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ORION: A LOW-COST DEMONSTRATION OF FORMATION FLYING IN SPACE USING GPS

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Abstract

This paper describes research on the design of a GPS based relative position and attitude sensing mission called Orion. The current Orion design consists of a fleet of 6 micro-satellites launched simultaneously into low Earth orbit to demonstrate coordinated attitude control, relative navigation and control, and formation initialization techniques in space. This approach represents a new systems architecture that provides many performance and operations advantages, such as reduced operating cost, enhanced mission flexibility, and improved science observations. A key objective of Orion is to demonstrate Carrier-Phase Differential GPS (CDGPS) techniques to autonomously track the relative position and attitude between several spacecraft. Based on a research program focused on low-cost spacecraft design, Orion represents a critical step towards the realization of formation flying and virtual platform capabilities. [1]

1 Introduction and Motivation

A revolution in spacecraft guidance, navigation and control technology has started through the use of GPS to autonomously provide vehicle position, attitude, and timing information. Not only will these innovations result in significant reductions in the weight, power consumption, and cost of future spacecraft attitude and orbit determination systems, they should also result in a significant reduction in ground operations costs through enhanced vehicle autonomy.

Recent results have demonstrated that Carrier-Phase Differential GPS (CDGPS) techniques can be used to autonomously track and then control the relative position and attitude between several spacecraft [1, 2, 3, 4, 5, 6, 7, 8]. This sensing technology can be used to develop a *virtual spacecraft bus* using automatic control of a cluster of micro-satellites to replace the monolithic bus of current Earth Sciences Enterprise (ESE) satellites (such as Landsat-7) [3, 6]. Many future space applications would benefit from using this formation flying technology to perform distributed observations, including: earth mapping (SAR, magnetosphere), astrophysics (stellar interferometry), and surveillance.

The goal is to accomplish these science tasks using a distributed array of much simpler, but highly coordinated, vehicles (e.g., micro-satellites). This approach represents a new systems architecture that provides many performance and operations advantages, such as:

1. Enables extensive co-observing programs to be conducted autonomously without using extensive ground support, which should greatly reduce operations cost of future science missions.
2. Increased separation (baseline) between instruments could provide orders of magnitude improvement in space-based interferometry. A distributed array of spacecraft will significantly improve the world coverage for remote sensing, and will enable simultaneous observations using multiple different sensors.
3. Replacing the large complex spacecraft of traditional multi-instrument observatories with an array of simpler micro-satellites provides a flex-

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ible architecture that offers a high degree of redundancy and reconfigurability in the event of a single vehicle failure.

4. Places the design emphasis on building and flying the science instruments, not on the development of the bus platform itself. Allows standardization of the satellite fabrication process, which will reduce costs.

5. Enables the low cost, short lead-time instruments to be built, launched, and operated immediately. The more costly, long lead-time instruments can then join the fleet when available. This approach would also allow new or replacement instruments to join the formation as they are developed or needed.

However, these benefits come at a cost because the new systems architecture poses very stringent challenges in the areas of:

1. Onboard sensing required to perform the autonomous closed-loop relative navigation and attitude determination and control.

2. High-level mission management to enable task allocation across the fleet of spacecraft.

3. High-level fault detection recovery to enhance the mission robustness.

As these points suggest, the onboard autonomy of these spacecraft must be significantly enhanced to reduce ground support required. Several research efforts are focused on this problem [9, 10]. Because these tools can be extended to the formation flying problem, we will not discuss this issue further.

Strong interest in the formation flying concept has developed as a result of two missions that are currently under development as part of the NASA New Millennium Program (NMP). The first, the New Millennium Interferometer (NMI), is a formation of three spacecraft in either GEO or solar orbit (0.1 AU from the Earth) to be used for long baseline optical stellar interferometry [11]. Two of the spacecraft will be light "collectors", separated by several kilometers, that focus light from a distant star onto a third "combiner" spacecraft that forms the interference pattern. To form this pattern, the optical path between the spacecraft must be controlled to within a fraction of a wavelength of light. To achieve this, a layered control approach has been proposed, one layer of which will use CDGPS type sensing to regulate the formation positions to within a centimeter [8]. Very precise formation flying of this type would also be required for distributed SAR and Earth imaging missions.

