Formation Flying Activities and Capabilities at Ball Aerospace[1]

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Abstract - Many future missions will be implemented with distributed spacecraft systems, which require formation flying capability. Here, formation flying means controlling the motion of one or more spacecraft relative to other nearby spacecraft to enable them to operate as a distributed sensor, in orbit about a planet or in deep space, at inter-spacecraft ranges from meters to many kilometers. This paper discusses BATC experience, modeling and study work in formation flying. We discuss our deep space formation flying work (StarLight, Terrestrial Planet Finder, and MAXIM Pathfinder), low Earth orbit formation flying work (CloudSat), and rendezvous and docking work (Deep Impact and Orbital Express).

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1. INTRODUCTION

Many future NASA missions will be implemented on distributed spacecraft systems, which require formation flying capability. The formation flying performance of these missions ranges from the loose formation control needs of the CloudSat-Aqua-Calipso and 00 CLUSTER Reflight formations to the tight formation control and maintenance requirements for the Terrestrial Planet Finder (TPF), DARWIN, LISA and MAXIM missions further into the future. Other formation flying missions include Mars Sample Return (MSR), Magnetospheric Constellation (MC), Magnetospheric Multiscale (MMS) mission, Geospace Electrodynamics Connections (GEC), Radiation Belt Mappers (RBM), and Ionospheric Mappers (IM). Other agencies such as the Department of Defense (DOD), Department of Energy (DOE) and commercial remote sensing companies are also interested in the same technology, particularly as applied to large sparse aperture systems for Earth observation. For example, the DOD is currently developing TechSat 21 which is a collaborative cluster of spacecraft for a radar mission and Orbital Express (OE) to demonstrate rendezvous and docking. They are also investigating the use of formation flying technology for communications, radiometry, and "on demand" coverage missions.

In this paper, the term "formation flying" is used to represent each of the following cases:

(i) a group of spatially distributed spacecraft acting together to form a single sensor,
(ii) the task of maintaining a satellite's position or state with respect to other satellites (e.g., maintaining a long baseline interferometer),
(iii) a satellite's active maneuver capability with respect to other satellites (e.g., circumnavigating satellites), and
(iv) on-orbit rendezvous and docking between two space vehicles.

Implementation of formation flying poses many challenges. The performance requirements and disturbance/operating environments vary substantially between orbiting and deep space missions. For orbital missions, orbital and environmental dynamics (J_2, drag, ground track coverage requirements, etc.) affect each spacecraft in a formation differently. These variations affect the formation stability and have to be accounted for to ensure the performance specifications can be met. For deep space formation flying missions, which typically require much tighter formation control capabilities (TPF, DARWIN, MAXIM, LISA), solar radiation pressure is the chief disturbance source. The precision formation keeping requirements of these missions (less than a centimeter in position control and an arcminute in bearing) must be met over inter-spacecraft separation distances of meters to many kilometers.

There are several major formation flying maneuver types: formation acquisition and initialization, formation maintenance, formation resizing, formation rotation, formation retargeting slews, stationkeeping, rendezvous and docking, and other close proximity maneuvers. For example, normal mode operations for a deep space interferometric mission include closed-loop formation maintenance in the form of deadband or linear position and

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attitude control in the presence of solar radiation pressure disturbances. Another type is formation retargeting via range and bearing angle maneuvers. Formation resizing is done to change interferometer baselines and, finally, formation rotation is accomplished to move to new baseline orientations around the line of sight.

This paper discusses BATC experience, modeling and study work in formation flying. We will discuss these topics in terms of our deep space formation flying work (StarLight, TPF, MAXIM Pathfinder), low-Earth orbit formation flying work (CloudSat), and rendezvous and docking work (Deep Impact (DI) and OE). We cover system and subsystem engineering (concept of operations, operational modes, tiered formation acquisition sequencing, fault protection and formation/vehicle safing), hardware trades and needs (range and bearing measurement and assessment, propulsion needs and propellant budgets, inter-spacecraft communications, sensors and actuators), estimation and control algorithms, simulations, flight software, and integrated modeling.

