 in^3. Requirements on the remaining functions, such as power generation, environment control, data processing, and structure, will naturally flow from the above requirements and constraints.

Prototype Maneuver Experiments

For discussion purposes here, all maneuvers will be performed "in-plane," although out-of-plane maneuvers are under study. Also, all maneuvers are planned to be performed autonomously. The class of maneuvers to be described are Relative Re-Positioning, Relative Position Station Keeping, Relative Re-Orientation, and Relative Orientation Station Keeping.

Relative Re-Positioning
Two types of relative re-positioning maneuvers will be described - type one is a relative phasing maneuver, and type two is a relative altitude maneuver. It is assumed that any in-plane re-positioning maneuver can be derived from a combination of these two types.

Type 1: Increase/decrease the relative separation or phase angle d between the two spacecraft in the same orbit, as denoted in Figure 1.

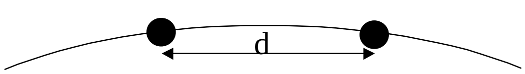

Figure 1. Spacecraft Relative Phasing Distance d.

An example of a Type 1 re-positioning maneuver involves a Hohmann type transfer as depicted in Figure 2. The initial relative position of the two spacecraft, S_1 and S_2, is indicated by the filled circles. At time T_0, S_1 performs a ΔV maneuver to decrease S_1's velocity and put it into the phasing orbit E. At time T_1=To+T_E, S_1 performs a second ΔV maneuver to increase S_1's velocity and return to orbit C. At time T_1, S_2 is now "closer" (indicated by the empty circle) to S_1 by:

$$\delta t = T_E - T_C.$$

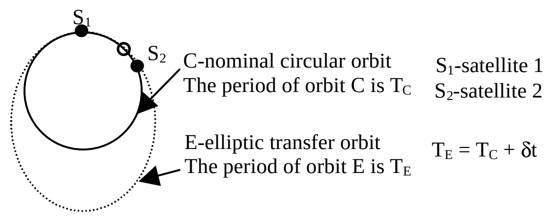

Figure 2. Hohmann Orbit Transfer for Type 1 Re-Positioning Maneuver.

The ΔV required to adjust the phasing is shown parametrically in Figure 3. The y-axis shows the total amount of ΔV required for the maneuver in meters per second, for a given phasing adjustment, plotted in degrees. Each line represents different amount of time allotted to perform the maneuver, in terms of orbits.

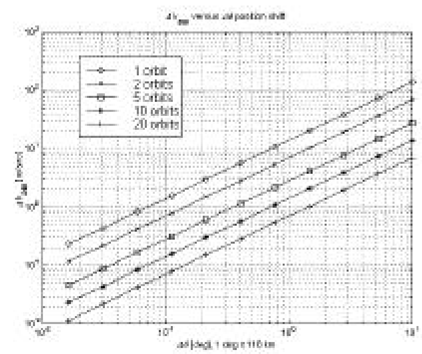

Figure 3. ΔV Trade Study.

Type 2: Increase/decrease the relative altitude separation h between the two spacecraft, as denoted in Figure 4.

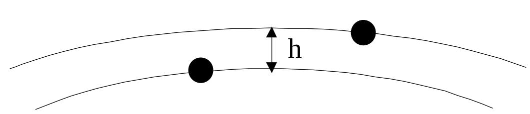

Figure 4. Spacecraft Relative Altitude Separation Distance h.

An example of a Type 2 re-positioning maneuver also involves a Hohmann type transfer as depicted in Figure

5. The initial relative position of the two spacecraft, S_1 and S_2, is indicated by the filled circles. At time T_0, S_1 performs a ΔV maneuver to increase S_1's velocity and place it into the transfer orbit E. At time $T_1=T_0+T_E/2$, S_1 performs a second ΔV maneuver to increase S_1's velocity and place it into orbit C_2. At time T_1, S_1 has an altitude relative to S_2 of h (indicated by the empty circles).

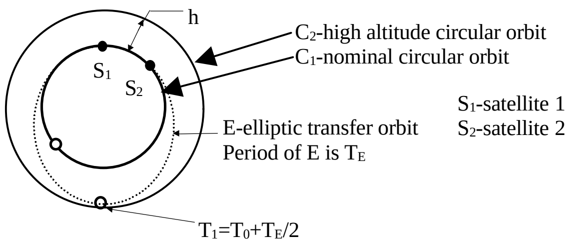

Figure 5. Hohmann Orbit Transfer for Type 2 Re-Positioning Maneuver.

Relative Position Station Keeping

Relative position station keeping, for the purpose of the NavGold mission, will be defined as follows. Given a commanded d and h, d_{com} and h_{com}; a specified span of time, $[t_1, t_2]$; and relative position tolerances, δd and δh, it is required that: $| d(t) - d_{com} | < \delta d$ and $| h(t) - h_{com} | < \delta h$ for all $t \in [t_1, t_2]$. This performance objective will be accomplished through small (pulsed) ΔV maneuvers.

An example of relative-position station keeping involves the hypothetical requirement that a relative separation, d, and altitude, h, is to be maintained within some tolerance, over the time period defined by sequential passes over the ground station. The Hohmann transfer is again one method by which this requirement could be met, as indicated in Figure 6.

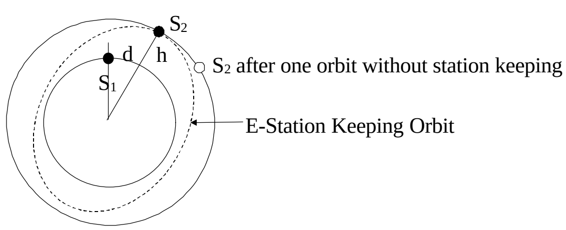

Figure 6. Hohmann Orbit Transfer for Relative Position Station Keeping Maneuver.

Relative Re-Orientation

In Figure 7, θ_1 and θ_2 represent the inertial orientation of spacecraft S_1 and S_2, respectively. The relative orientation is denoted as $\Delta\theta$, Figure 7. An example re-orientation maneuver is depicted in Figure 8 below.

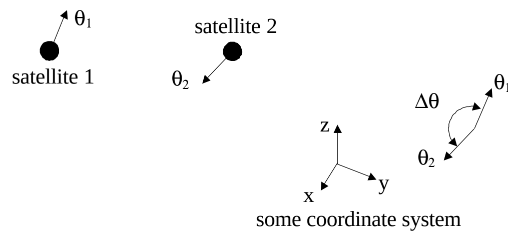

Figure 7. Geometry of Relative Orientation

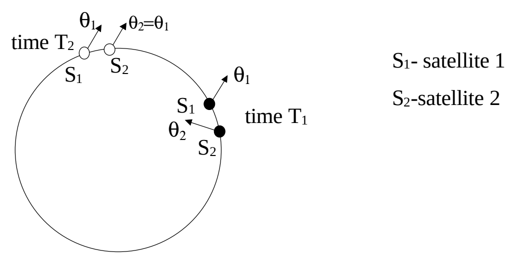

Figure 8 - Example Relative Reorientation Maneuver.