The second is the EO-1 mission which is planned to be a co-flyer with the Landsat 7 spacecraft. The scientific goal of the EO-1 mission is to validate the results obtained with the multi-spectral imager onboard Landsat 7 by taking images with a similar instrument onboard the EO-1 spacecraft. To achieve this, the two spacecraft must be flown in formation so that the relative distance between them can be controlled such that the EO-1 imager is viewing the Earth through the same column of air as the Landsat 7 imager. This will require a formation flying accuracy on the order of 10-20 m, which is an example of coarse formation flying [1]. The initial objective was to demonstrate formation flying using the EO-1 and Landsat 7 spacecraft, but because of a variety of budget and time constraints, no cross link will be possible between the two spacecraft. EO-1 will demonstrate onboard closed-loop autonomous orbit control using AutoCon™, which represents a key step. However this will not provide a true demonstration of formation flying spacecraft.

As a result, the *Orion* mission was developed to demonstrate the true formation flying concept that involves several spacecraft (6) navigating collectively. This mission will be used to validate key sensing and control issues associated with formation flying, and it represents an important step towards the virtual platforms envisioned for future Earth Sciences Enterprise missions. This paper discusses recent work on Orion with a particular focus on the GPS sensing, the mission plan, and the spacecraft bus design. The micro-satellite design is based on a modified version of the low cost, low weight spacecraft bus developed by Stanford University and the amateur radio satellite community called SQUIRT (Satellite Quick Research Testbed) [12].

2 Formation Flying Testbeds

Precise formation flying requires an accurate measurement of the formation states, *i.e.,* the relative

Fig. 1: Formation Flying Testbed

attitude and positions of the vehicles. GPS provides a promising technique for sensing these variables at a much lower cost than combinations of conventional spacecraft sensors such as star trackers, horizon/sun sensors, and inertial measurement systems. These techniques are based on Carrier-Phase Differential GPS (CDGPS) that was developed to improve the accuracy of GPS measurements [13]. CDGPS is a relative position measurement technique that helps circumvent S/A and many other error sources in the basic GPS measurement. Given GPS measurements at two nearby antennas, relative position between these antennas can be estimated to a high degree of accuracy based on tracking the relative phase of the GPS carrier waves. Of course, there is an integer number of wavelengths difference between the phases measured at the two antennas. This integer ambiguity is not directly observable and must be estimated or calibrated by an additional sensor. Nevertheless, CDGPS has been used successfully in a number of applications including automatic precision landing of commercial aircraft [14], and attitude estimation for spacecraft in Earth orbit [15].

To investigate the guidance, navigation, and control issues associated with precise formation flying, a formation flying testbed has been created in the Stanford Aerospace Robotics Laboratory [2, 3, 6, 17]. The testbed consists of 3 active free-flying vehicles that move on a 12 ft $\times$ 9 ft granite table top (see Figure 1). These air cushion vehicles simulate the zero-g dynamics of a spacecraft formation in a horizontal plane. The vehicles are propelled by compressed air thrusters. Each vehicle has onboard computing and batteries, and communicates with the other vehicles via a wireless Ethernet, making them self-contained and autonomous.

One of the limitations of GPS for ground testing these formation flying concepts is the reliance on observability to the twenty four NAVSTAR satellites in Earth orbit. This precludes the use of GPS indoors. This limitation has been overcome by the use of pseudolites: transmitters that emit GPS-like signals [16]. Pseudolite technology enables the application of GPS techniques to situations where the NAVSTAR spacecraft are not visible (*e.g.* indoors or deep space). Pseudolites can also be used to augment the existing NAVSTAR system as well. Recent results have demonstrated the feasibility of using a CDGPS based sensing system (with pseudolites) to achieve relative navigation accuracies on the order of 2 cm and $0.5°$ on this formation of prototype space vehicles. Refs [6, 17] provide more details on this experimental facility.

A second testbed has recently been developed to demonstrate formation flight in three dimensions using lighter-than-air vehicles (blimps) [18]. The blimp formation is also operated indoors, but in a large highbay. This second testbed will be used to demonstrate that various GPS errors, such as the circular polarization effect, can be modeled and eliminated from the measurement equations. The errors were not present in the 2D testbed because the antennae on each vehicle all roughly point in the same direction, but the errors will play a crucial role on-orbit because the spacecraft can undergo more general 3D motions.

These testbeds are being used as a precursor to the Orion flight vehicles to validate the GPS sensing algorithms. In the future, they will also be used to test the flight control software, the inter-vehicle communication, and the actuators.

3 GPS Estimation Issues

This section provides a brief overview of the equations used to estimate the relative position and attitude of the vehicles using the GPS carrier phase measurements. We first consider the approach used for the ground testbeds an then discuss the changes required to use this measurement approach on-orbit.