2. FORMATION FLYING STUDIES AND MISSIONS

Ball is working on several missions and studies which involve formation flying. The missions include CloudSat (with Calipso), OE, and DI with focused studies on StarLight and TPF.

CloudSat

CloudSat is scheduled to launch in April 2004 with Calipso on a Delta II 7420-10C launch vehicle and will make detailed three-dimensional maps of cloud formations in an effort to better understand weather patterns. CloudSat will formation fly with four other satellites; Aqua, Calipso, Parasol and Aura. During the 30 day commissioning period, CloudSat will maneuver itself into its desired location relative to Aqua and Calipso. Given that CloudSat's phasing relative to Aqua will be unknown prior to launch, it will be required to begin adjusting its orbital parameters anywhere from 3 to 16 days following insertion. It is expected that CloudSat will perform several orbital adjustments that will adjust the orbit's semi-major axis, its inclination, its eccentricity, the longitude of the ascending node and its phasing within the orbit.

Aqua and Calipso's orbits are tied to the worldwide reference system (WRS) and have error boxes associated with their orbits. The overall mission requirements are written such that Calipso is required to be no greater than 2 min behind Aqua. Both of these vehicles orbits are controlled relative to a desired location over the ground track as a function of time and do not directly control their orbits with respect to each other. Both Aqua and Calipso are allowed to drift up to ±22.5 s from the desired location within each of their orbits in the along track direction. CloudSat's orbit will be controlled relative to Calipso to 15 ±2 s, and will be anywhere from 15 to 105 s behind Aqua. Their cross-track control is required to be within ±20 km with respect to the WRS. Figure 1 shows a representation of the relative orbital locations of CloudSat, Aqua, and Calipso.[1] Parasol and Aura follow behind Calipso by 22.5 s and ~15 min, respectively.

Aqua and Calipso are required to remain within error boxes around their desired orbital locations similar to the way EO-1 formation flew with respect to LandSat 7 [2]. Both vehicles will have circulation orbits within each of the error boxes, and will be required to boost their orbits once every 2 or 3 months. Following that, their orbital altitude will decay slowly due to atmospheric drag. As the orbit decreases, the relative location of each vehicle will fall backwards within each orbital error box until the semi-major axis of the vehicle's orbit is less than that of the ideal orbit. At that time, the vehicle will move forward and below its ideal location within the box. After the semi-major axis of each vehicle's orbit has decayed to a certain point, a semi-major axis-increasing maneuver will be performed that will bring each spacecraft to near the leading corner on the upper side of their orbital error box. Since CloudSat's allowable error box is much tighter than the boxes for Aqua or Calipso, it is expected that its orbital adjustments will occur approximately 10 times as often as for Calipso or Aqua. In addition, if Calipso's average area to mass ratio is higher than CloudSat's, CloudSat may also be required to perform orbit-lowering maneuvers to maintain its relative orbit with Calipso. [1]

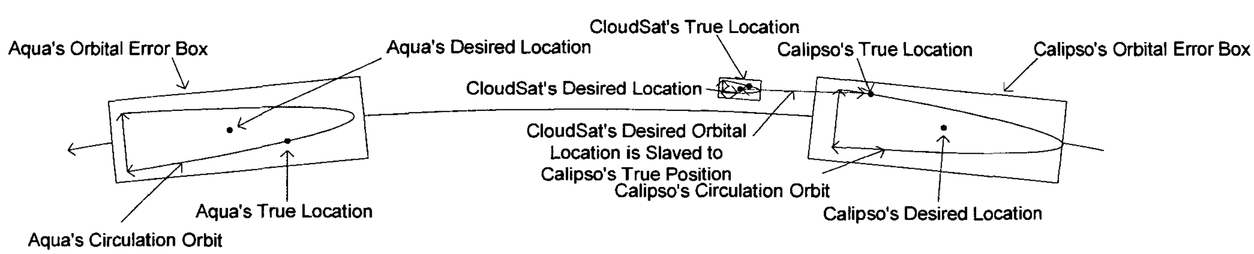

Figure 1. Pictorial representation of the orbits of CloudSat, Aqua and Calipso.