Relative Orientation Station Keeping

Relative orientation station keeping, for the purpose of the NavGold mission, will be defined as follows. Given a commanded relative orientation, $\Delta\theta_{com}$ and a relative orientation tolerance, $\delta\theta$, it is required that:
$|\Delta\theta(t) - \Delta\theta_{com}| < \delta\theta$. An example of a relative orientation station keeping maneuver is depicted below in Figure 9.

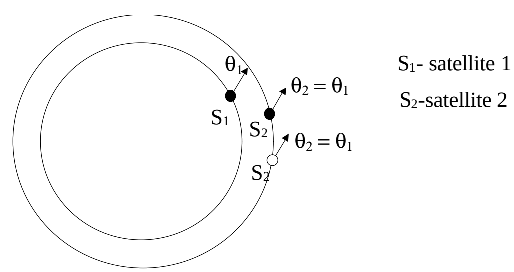

Figure 9. Example Relative Orientation Station Keeping Maneuver.

Spacecraft Bus

The spacecraft geometry is shown in Figure 10. Rather than partitioning the spacecraft into traditional subsystems, the spacecraft is broken into functions. These functions will share hardware and software wherever possible to maximize efficiency. One undesirable side effect of this efficiency is increased exposure to single-point failure. To mitigate this, simple, robust designs will be chosen over more complex ones. Additionally, critical control functions of the bus will be incorporated in a radiation hardened controller. The functional areas are Navigation, Guidance and Control, Flight Computer, Electrical Power, Communication, and Structure and Thermal Control.

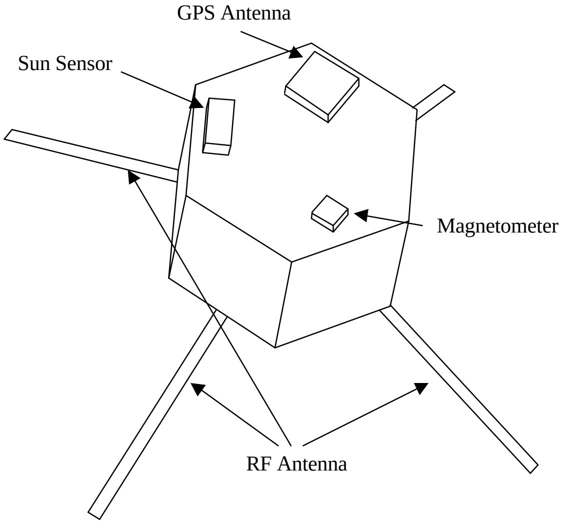

Figure 10. Spacecraft Geometry.

Navigation, Guidance, and Control

The primary payload on the NAVGOLD spacecraft will be the Navigation, Guidance and Control (NGC) package, including the sensor suite, and actuator/propulsion components.

The design requirements for NGC package fall under 3 main categories. First, the spacecraft must be able to perform high-accuracy relative navigation, to significantly greater accuracies than that available from current ground or space (TDRSS) tracking. Second, the NGC system must provide three-axis attitude stabilization and control of the spacecraft, both during maneuvers and coast. Third, the spacecraft must be capable of performing autonomous re-positioning and relative attitude maneuvers and station keeping.

To meet these requirements, the navigation, guidance, and control concepts must be defined. Two methods for relative navigation are being considered, both relying heavily on GPS. The first involves using a auxiliary measurement to augment the GPS position estimate. The auxiliary measurement will be direct relative range, which may be derived by a variety or means. The ranging sensors being considered include both cross-link r.f. ranging or laser-based ranging, both of which require precise onboard clocks. By optimally combining the GPS estimates and relative range measurement, the accuracy of the relative position estimate will be significantly improved, with meter-level range errors the goal.

The second relative-navigation technique would be based on differential-GPS concepts. Given that the relative navigation solution is of interest, and not the absolute navigation solution, the signals received by two closely-spaced GPS receivers can be combined to greatly improve the accuracy of the relative navigation solution. This method is valid provided the receivers are in relatively close proximity such that error sources affect both receivers identically.

With regard to the GPS system itself, a variety of systems are under consideration. Such systems vary from a complete COTS “black-box” system; to a custom built, in-house system; to a hybrid system. Examples of GPS systems being considered include the 1) Trimble TANS Vector receiver, 2) a unit developed by the University of Surrey in partnership with GEC Plessy, 3) JPL's microGPS unit, 4) a unit being developed by NASA Goddard or 5) a hybrid GPS system including a TIDGET™ GPS front end from NAVSYS Corp., coupled with custom estimation algorithms.

The guidance component of the NGC function consists of the algorithms used for either maneuvering or station keeping. The maneuvering algorithms include discrete-burn timing logic, along with the associated spacecraft pointing algorithms. The station keeping guidance will be based on algorithms derived from Hill’s equations, governing the relative position and velocity of orbiting spacecraft.

Baseline Design

The primary measures of merit used to evaluate the various design approaches are mass, power consumption, cost, computational complexity, and maturity of the technology. For the propulsion system, emphesis is placed on maturity, while the most risk will be accepted in the navigation and guidance functions (e.g., sensors and algorithms).

The baseline design concept for the navigation and guidance functions are depicted schematically in Figure 11, while the schematic for the baseline attitude determination and control concepts is shown in Figure 12. The solid lines indicate direction of data flow, and the dashed lines indicate selectable connections. For example, while the spacecraft are flying autonomously, updates of the desired state of the spacecraft from the ground may not be needed. The heavy dash-dot line indicates a critical command. These connections will only be used if a major malfunction occurs, and allow for limited spacecraft control in the event of the malfunction. The circled “T” indicates information that will be telemetered to the ground station. The lighter