Consider the case shown in Fig. 2, which corresponds to the situation for the ground based testbeds. The unknown state of the ith vehicle is the 7×1 vector $X_i = [P_i, \quad E_i]$, where P_i is the position of vehicle

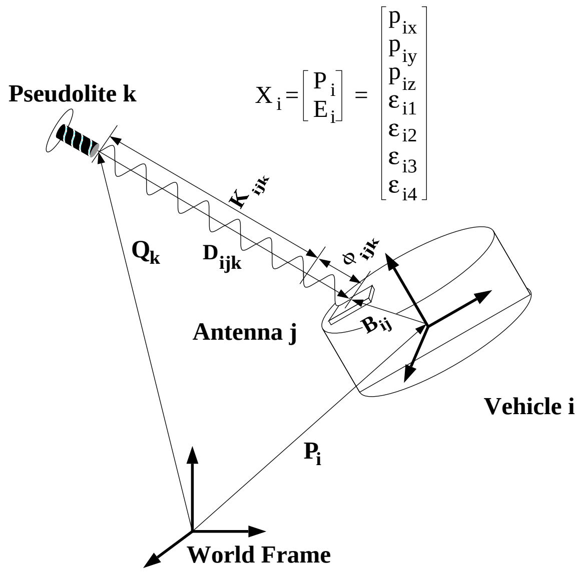

Fig. 2: Definition of the vehicle state and GPS measurements for the indoor formation flying testbeds.

i with coordinates (p_{ix}, p_{iy}, p_{iz}), and E_i is the orientation of the vehicle, represented in quaternion form $(\epsilon_{i1}, \epsilon_{i2}, \epsilon_{i3}, \epsilon_{i4})$.

The measured carrier phase at antenna j of vehicle i from pseudolite k is then

$$\phi_{ijk} = |D_{ijk}| + c\tau_{vi} + c\tau_{pk} + \lambda K_{ijk} \qquad (1)$$

where D_{ijk} is the line-of-sight vector from the phase center of the pseudolite antenna to the phase center on each receive antenna. The terms $c\tau_{vi}$ and $c\tau_{pi}$ represent the portion of the phase incurred by clock errors between the transmitter and the receiver, and are the dominating terms in the measurement equations. These terms are eliminated by differencing over measurements from multiple antenna and vehicles. The term K_{ijk} is the integer ambiguity for each antenna and pseudolite pair. The antenna location relative to the nearby pseudolite is clearly a function of the vehicle position and attitude, and thus D_{ijk} is a function of the entire vehicle state X_i.

The intra-vehicle single differences contribute primarily to the attitude determination for each vehicle. These measurements are obtained by differencing between the master antenna ($j = m$) and each of the slave antennas j of vehicle i for measurement from pseudolite k

$$\Delta\phi_{ijk} = |D_{imk}| - |D_{ijk}| + \lambda M_{ijk} \qquad (2)$$

$\forall k$ and $\forall j \neq m$. The M_{ijk} in Eq. 2 are the intra-vehicle integers.

The inter-vehicle double differences contribute primarily to the determination of the relative positions between each vehicle. Given N pseudolites, there are $N-1$ unique double differences between pseudolites k_1 and k_2 ($k_1 \neq k_2$). These differences are calculated in order to eliminate the remaining effects due to clock differences $c(\tau_{v1} - \tau_{v2})$

$$\begin{aligned} \nabla\Delta\phi_{ijk_1k_2} &= |D_{imk_1}| - |D_{jmk_1}| \\ &-(|D_{imk_2}| - |D_{jmk_2}|) + \lambda N_{ijk_1k_2} \end{aligned} \qquad (3)$$

$\forall k_1$, k_2 with $k_1 \neq k_2$. The $N_{ijk_1k_2}$ in Eq. 3 refers to the inter-vehicle integers.

The double difference measurements are coupled to the states of the two vehicles used in each pairing, so all of the measurements must be combined to calculate the entire formation state. From equations (2), (3), and the quaternion constraints ($\epsilon_{i1}^2 + \epsilon_{i2}^2 + \epsilon_{i3}^2 + \epsilon_{i4}^2 = 1$) , the complete set of measurements can be related to the vehicle states:

$$\begin{bmatrix} \Delta\phi_{1jk} \\ 0 \\ \Delta\phi_{2jk} \\ 0 \\ \Delta\phi_{3jk} \\ 0 \\ \nabla\Delta\phi_{12k_1k_2} \\ \nabla\Delta\phi_{23k_1k_2} \end{bmatrix} = \begin{bmatrix} h_1(X_1) \\ h_c(X_1) \\ h_2(X_2) \\ h_c(X_2) \\ h_3(X_3) \\ h_c(X_3) \\ h_{12}(X_1, X_2) \\ h_{23}(X_2, X_3) \end{bmatrix} + \lambda \begin{bmatrix} M_{1jk} \\ 0 \\ M_{2jk} \\ 0 \\ M_{3jk} \\ 0 \\ N_{12k_1k_2} \\ N_{23k_1k_2} \end{bmatrix} \qquad (4)$$

In this equation, h_i is a set of nonlinear functions of X_i, $h_{i_1i_2}$ is a second set of nonlinear functions of the given vehicle states, and h_c is the quaternion constraint function for each vehicle. For simplicity, we have written the measurement equations assuming that the double differences are performed between sequentially numbered vehicles.