Given CloudSat's orbital accuracy requirements, it will be required to perform orbit raising, lowering or inclination changing maneuvers approximately once per week. Since these maneuvers are relatively infrequent, they will be designed and executed via ground commands. Table 1 shows a list of the requirements that are relevant to CloudSat formation flying with Calipso. CloudSat will perform ΔV's in the open-loop or closed-loop mode depending upon what level of ΔV is required. Open loop maneuvers are performed when small, highly accurate ΔV's are required. In this case, a set of the vehicle's thrusters are commanded on for a specific number of seconds while the reaction wheels are used to absorb any excess momentum that builds up due to off-axis thrust components. Off-axis thrust results from thrust vectors that do not pass directly through the vehicle's center of gravity. When larger ΔV's are required, the thrusters will be fired in a closed-loop mode. Here, the specific duty cycles for each of the thrusters is continuously computed, so the thrusters are used for both controlling the vehicle's velocity and its attitude which will minimize the accumulated momentum. Figure 2 shows CloudSat's expected ΔV accuracy as a function of the commanded ΔV and the level of desired ΔV.

Table 1. Relevant requirements to CloudSat formation flying with Calipso.

Requirement	Value	Performance
ΔV direction accuracy	6 deg (3σ)	6 deg (3σ)
ΔV range	1 m/s - 50 m/s	1 m/s – 50 m/s
ΔV range	0.5 cm/s to 20 cm/s	0.5 cm/s to 20 cm/s
ΔV accuracy	6% (3σ)	3.2% (3σ)

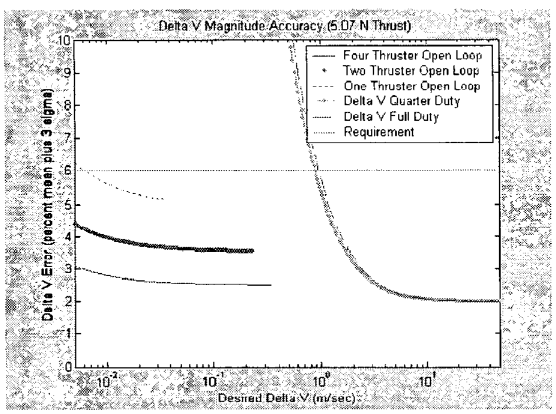

Figure 2. CloudSat's ΔV accuracy as a function of the desired ΔV.

Orbital Express

The Orbital Express mission, being developed by the Defense Advanced Research Programs Agency (DARPA), is designed to have two spacecraft rendezvous in LEO, and for the "active" spacecraft to transfer commodities such as batteries, propellant, or spacecraft components to the "passive" spacecraft. The active spacecraft could also be used to transfer the passive spacecraft to a high-energy orbit to preserve fuel on the passive spacecraft. Formation flying and rendezvous and docking are the key attributes of the mission.

BATC is part of Boeing's Team on the DARPA Orbital Express program. The Orbital Express program will demonstrate autonomous on orbit servicing in 2006. This demonstration will employ the ASTRO servicing spacecraft (built by Boeing) and the NEXTSat spacecraft, Fig. 3, (built by BATC). The NEXTSat spacecraft represents future spacecraft that will utilize servicing capabilities. While the bulk of the servicing requirements fall onto the ASTRO spacecraft, conventional spacecraft that require servicing will require some modifications, including capture mechanisms, fluid couplers, power couplers and a capture sensor. These minimal modifications will allow future spacecraft to focus on their primary mission, while adding some limited features that will allow autonomous on-orbit servicing by other spacecraft.