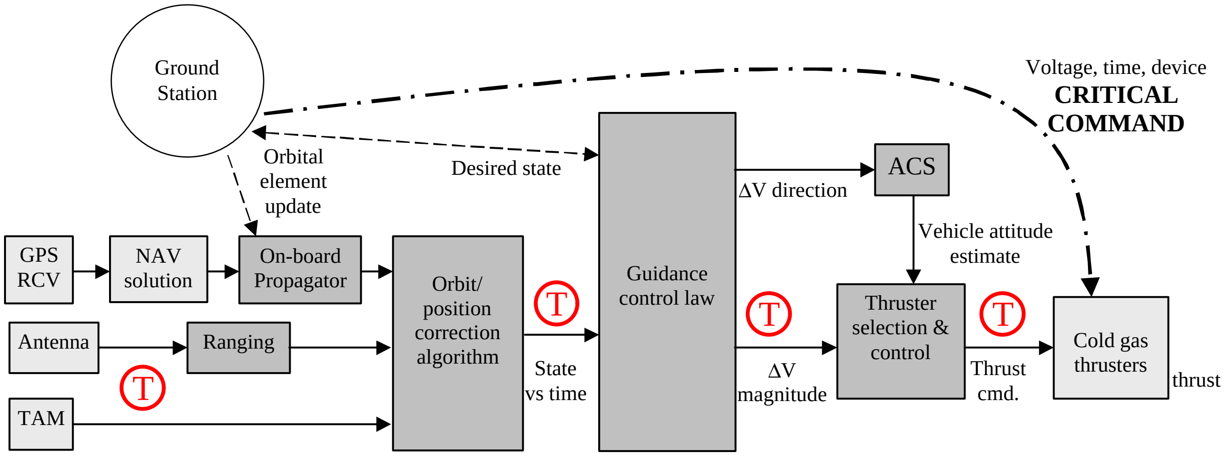

Figure 11. Block Diagram of Navigation and Guidance Function.

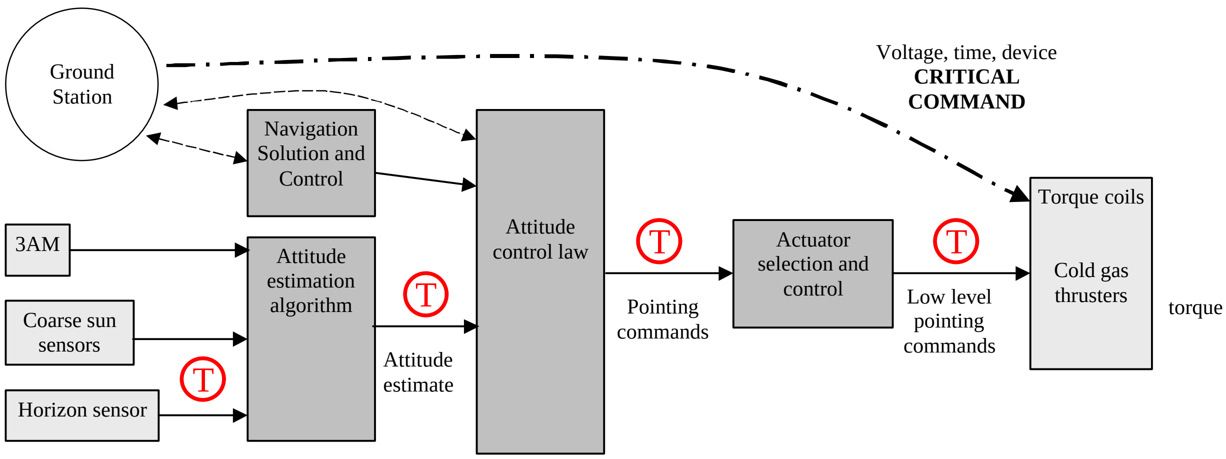

Figure 12. Block Diagram of Attitude Determination and Control Function.

boxes indicate actual hardware, whereas the darker boxes indicate software.

Regarding the NGC software, the development environment will be MATLAB/SIMULINK®, include maximum utilization of the Real_Time Workshop to enable rapid prototyping and real-time code generation.

The baseline navigation concept utilizes GPS, augmented with ranging augmentation using radio-frequency (r.f.) signals. The ranging system requires very precise clocks on both spacecraft, and candidates have been identified. The ranging system uses the same hardware as the communications cross-link function, and was a key reason why the r.f.-based method for ranging was selected over a laser-based method.

The baseline propulsion system used in the spacecraft design will be cold gas thrusters. This was chosen primarily due to its proven reliability and low risk. The cold gas thruster will include the propellant tank, propellant (N_2), valves, plumbing and six individual thrusters. The mass of the propulsion system will be approximately 1.5 kg, require approximately 10 watts of power for continuous burn, and supply approximately 25 m/s ΔV total.

The sensors used in the attitude control system will be digital sun sensors, a three-axis magnetometer, and a horizon sensor, all developed in-house at CU-Colorado Springs. A picture of the sun sensor is shown in Figure 13. The three-axis magnetometer, shown in Figure 14, will be implemented in-house, though the design will be based on a commercially available magneto-sensitive IC. The attitude sensor package will be approximately 0.5 kg in mass and consume very little power. The overall accuracy of each sensor is on the order of 0.5 degrees.

The actuators to be used by the attitude control system will be magnetic torque coils and the cold-gas thrusters.

Four torque coils will be located around the hexagonal prism structure on the outer panels of each spacecraft, as shown in Figure 15. The coils will be manufactured in-house, will be approximately 20 cm outer diameter by 2.5 cm cross section diameter, will have an approximate mass of 2 kg total, and be capable of generating a magnetic moment of approximately 1.25 A-m^2. These sizing results are based on the information presented in Figure 16, showing maximum torque per unit inertia plotted versus desired settling time.

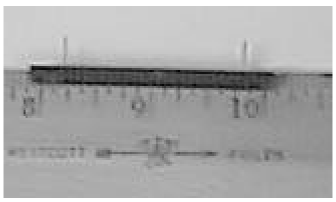

Figure 13. Fine Sun Sensor (Scale in Inches).

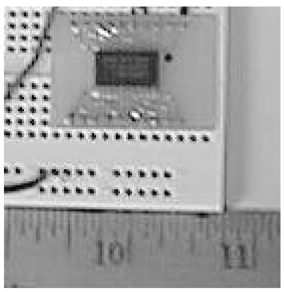

Figure 14. Three Axis Magnetometer (Scale in Inches).

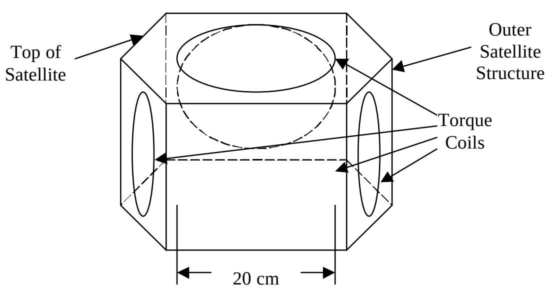

Figure 15. Torque Coil Location.