Given the phase measurements for the vehicle formation, the optimal estimate of the formation states $X_i(t)$ can be solved in real-time using nonlinear estimation techniques. However, as shown in Eq. 4 it is also necessary to obtain an estimate of the single and double difference integer ambiguities so that these can be subtracted from the phase measurements. Refs [3, 6, 19] discuss integer resolution approaches based on the motion and search techniques [20, 21]. These initial results are promising, and work continues on implementing these approaches on the testbed shown in Figure 1.

This calculation solves for the absolute positions of the vehicles in the formation. The relative positions can then be found by differencing these absolute estimates. A CDGPS sensing system typically provides a much more accurate estimate of the relative position between the vehicles because many of the most important error sources are predominantly common-mode and their effects are removed by differencing the absolute position estimates.

3.1 On-orbit Operations

To this point the discussion has focused on formation flying techniques in an indoor environment using pseudolites. It is important to note how this work relates to operations in a Low Earth Orbit environment using the NAVSTAR satellite constellation. This section discusses some of the key differences between the indoor and LEO environments and the impact of these differences on the Orion mission.

Synchronous transmitters: One key difference is that the indoor pseudolites are not synchronized, whereas the NAVSTAR satellites have extremely accurate clocks that are closely synchronized. In moving to the LEO environment, this synchronization is very beneficial, as will be discussed in what follows.

Motion: Spacecraft in a low Earth orbit environment will generally experience very different motion than most terrestrial vehicles. Furthermore, the NAVSTAR satellites are in motion, changing position relative to the LEO satellites (which differs from the indoor case with fixed position pseudolites). This relative motion between the NAVSTAR satellites and LEO spacecraft will result in a large Doppler shift of the received RF signals. As discussed in Ref. [15], these large Doppler shifts significantly increase the frequency range that must be searched to acquire and lock onto the GPS signals. These changes impact the selection of the integer initialization approach.

The LEO spacecraft may also rotate at rates and in directions that are not common to terrestrial vehicles, resulting in an issue of how to keep the NAVSTAR satellites in view of the vehicle antennas. This can be resolved using many (non-aligned) antennas to increase the sky coverage.

Geometry: The scale of the relative distances are quite different for the indoor and LEO environments. In the indoor case, the distance from the formation to a pseudolite is approximately the same as the distance between vehicles within the formation. This presents a number of issues that are discussed by Zimmerman [2]. One of the most important issues is the difference between spherical or planar RF signal wavefronts. In the indoor case, with a near constellation, the RF signal wavefront must be assumed to be spherical. This requires a nonlinear mapping of carrier phases to the vehicle states (as given in Eq. 4). For the LEO environment (or any outdoor terrestrial case using the far constellation) this mapping is greatly simplified because we can assume planar wavefronts.

Environment: The differences in operating environments will have many other impacts as well. For example, we would expect much less multi-path in LEO. Multi-path is reflected GPS signal interference which degrades the performance of the sensor. However, the large temperature fluctuations characteristic of the space environment will change the GPS line biases, which could degrade the sensing performance. Design of the spacecraft, including placement of antennas to minimize multi-path and thermal control, will be required to resolve these issues.

Resolving these differences: As discussed, some of the changes associated with moving from indoor to LEO operations can be accounted for in the design of the spacecraft. Other changes, such as the increased Doppler shifts, the satellite motion, the far constellation geometry, and the synchronous signals will result in changes to the GPS estimation algorithm and measurement equations.