There will be three attitude control modes, one for each main operating mode of the spacecraft. These are safe-hold mode, sun-pointing mode, and maneuvering mode. Safe-hold mode, implemented both analog and digitally within the Command and Telemetry Unit (CTU), is designed to keep the spacecraft under positive attitude control after initial orbit insertion to counteract any initial attitude rates, and to provide a simple, reliable reversion mode. In this mode, the initial spin rates must be damped with a settling time of at most 3 orbits. From Figure 16, this requires a required maximum torque per unit inertia of $4 \cdot 10^{-4}$ sec^{-2}. For a body with a moment of inertia of 0.5 kg-m^2, this requires a torque of 0.2 mN-m. With an initial angular rate of 1 deg/sec, the safe-hold mode is capable of reducing the tumble rate to less than 0.1 degrees per second within 3 orbits, as shown in Figure 17. The time response of the torque generated by the torque coils during this maneuver is shown in Figure 18.

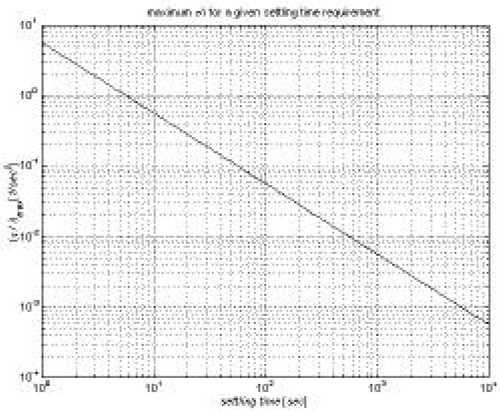

Figure 16. Settling-Time Torque Requirement.

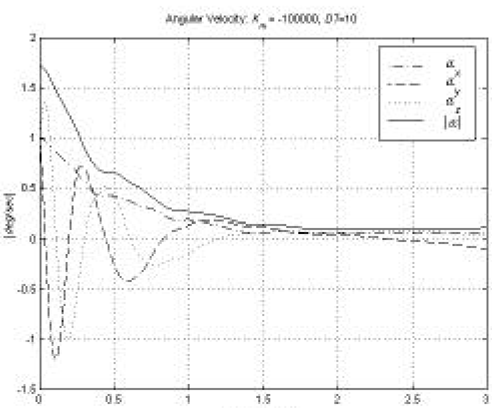

Figure 17. Angular Velocity Time Response.

Sun-pointing mode will be the mode that is used most frequently. It is designed to orient the spacecraft towards the sun for maximum power generation. Torque coils will also be used for actuation in this control mode.

Maneuvering mode will be employed only during propulsive maneuvering. This mode must be capable of performing (potentially) large slewing maneuvers, and especially orienting the thrusters properly for a maneuver. In this mode, the cold-gas thrusters will be used for actuation.

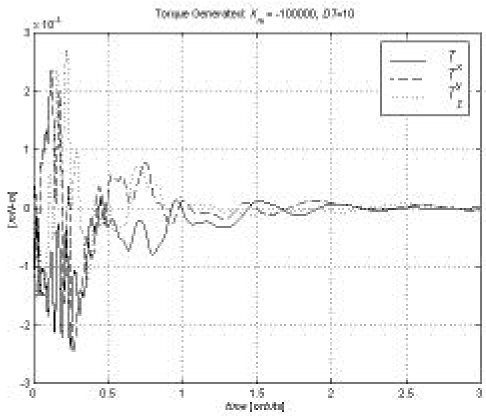

Figure 18. Torque Time Response.

Finally, the total mass of the entire NGC package, not including hardware shared with other systems (such as the flight computer and communications antennas), will be approximately 4 kg.

Flight Computer

The autonomy requirements of navigation and attitude control mandate the use of an on-board processor. Also, the need to control the spacecraft and communicate with the ground dictates the adoption of a traditional Command and Data Handling (C&DH) function. The flight computer provides the resources for these functions as well as all other telemetry, data handling, and on-board processing.

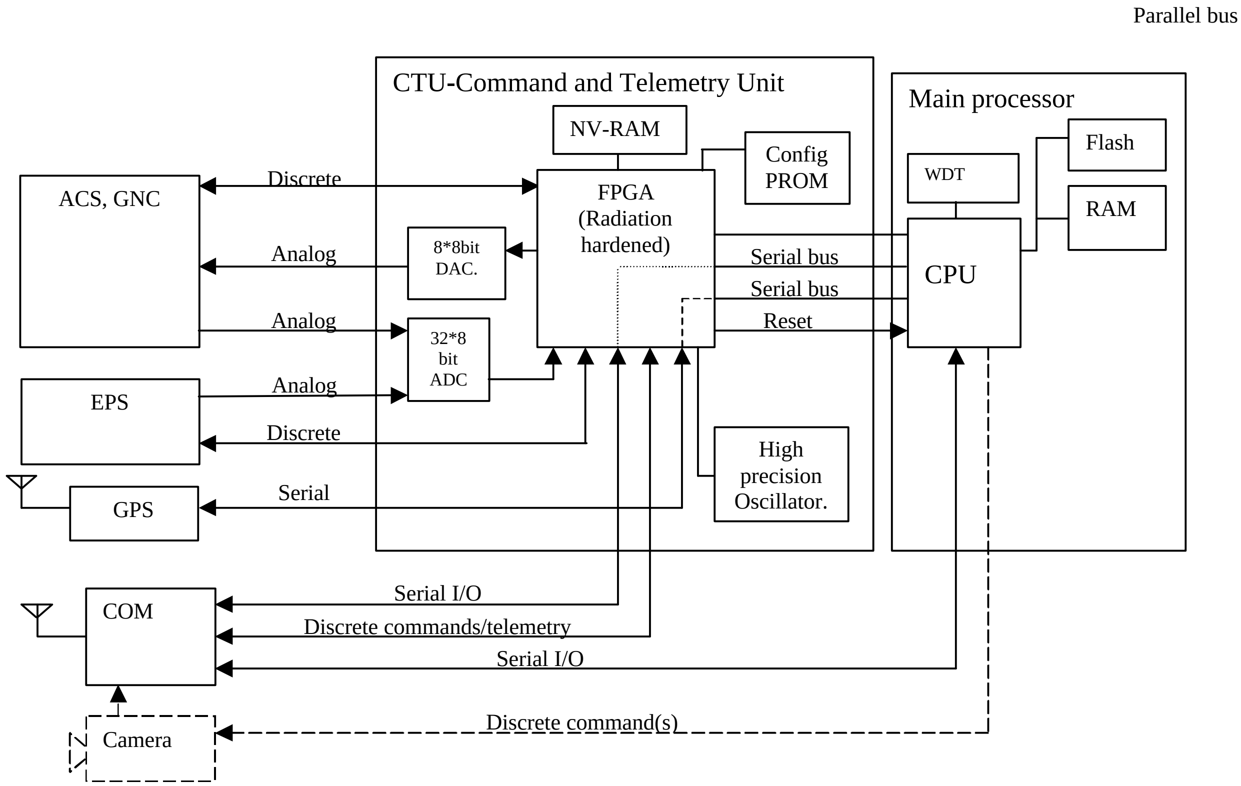

Figure 19 - Flight Computer Block Diagram

The flight computer is divided into two main parts, a Command and Telemetry Unit (CTU) and a main processor. The two parts are logically divided. Physically, they may either be separate or integrated on the same electronic circuit board. Logical separation facilitates parallel development of each function. Keeping the two parts physically separate improves the flexibility of development. Integrating the two physically can make the design smaller and more power efficient.

The main processor will be a COTS unit, presumably having never been flown in space, which makes that part a high risk element from a space environment standpoint. To mitigate against that situation the CTU, which is based on a radiation hardened field programmable gate array (FPGA), takes care of the basic and critical command and data handling. Since it is radiation hardened, it provides a more reliable control unit.

Command and Telemetry Unit
The CTU provides five basic functions:

1. Critical command decoding and execution.
2. Failure detection and spacecraft mode control.
3. Critical telemetry.
4. Electrical power and communication function controls.
5. Timing.

All serial commands from ground pass through the CTU. The CTU decodes the command and determines if it should be executed directly by the CTU. There are a limited number of critical commands that will be executed in the CTU. The CTU is designed to be as simple as possible, since there is no room for error.

A separate state machine in the FPGA will monitor select signals and will store the spacecraft mode. This provides the CTU failure detection capability as well as keeping the spacecraft operating mode in the rad-hard portion of the circuitry.

The functionality of the FPGA is stored in a configuration PROM that cannot be reprogrammed after the spacecraft is launched. To provide non-volatile memory for storing spacecraft status information etc., the CTU is equipped with a small non-volatile random access memory (NVRAM). The CTU is equipped with 32 channels of 8-bit analog to digital converter and 8 channels of 8 bit digital to analog converters. This gives the CTU access to critical telemetry and to critical control functions. Access to telemetry points and control outputs for the main processor is provided through the CTU. These control outputs include electrical power controls for critical battery charging and load control.

To be able to provide accurate timing for relative navigation, the CTU is equipped with a high precision oscillator.

Main Processor
The main processor provides five functions:

1. Non critical command decoding
2. Attitude control main processing.
3. Navigation, guidance and control processing
4. Test and diagnostic algorithms.

Commands for controlling payload devices and non-bus critical commands will be executed by the main processor. Attitude control and NGC processing will take the bulk of the main processor's capability. Other processing will include optimal battery charging and power management, spacecraft cross-link internetworking, and processing of any other payload devices in the spacecraft. Test and diagnostic algorithms, primarily used during development, integration, and test will also reside and be run on this processor.

The main processor will use a real-time operating system (RTOS). There are two operating systems being considered: Windows CE and VxWorks. The RTOS based on Windows CE makes it easy to develop software, as development can be performed on most any PC compatible computer. It also allows complete customization of the operating system functions. Since it is a modular operating system, it provides flexibility in design as components are added or removed, i.e. solid-state disks, PCMCIA devices, etc. VxWorks on the other hand is widely used for powerful embedded applications and in addition to being a modular OS, has development advantages. In this project, most of the software development will be in the NGC function. The bulk of the algorithms developed there will be done with MATLAB. The availability of cross compiler utilities enabling conversion and download of MATLAB code directly to VxWorks makes it attractive to shorten integration and test.

The main processor must be able to communicate through a serial line and store the code in flash memory, making it possible to update the software during flight. The interface between the CTU and the main processor is *via* serial lines, a parallel bus, and a reset signal that allows the CTU to reset the main processor.

Development Requirements
The requirements for the development are divided into those for the CTU and the main processor.

<u>CTU</u> **-** The basic cornerstone for the development of the CTU is an evaluation board specific for the FPGA chip being used. To develop the software that is needed for the FPGA, a VHDL compiler and synthesizer are needed. Finally, to integrate the design and put it on a circuit board a schematic and printed circuit board (PCB) editor are needed.

<u>Main Processor</u> **-** To customize and build a real time operating system for the main processor, an operating system specific development system is needed. As a development system for the general code (electrical power control, spacecraft internetworking algorithms, etc.), the C-language compiler for the operating system will be used. For Windows CE, these are Visual Studio and Visual C++. For VxWorks, it will be the development environment for the processor board

chosen. To be able to test and run the developed code before the hardware is fully developed, a reference or evaluation board equipped with the target processor is required.

Requiremenst

To allow support functions to size the needs of the flight computer, requirements were assumed. These are listed below.

<u>*Electrical*</u> -

- +3.3 V 100 mA peak, 30 mA typical
- +5 V 250 mA peak, 20 mA typical, <1 mA standby
- A/D channels: 8-bit, 0-4.096VDC input range
- D/A channels: 8-bit, 0-5VDC output range

<u>*Thermal and Other Requirements*</u> **-** All heat needs to be removed to keep the operating temperature within the commercial range: $0<T_0<50°C$. The parts are to be conformal coated to prevent arcing in the event of spacecraft charging.

Electrical Power

The Electrical Power Subsystem function (EPS) provides power and power management for the satellite. This function takes power from solar arrays or a ground power source, and uses it to power spacecraft devices and charge spacecraft batteries. The EPS must supply enough power to run the spacecraft throughout all modes of operation including during eclipse.

The EPS provides power to all other electrical dependent functions throughout the life of the spacecraft. The EPS accomplishes this by generating or accepting raw electrical power, converting the raw power into the useable regulated power, storing excess power for use during eclipses, and managing the power distribution throughout the spacecraft. For risk mitigation, the EPS is designed as simple, robust, and reliable as possible.

Power Requirements

The worse case power requirement for the satellite has been determined to be 7.7 Watts orbital average with a 28.8 Watts peak demand. This requires that the solar arrays to generate 18 Watts while in sunlight to support the spacecraft's power requirements and recharge the battery for the next eclipse. These requirements were developed with the following assumptions:

1. Orbit altitude, h_a = 400km (period ~92 min)
2. The satellite is in solar eclipse for 40% of the orbital period
3. Ground station visibility duration, t_{vis} = 8 mins. max (transmit time to download data)
4. The ACS is active during the orbit

Design

The EPS has three modes of operation: Charge, Discharge, and Maintenance. Charging occurs when the flight computer determines that the batteries are not at 100% charge and there is sufficient power available from a power source to charge them. Discharge occurs when power requirements exceed available power. Maintenance occurs when the batteries are fully charged and there is sufficient power to operate the spacecraft. In this mode, the batteries are trickle charged to maintain full charge. Based on solar array and battery parameters, the flight computer autonomously selects the operating mode. In addition, the mode can be over-ridden by ground command.