In particular, in the LEO case, the vector D_{ijk} changes, and can be considered to be only a function of position, decoupling the attitude and position solutions. Because of the availability of a synchronized signal, the position can be determined by pseudoranging. The pseudorange equation is similar to Eq. 1, with $K_{ijk} = 0$, τ_{pk} assumed known and small, and τ_{vi} is the user time bias that must be calculated. These pseudorange measurements are typically linearized [13] and then solved to determine crude estimates (accuracy on the order of 10 m) of the absolute vehicle positions (in Earth-centered reference frame). The attitude of the spacecraft is calculated using single differences [15]

$$\Delta\phi_{ijk} = \left[S_k^T\right]_E \mathcal{A}_{BE} \left[b_j\right]_B + \lambda M_{ijk} \qquad (5)$$

where S_k is the normalized line-of-sight vector to the kth GPS satellite represented in the Earth-centered frame of reference, and b_j is the baseline vector from the jth antenna to the Master antenna represented in the body frame. The direction cosine matrix $\mathcal{A}_{BE}$ is a function of the vehicle attitude. Note that this single difference equation is similar to Eq. 2, except in this case, given knowledge of the line-of-sight vectors, the attitude calculation decouples from the vehicle position. The relative positioning between the spacecraft could then be improved using double differences, once again resulting in centimeter-level accuracies. The procedure for calculating the double difference equations is the same as the one used to develop Eq. 3 from Eq. 2. Note that in this case the time information available from the NAVSTAR transmitted data can be used to time tag the measurements on different spacecraft, resulting in synchronization accuracies on the order of 10's-100's nanoseconds.

Initialization: The approach to determining position and attitude on start-up will also change in the LEO environment. The first issue is the increased Doppler shifts. Without any modification, the time required to acquire a particular satellite's signal could be longer than the total time the satellite is in view, potentially resulting in the situation that a position solution is never acquired. This problem can be remedied by including an orbit propagator (also required for the dynamic state estimator). This propagator would estimate the satellite's position and velocity, as well as estimate the NAVSTAR satellite's position and velocity (based on the last recorded almanac data). With this information, the frequency search space would be greatly decreased, thus reducing the time to acquire the GPS satellites [15]. Another difference in the initialization is the resolution of integer cycles in the attitude and relative position equations. Unlike the fixed pseudolites, the NAVSTAR satellites are in motion, changing the relative line of sight to the user formation. Therefore the initialization would not require additional motion from the user vehicles, because they need only wait and the line-of-sight directions to the GPS constellation will change sufficiently. Fortunately, these initialization algorithms can be tested prior to flight using the hardware-in-the-loop GPS simulator available at Stanford.

4 The ORION™ GPS Receiver

As is clear from the preceding sections, GPS sensing is a key element in the autonomous navigation and control system for the Orion formation flying mission. Thus the micro-satellites that are discussed in more detail in the next section must include flexible, inexpensive, but very capable GPS receivers. Several GPS receivers have already been developed for space applications and these were compared using a variety of criteria: physical size, power required, number of channels, space experience, degree of familiarity at Stanford, cost, access to the source software, and performance. Based on this analysis, it was clear that none of the available receivers was a good match in terms of the most important features: power, code access, and cost. Thus the decision was made to develop an in-house GPS receiver that would be expandable, modular, and have an open software architecture. These goals were met by modifying the Mitel GPS ORION™ chipset[2].

Hardware: The ORION design was modified to incorporate dual front ends on a single circuit board, with a capability to receive an off-board clock signal to drive the receiver circuitry. These modifications allow for an arbitrary number of dual front-end receiver cards to be chained together using a common clock. In this configuration any number of antennas can be used to form a GPS based attitude system. A four antenna receiver has been constructed using this approach. The ORION lower board has 3 separate serial connections on-board. Each receiver card has one dedicated link, while one serial port is shared by both cards. Although each receiver card has its own processor (ARM-60), the dedicated serial links can be used by the receiver boards to share information. This allows for a distributed processing environment wherein the CPU's share the work load associated with the higher-level relative navigation and control algorithms.

Software: A key benefit of using the Mitel GPS

[2] Name should not to be confused with the mission name.

hardware is that this also provides access to the GPSBuilder Software. This software is entirely written in C code. Complete access to the code has allowed us to extensively modify the code and carrier tracking loops, the signal acquisition algorithms, the cycle-slip detection routines, the input/output routines, and the frequency search region during startup.

Testing: Several tests have been performed to determine the contribution to the differential carrier phase (DCP) measurement noise due to the clock synchronization method. The DCP was measured using a zero antenna baseline, and the receive antenna stationary and moving. A splitter was used to route the antenna signal to all four RF inputs on the receiver and integrated carrier phase was tracked for approximately two minutes at a 1 Hz data rate. Measurements were made of the integrated carrier phase from an array of indoor pseudolite transmitters in a high multipath environment. The results for the stationary case showed that the error in the DCP measurement between tracking channels on the same RF input is very small (STD 0.2 mm), and is within the resolution of the integrated carrier phase measurement. When DCP measurements are made using two RF front ends on the same card, the error grows to a STD of 1.5 mm. And for a DCP measurement between two RF front ends on separate cards, the STD of the measurement error is only 2.2 mm. These tests indicate that it is possible to make DCP measurements using multiple boards synchronized at the 10 MHz clock level and obtain errors that are comparable to single board receivers (such as the Trimble TANS Vector used on the formation flying testbed in Figure 1).

The results for the moving antenna case are comparable, but slightly higher (3-4 mm) than the stationary case. The error is slightly higher due to the presence of two cycle slips during the collection period. These slips occurred when the antenna was moved quite rapidly from side to side, but the cycle slip detection algorithm was able to detect and correct for the integer wavelength jump in the DCP.

This modified ORION receiver is currently in use on the 3D blimp testbed, and a similar device has been tested in a LEO environment on the GPS simulator. The results from these tests are very promising, and the indication is that, with some basic changes to improve the radiation tolerance of the receiver, this design should meet the needs of the Orion flight mission.

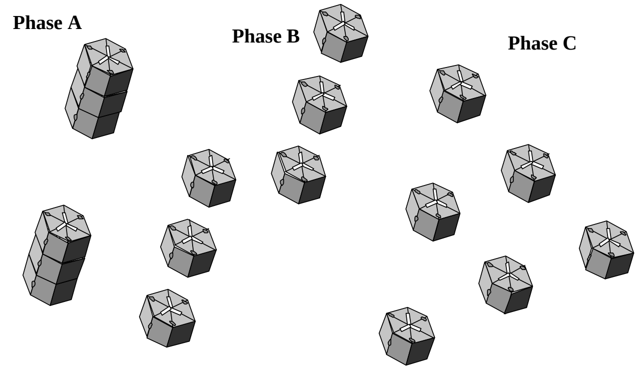

Fig. 3: Basic phases of the Orion flight mission.

5 Orion Mission Overview

The objective of the Orion mission is to demonstrate several key sensing and autonomous control technologies that are necessary to develop a *virtual spacecraft bus*. This will be accomplished using a distributed array of simple, but highly coordinated micro-satellites designed and built in-house. As illustrated in Figure 3, the current plan is to use a constellation of six satellites to demonstrate the relative ranging techniques. The satellites will be launched and deployed in one or two stacked sets (**A**). This configuration will be used to perform an initial reference calibration of the GPS receivers. The next step will be to split the stacks and perform coarse station keeping of the micro-satellites within each trio (**B**) (possible scenario: 1 km separation with tolerances of approximately 100 m). The vehicles will be controlled within an error box and 3-axis stabilized using feedback from the onboard GPS receiver.

When we have determined that the six satellites are functioning properly, the two groups will be combined into a single coarse formation. The next phase (**C**) will be to perform precise station keeping maneuvers for periods of approximately 1/2 an orbit (possible scenario: 100 m along track vehicle separation controlled to approximately $\pm$ 5 m tolerance along-track and radial). The real-time relative separation and attitude measurements will be validated using onboard cross checks between the six vehicles. A simple digital camera will be used to verify the pointing accuracy within the formation. More sophisticated real-time validation techniques, such as

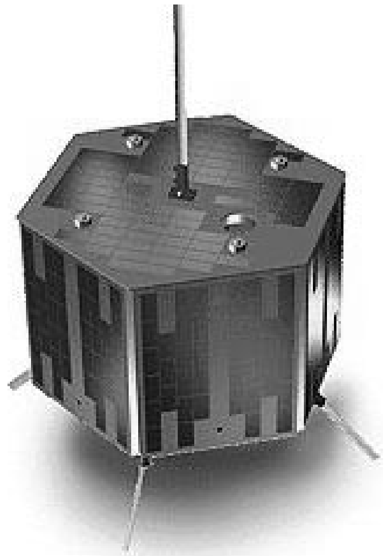

Fig. 4: SQUIRT Satellite

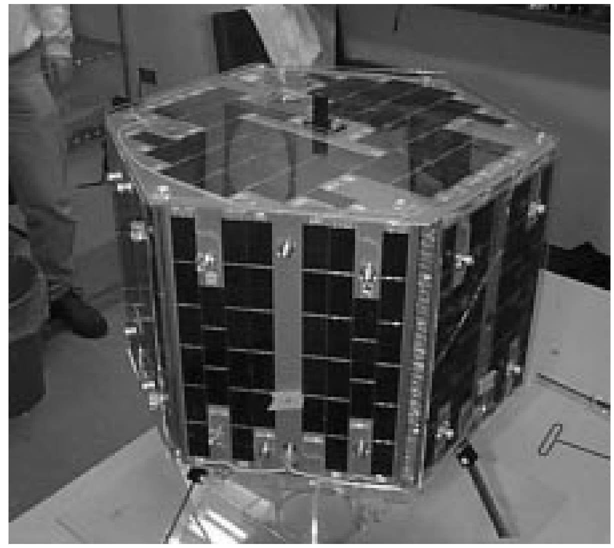

Fig. 5: Actual Sapphire satellite hardware.