The EPS is partitioned into four main functional areas:

1. Generation
2. Regulation
3. Control, Distribution, and Telemetry
4. Storage

Together, these functional blocks work to receive power from the sun (or external power supply during development and testing), convert it to a usable range of voltages, charge the batteries, and distribute power to the various functions.

Generation

Power comes from one of the two sources **-** solar arrays or external power from ground support equipment (GSE).

Three types of solar cells were considered. Silicon (Si), Gallium Arsenide (GaAs) and high efficiency cells. GaAs was chosen because Si uses almost twice as many cells to produce the same amount of power, and the high efficiency cells are not available to us at this time The solar cells will be assembled into arrays, and these will be mounted to and insulated from the body of the spacecraft. This arrangement will provide the necessary power to the EPS without driving unreasonable Sun pointing attitude requirements.

The GSE power input will provide power during development, integration, and test. This will be supplied by an external power supply through the GSE interface, which will also contain signal lines for communicating with the flight computer.

Regulation

This function takes a raw voltage output from the solar arrays or GSE power and converts it for powering the spacecraft and charging the battery DC-to-DC converters are used in each case.

Solar array output will feed a battery charge regulator and the DC-to-DC converters, which generate the regulated power required by the spacecraft components. Battery charge regulation is determined by available power and loading A peak power tracker (PPT) algorithm running in the flight computer will control the battery charge regulator to extract maximum power from the illuminated solar arrays during normal operations. During anomalous operations, the CTU will control basic battery charging. During eclipse and low solar incidence conditions, the battery will feed the DC-to-DC converters.

Regulated power will be provided in two forms: low power, high regulation and high power low regulation. Low power, high regulation will provide voltage ranges of +/- 12VDC, +5VDC, and +3.3VDC. These sources will power the sensors and flight computer, and other low power devices requiring tightly regulated supplies. The high power, low regulation converter will provide nominally +12VDC and will be used to run the transmitters, magnetic torque rods, thruster solenoids, deployment mechanisms, and any other high current devices.

Control, Distribution, and Telemetry

The CTU controls spacecraft activation from stand by mode, provides control of power sources, over current protection, and power switching for load shedding. The flight computer controls certain devices that need to be powered on and off through commands to the CTU. The EPS internal busses will be protected with fuses and isolated with diodes. There is also a ground support equipment (GSE) interface that supplies power during development and pre-flight testing. The spacecraft will be equipped with a launch vehicle interface that will supply trickle charge power and inhibit signals to maintain the spacecraft in standby mod. Battery status (voltage, charge, and temperature) will be acquired and relayed as part of the housekeeping telemetry.

Storage

The spacecraft power requirements and maximum eclipse duration of approximately 40% of the orbit drive the storage requirements. Due to the 28.8W peak power demand, battery choice is driven by the electrical current demand rather than allowable depth of discharge. A battery constructed from rechargeable nickel-cadmium (NiCd) C size cells meets the requirements. To match the voltage and current requirements of the generation section, a single battery pack of 15 cells in series will be used.

<u>NavGold Communications</u>

Ground Station TT&C

The CU-Colorado Springs ground station will utilize a Yaesu FT-736R Amateur band transceiver capable of providing 25 watts of output r.f. power. The frequencies used for TT&C communications will be in the 430 MHz amateur radio band for both uplink and downlink. Simplex operation will be used. According to communications link budget analysis [7], an existing 15 dB gain 430 MHz cross element Yagi antenna with the previously mentioned uplink r.f. power should be adequate for communications to the spacecraft. The antenna is mounted on a 20 foot tower atop the CU-Colorado Springs Engineering building.

The antenna tracking mechanism consists of Yaesu G-5400B® azimuth and elevation rotors driven by Northern Lights Software Associates NOVA® software. This software accommodates satellite passes in all quadrants by incorporating a “flip” mode to overcome the mechanical stops at a south bearing. The tracking software also provides a scripting function by which selected future passes may be programmed for unattended operation. This feature may be used in times of minimal staffing.

The ground station TT&C facility will utilize United States radio amateur assigned frequencies in the UHF range to enable the acquisition of low cost TT&C equipment and insure that NavGold operation can be observed by radio amateurs around the world, thus providing an “extended” tracking network. A simplified block diagram of the ground station is shown in Figure 20.

Spacecraft to Ground Station TT&C

The NavGold satellite is extremely power limited due to its size and use of body mounted solar photovoltaic cells, therefore the TT&C communications function must operate with minimal power consumption. Link budget calculations [8], given the ground station output power and ground station receiver sensitivity, indicates that the onboard TT&C transmitter could be limited to approximately 2 watts of output r.f. power. Commercial r.f. equipment will be used onboard the spacecraft to minimize development and test requirements. At this time trade studies are being carried out for equipment selection. Current designs call for the use of 9600 bps TT&C data transfer rate to be compatible with available low cost satellite r.f. components. The data will be transmitted using FSK at the UHF operating frequency. A Terminal Node Controller (TNC) will decode data received at the ground station. A similar configuration will be onboard the spacecraft. Data recovered at the ground station