laser ranging, will be included as permitted by the power and mass budgets. The primary means of validating the real-time measurements will be to store and then downlink the raw carrier phase and pseudorange data. Measurements will be taken on-orbit while selected ground stations have the same GPS constellation in view. Down-linked data will then be post-processed to validate the real-time measurements using techniques already demonstrated on the JPL TOPEX mission.

During all phases of the mission, the commands from the ground will specify maneuvers for the entire formation. Each satellite will then independently calculate the maneuvering commands required to perform relative separation and rendezvous operations. The onboard GPS receivers will serve as the primary means of orbit and attitude determination. Other, more traditional sensors will be included if possible under the power and mass budgets. However, the concept described above is expected to provide the lowest cost, lowest risk approach to demonstrate these vital technologies.

6 The Orion Micro-satellites

The Orion Spacecraft is being designed to support a technology demonstration of precision satellite constellation formation flying using GPS as the primary relative position determination. One of the major design goals of the spacecraft bus development is to use commercial-off-the-shelf parts to keep the cost low while still achieving a 6-12 month operational life.

To meet the mission goals, the micro-satellites will need both attitude control and station keeping ability. For autonomous operation, a high performance inter-satellite communications system would also be required. The major effort to date has focused on determining the general requirements for:

1. Size, weight and power capability of the bus structure,
2. Candidate C&DH processors,
3. Station keeping hardware,
4. Attitude control system hardware,
5. Inter-satellite communications, and
6. Inter-subsystem communications.

These characteristics represent new additions (or major changes) to the previous SSDL SQUIRT designs, examples of which are shown in Figures 4 and 5.

Shape, size, weight and power: A cube was chosen as the basic shape of the bus structure for optimum utilization of the surface area and the internal volume. The initial size was chosen as a 0.5 m cube. This size spacecraft would produce about 30 W average power (assumes the spacecraft has attitude control, 16% efficient body mounted solar cells, is in a LEO orbit with a 35% eclipse time). This 30 W average was chosen as the maximum power budget for continuous operation, knowing that GaAs solar cells with up to 21% efficiency could be added at a later date. The design effort is to use less than 30 W such that lower cost Si cells could be used. A initial target weight of 40 kg was selected based on the estimate of the components required.

Previous experience from the SSDL SQUIRT program indicated that the structure should be fabricated from honeycomb sheets with a stacked tray arrangement. This method reduces the use of costly, precision and intricate CNC machined parts. The approach is also modular, which is essential for a rapid prototyping development.

Candidate C&DH processors: Space rated processors are not feasible for this mission as they are typically too expensive and require too much power. The Intel 386 radiation hardened processors and versions of the PowerPC were initially evaluated for size, processor speed, availability on commercial single board computer (SBC), power consumption and support software. However, most of these computers are not available in low power versions on SBCs. Thus, to maintain compatibility with ongoing development efforts at the Goddard Space Flight Center the *StrongARM*™ processor was selected as the baseline C&DH computer for this mission. The key advantages of this computer are that it is available in several versions, it offers very high processing speed, and has a low power consumption. The StrongARM has another advantage in that a version of the Linux operating system is available for it. The high level of current industrial interest in this computer strongly suggests that SBC's will be available for Orion.

The current plan is to use a modified version of the AutoCon™ flight control software, which was developed by NASA GSFC to perform autonomous control on E0-1 [1]. The AutoCon control architecture uses an innovative mix of fuzzy logic and natural language to resolve multiple conflicting constraints and autonomously plan, execute, and calibrate routine spacecraft orbit maneuvers. A development StrongARM computer board will be used to evaluate the AutoCon flight control software for the Orion mission.

Station keeping hardware: There are several different types of thruster systems that are capable of providing the station keeping for Orion. However, for simplicity, we did not consider mono- or bi-propellant systems that use highly reactive fluids or gases. These thrusters pose a serious danger and cannot realistically be designed by students.

A non-volatile compressed gas (Nitrogen) system was chosen for Orion. The design followed two recent examples on the NASA Safer system and AERCAM under development at JSC. The tank volume and pressures are being determined with a propulsion simulation that allows a trade study of the various mission scenarios. Commercial valves, nozzles and regulators have been evaluated and some initial valves ordered. Generally the limiting factor in this selection is the availability of commercial small, low-power valves.

Several issues will have to be addressed during the development of the station keeping subsystem, including the system simplicity, the mission lifetime and requirements, the power required, and the cost. Simplicity is key if the system is to be developed inhouse. To satisfy the power issue, low-power vacuum rated valves will be investigated. Mission lifetime requirements will be addressed by selecting the nozzle size and maximizing the fuel storage. To avoid developing a very high pressure fuel storage tank, the current baseline is to use manned-flight-qualified storage and high-pressure systems similar to those on SAFER and AERCAM. This high pressure device will then be connected to a low-pressure system built by SSDL. The thruster system will also have a micro-controller that receives high-level commands such as the thrust vector, level, and duration from the C&DH, and then activates the appropriate valves. This decoupling between the thruster system from the rest of satellite is aimed to accelerate the development process.

To perform the station keeping, the baseline plan is to use 0.05 N thrusters. This thrust level can be scaled to trade-off mission life with maneuvering time. Work continues on evaluating this trade-off using the linearized relative orbital dynamics (Hill's equations). Further work is required to finalize this analysis, but the current studies indicate that 6 kg of fuel will provide a useful mission life of 4-6 months.

Attitude control system hardware: A zero bias momentum wheel or three reaction control wheels will be required to perform the attitude control. Note that these wheels are typically not available for space missions for less than \$100K per device. Thus the current plan for Orion is to use vacuum rated, brushless DC motors to build our own reaction wheels.

For the baseline satellite (mass 40 kg, 0.5m cube) in a 500-600 km altitude orbit, the Earth magnetic field derives external torque of 10^{-5} N-m for a residual dipole of 1 Amp-m^2, which is 10 times and 100 times stronger than aerodynamic and solar torque, respectively. If we assume the maximum slew rate as 2 degree/sec, the corresponding maneuvering torque is 3×10^{-3} N-m. Adding a safety factor to account for the uncertainty in the mass distribution of the satellite, the torque of a motor in the reaction wheel system should be approximately 5×10^{-3} N-m. If the wheel dumps angular momentum three times a day, the amount of momentum storage should then be approximately 0.1 N-m-sec. These specifications can be achieved using a very small flywheel with a moment of inertia of 2.2×10^{-4} kg-m^2 combined with a 5000 RPM motor.

Inter-satellite communications: The main requirement for this subsystem is to provide communications between the spacecraft in the constellation. Again, there are many commercial systems that can be used to link computers in a network without being physically connected. Wireless Ethernet is an ideal choice, since it can support the high data rates required. This solution would also take advantage of spread-spectrum technology. Because the transmitter power will most likely be under 1 W, we will not have to obtain FCC approval to use it in space. The Lucent Technologies WaveLAN PCMCIA wireless Ethernet card looks promising for Orion because it fits into a PC card slot that is standard on all current model laptop computers, and it also exists on the StrongARM evaluation board. Another reason for the WaveLAN is the availability of source code, and wide operating system support.

Inter-subsystem communications: In designing the inter-subsystem communication bus, the objective was to develop a system that meets the high performance needs of Orion, but is flexible enough to be used on future SQUIRTs. An additional objective was to reduce the number of wires required and to allow subsystems to be easily added or removed. These goals narrowed the choices to synchronous serial designs, which allow fast data transfer over only 2 or 3 wires. Among the industry standard protocols, the Serial Peripheral Interface (SPI) was attractive for its simplicity and wide device support, but requires an external address bus, which limits the number of subsystems for a given number of address lines. To allow for more expansion without adding more lines, one address will be reserved for communication using the Inter Integrated Circuit (I2C) protocol, which sends addresses over the data lines. Adding the explicit ability for multiple devices to initiate SPI communications (Multi-mastering) provided a protocol that meets or exceeds all of our requirements, the SPI-MM/I2C. With 4 address lines, 2 data lines, ground, and an additional handshaking line everything fits in a DB9 connector.

Summary: The emphasis of this initial work has been to investigate the various alternatives for the main subsystems of the satellite. The main result is a set of tradeoffs that can be analyzed to compare the key features, namely performance and power. The final decisions for each subsystem will be made as the mission development proceeds. However, these initial studies indicate that the desired Orion mission can be achieved using micro-satellites that meet the target goals of mass, power, and cost.

Conclusions

This paper has outlined a new GPS based relative position and attitude sensing mission called Orion. Using a fleet of 6 low-cost micro-satellites designed in-house, Orion will investigate a variety of GPS sensing and autonomous control related issues. A successful Orion mission will complete a key step towards formation flying and virtual platform capabilities on future Earth Science Enterprise missions.

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