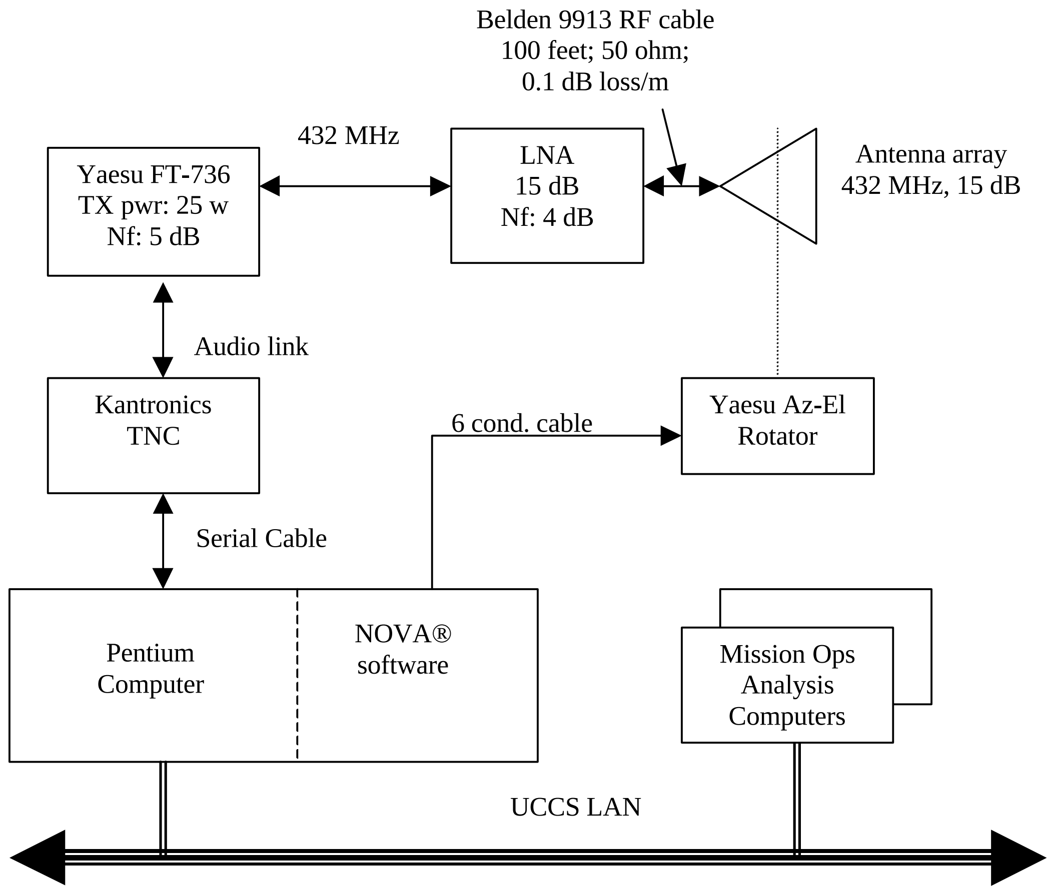

Figure 20. NavGold Ground Station Block Diagram

will be archived in a local computer. The data will also be transmitted over the CU-Colorado Springs LAN to the Mission Operations Center (discussed below) where health and status data will be decoded using LabView® software. This data will be displayed in real-time during a downlink session. Anomaly detection software may be used as an experimental analytical tool for health and status determination. Payload data such as position, relative range between satellites, magnetometer data, thruster action, as well as satellite status data may be formatted and input to the NavGold homepage.

Spacecraft to Spacecraft

The inter-spacecraft data link will perform two major functions. One is to relay status of the "follower" spacecraft to the "leader" spacecraft. This data will be identical to that transmitted from the "leader" to the ground. In the proposed NavGold mission data from the "follower" will be acquired by the "leader" and transmitted in sequence with "leader" health and status to the ground. Additionally, GPS position data as determined on board the "follower" will be transmitted to the "leader". Both sets of position data may also be transmitted to the ground. The second major function of the communications link is to provide relative spacecraft ranging measurements. One concept being explored is the use of a pseudorandom code sequence referred to a high stability clock on board both spacecraft. With one spacecraft transmitting the sequence at a known rate and beginning at a known time the second spacecraft can measure the difference between the received sequence and its internal clock generated sequence thus deriving a "time of flight" measurement hence distance information. This PR code must be of sufficient length to unambiguously accommodate large (e.g. tens of kilometers) separation distances. The PR code may be transmitted at regular intervals or provided on a crosslink subcarrier frequency.

Another concept being explored is the use of transponders to determine range. Inter-satellite ranges between 10 kilometers and 10 meters are planned for the NavGold experiment. Given this range envelope, on-board timing circuits may prove stable enough to permit measurements of either radio frequency or optical transponder-like beacon signals passed between the satellites. Such range data, when combined with GPS position data transmitted from one satellite to the other and treated as Local Area Augmentation data may permit a position vector to be derived for GNC use to achieve formation operation.

Inter-spacecraft communications will occur on a UHF frequency but non-concurrent with "leader" spacecraft to ground transmissions to avoid radio frequency interference issues. The inter-spacecraft link may also require a very low power r.f. transmitter due to the relative proximity of the two vehicles thus not requiring significant electrical power. The inter-spacecraft data will include GPS information, payload status, bus status and, potentially, range beacon data. The link margin for this inter-spacecraft communication is shown in Figure TBD. Relatively simple and low power output transceivers are planned for this operation and will be controlled, as the TT&C links, by the C&DH computers. Separate antennas will be used for inter-spacecraft communication to reduce complexity.

Structure and Thermal Control

The spacecraft shall survive all environments induced by the launch vehicle and low earth orbital conditions without structural, mechanical, or thermal control failure. Based on anticipated launch-geometry constraints, each spacecraft's dimensions shall not exceed a diameter of 19.75 inches with a vertical height of 12.63 inches. These dimensions will provide minimal spacing between the spacecraft stack and the enclosure. This baseline dimension requires that no induced vibrations from the launch vehicle shall cause the spacecraft stack to deflect more than one-quarter of an inch. In addition the spacecraft's stiffness shall be sufficient to prevent induced oscillations resulting in structural instability or the failure of any spacecraft components (e.g. structural frequencies which coincide with operating frequencies of computer, etc).

To meet these requirements three major design variables must be addressed. These variables consist of the spacecraft geometry, the structural support method, and the materials used in the structure of the spacecraft.

The spacecraft's geometry was designed as a trade-off between maximizing surface area and minimizing wasted space in the launch enclosure/fairing, while keeping the fabrication requirements for construction at a minimum. Initially, the shape was designed be to a cylinder based on the assumption that the launch enclosure is cylindrical. This structure has the advantage that it maximized the surface area and minimized wasted space in the canister; however, a cylindrical structure is more difficult to machine and to mount typically planar components onto the curved surface. Therefore, other geometries were considered including rectangular, hexagonal, and octagonal prisms.

The cube has the advantage that it was simple to machine and construct, but the cube had nearly 22% less surface area than the cylinder. This lead to the hybrid shape of either a hexagonal or octagonal prism, which have machining and construction characteristics similar to the cube, while providing surface area characteristics similar to a cylinder. The hexagonal prism was judged to be easier to construct, has a greater length of flat surface for mounting components than the octagon, and as a result is lighter in weight due to fewer joints. So the hexagonal prism was chosen for the spacecraft's geometry.

Three separate structural configurations where considered for the NavGold spacecraft. The first configuration considered was an endoskeleton support system, with the load-bearing components located within the spacecraft. The endoskeleton structure has the advantage of a high strength-to-weight ratio, but this type of structure requires complete knowledge of the spacecraft components and careful design to be fabricated. Worse, if the spacecraft must be altered in some way, the structure may need to be completely redesigned. The second configuration was a "space truss" exoskeleton structure, which has an extremely high strength-to-weight ratio, but requires the most design effort and is the most complex to construct. Finally, a simpler approach was considered where the skin of the spacecraft was used as the structural support for the components of the spacecraft, akin to a box or "panel body mounted" structure. This structure has the advantage that it can be easily redesigned and built quickly, a necessary characteristic when designing and building a spacecraft with such an aggressive schedule.

Conventional monocoque metallic alloy designs using aluminum, titanium, aluminum lithium, and steel were investigated. Each of these materials has merits, high stiffness to weight and strength to weight ratios, and most are fairly ductile which allows for better distribution of loads and generally a yield before ultimate failure situation. However, the advantage of having a precision machine shop to machine, mill, weld, and rivet the structure together is not available at our discretion. We therefore intend to fabricate sandwich construction panels using intermediate modulus graphite/epoxy face sheets with a honeycomb or closed cell foam core. Due to the light weight of the nanosatellite and its components, the strength to weight ratio of the structure shall not be the primary sizing factor of the structure, but instead it will mostly be driven by stiffness to weight ratios. As a result, the sandwich panel construction is the most efficient structure for stiffness to weight.

The current baseline design of the spacecraft's structure consists of a hexagonal shaped "panel body mounted" structure. Each sandwich panel will be approximately one quarter of an inch thick to provide the strength and stiffness needed to survive the launch environment. The top of the spacecraft will be designed to be removable to allow easy access to the spacecraft components and for subsystem checkout. All components will be mounted to the inside or outside of the skin structure, except for the propulsion tank, that will be supported by internal braces.

Passive thermal control will be used to maintain components within their operating limits. This can be accomplished through the effective placement of components and use of thermal coatings, fins, multi layer shields, and low or high thermal conductive materials (e.g. graphite/epoxy or copper foil respectively). The components of the spacecraft will be oriented to minimize the cross products of inertia about the spacecraft center of mass, and the variations in inertias and location of center of mass, to minimize the effect on the attitude dynamics and control of the spacecraft.

The two spacecraft shall be mechanically and electrically attached to each other as they are stowed in the launch vehicle canister. After deployment from the launch vehicle, a health check shall be performed prior to the separation of the two spacecraft from each other.

Mission Operations

The NavGold Mission Operations Center will be linked by the CU-Colorado Springs Local Area Network to the mission ground station. NavGold will make use of an existing ground station facility first designed for the Falcon Gold spacecraft. The ground station consists of UHF and VHF communications equipment, C&DH computers, spacecraft tracking software for providing ground station antenna tracking, and software for satellite pass prediction and visibility opportunities determination.

Orbit analysis will be performed using a suite of software developed previously for mission support in flight dynamics and ground-based navigation. Data input to the ground station facility will include orbital data derived from the space-born GPS, plus U.S. Space Command spacecraft ephemeris data acquired through a dial-up bulletin board and pass schedule information from the Mission Operations Center. The ground station will acquire and archive all downlinked data. Following successful archiving, the data will be transferred to the Mission Operations Center for data reduction and analysis. Furthermore, uplinked data will be archived to assure historical problem resolution efforts operate on actual transmitted information.

Mission Planning

The CU-Colorado Springs NavGold Mission Planning function will make use of an automated mission planing and scheduling tool. (Selection of which specific tool is currently under study.) This tool will facilitate development of the spacecraft command structure by which not only flight safety but also the scientific and engineering experiments will be performed. The tool will also be used during test and checkout of the completed spacecraft prior to delivery to the launch facility. Post-launch, the mission operations tool will be used to plan operations during satellite passes over the ground station receiving facility. In this manner a timeline of required operations will be developed, simulated and validated prior to a given pass.

The planning tool will also assist the operators in resolving potential resource conflicts such as scheduling mutually exclusive events onboard the spacecraft. An example of this situation for NavGold would be the simultaneous commanding of magnetic torque bar operations and reading of magnetometer data. Another example would be the simultaneous transmission of data from the spacecraft to the ground facility while commanding a spacecraft to spacecraft link thus jeopardizing the overall electrical power budget.

Inputs to the mission operations function include ephemeris data that would be used, among other functions, to ascertain the time in view for the ground station facility. This constraint determines the command uplink and data downlink times thus effecting the amount and type of data to be exchanged between the ground and the spacecraft.

Activity Planning and Development

Once the appropriate mission operations planing and scheduling software tool is defined, interfacing to the TT&C facility will be defined. Most high level planing and scheduling tools develop output not directly suited to transmission to an onboard flight command and data handling (C&DH) computer. Therefore the mission operations pass scheduler must operate in concert with a suitable translating interface to provide the appropriate command and control information for the C&DH flight computer.

NavGold is investigating available translating tools in this arena. The ideal tool would be one in which allows for data, timed event (such as pass scheduled inputs), operator input and autonomous operation of the NavGold control system. Once the rules for operation have been developed using such a tool the resulting code would be uplinked to the onboard C&DH computer.

Mission Control and Management Functions

Operation of both the Mission Operations Center and the ground station may involve few personnel. In this event the NavGold team is investigating automatic tools which could be used to acquire, track and download timed-event driven spacecraft data. One such tool was used during the Falcon Gold and High Altitude Balloon project flights at CU-Colorado Springs.

TT&C computers receiving the downlinked data would be programmed to automatically receive and store data to a suitable database during unattended operation periods. Following modern practices it is conceivable that anomaly detection software, such as used for certain commercial spacecraft operations (e.g., the French SPOT Image [5]) and scientific spacecraft (Japan's Nozomi Mars probe [6]) could be used to detect and alert key personnel during unattended operating periods.

As indicated previously, certain critical "safety of flight" bus and payload engineering data will displayed in real time in the Mission Control Center. LabView® software will be used to display data such as bus electrical power subsystem (EPS) status, critical temperature measurements, etc. LabView® may also be employed by the payload specialists for indications of thruster activity, attitude, and spacecraft derived position. More detailed analysis will be carried out post-pass by specialized applications.

System Integration and Test

Acquisition of a resource management system and spacecraft control language development tool such as mentioned above will also greatly facilitate system integration and test by providing a pseudo-operational environment. In this situation various experiment modes as well as flight control modes can be simulated, evaluated, verified and validated in a controlled test environment. Following completion of the communications links these tools can be used over the r.f. link to provide a near operating situation test.

Summary and Conclusions

The NavGold project represents a unique opportunity for developing and demonstrating new approaches to formation flight, autonomous relative navigation, and low-cost spacecraft development in a university environment. The major programmatic challenges include an aggressive schedule and limited financial budget. The major technical limitations include power generation and storage, and the tight mass and volume budgets. However, the NavGold team has enthusiastically taken on these challenges, and are eager to move ahead towards a successful 2001 launch.

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