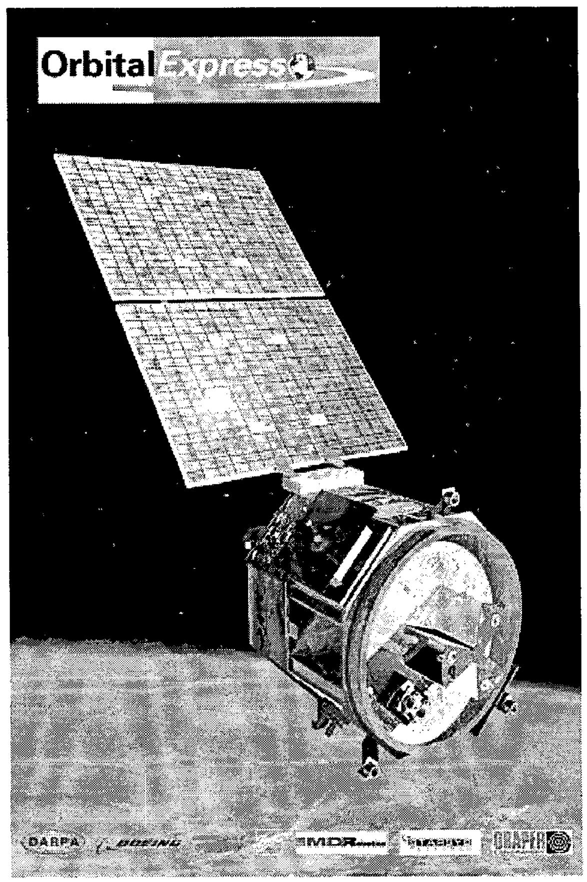

Figure 3. BATC NEXTSat Spacecraft

Deep Impact

The NASA Deep Impact Discovery Mission is being developed to determine the materials, structure and properties of a comet's nucleus [3,4]. The two-spacecraft mission is designed to probe comet Tempel 1 by flying an instrumented Impactor spacecraft into the comet nucleus and looking at the ejected material and crater formation with an instrumented Flyby spacecraft. As the Impactor spacecraft approaches the comet, it transmits data to the Flyby spacecraft via an inter-satellite communications (ISC) link. BATC is providing both the Impactor and Flyby spacecraft along with the instrument suites.

The two DI spacecraft fly to the comet mated together. During this phase, the Flyby spacecraft is tasked with setting up an "energetic rendezvous" with the comet nucleus. The two spacecraft separate one day before the encounter at which point the Flyby and Impactor spacecraft can be considered to be 'formation flying' with respect to the comet while cross-linked with each other. The Impactor follows a collision trajectory to the comet while the Flyby spacecraft maneuvers to fly past the comet and impact event. This process is complicated by a closing velocity with the comet of over 10 km/s.

The encounter phase begins when the two spacecraft separate. Following separation, the Flyby spacecraft performs a trajectory control maneuver to slow itself down relative to the Impactor spacecraft. This maneuver also includes a small cross-track velocity component (divert maneuver) to affect a 500 km separation between the Flyby spacecraft and comet at closest approach. During this phase, the ISC coupled with the DS-1-derived AutoNav system[5], ensures that the impactor can autonomously perform any needed course corrections to assure impact at the comet's center of brightness. The ISC has the capability to transmit close-up images of the comet nucleus surface prior to impact enabling prediction of the impact point and contingency commanding to the Impactor. The closest approach occurs ~15 minutes after the Impactor spacecraft strikes the comet nucleus.

StarLight

The NASA StarLight Mission (a two spacecraft optical interferometer) has been described in detail elsewhere [6-10]. Here we will focus on the formation flying aspects of StarLight including; propellant needs, tiered formation acquisition, formation sensors, and fault protection. The ESA SMART-2 mission, currently being defined, may also demonstrate separated spacecraft interferometry [11].

Formation flying in the Starlight program requires the coordinated, precision flying of two spacecraft, Fig. 4, with separations ranging from 40 to 1000 m, in an Earth trailing Solar orbit, to perform interferometry on distant starlight. This system architecture can potentially detect distant orbiting planetary bodies and has been studied as a precursor option to the upcoming TPF mission. The two spacecraft are required to hold a specific formation to within centimeters in range and arc-minutes in bearing. Instrument steering mirrors and adjustable path delays effectively take the formation to an accuracy of nanometers and milliarcseconds. Sixteen nitrogen cold gas thrusters, providing about 7 mN each, are configured to provide fully redundant and decoupled translation and rotation capabilities on each spacecraft. Small reaction wheels are used for rapid reorientation maneuvers and when high-performance low-jitter formation flying is not required. Quaternion-out Ball star trackers are used for absolute inertial attitude on each spacecraft. JPL's Autonomous Formation Flying (AFF) sensor [9], a GPS technology-based proximity sensor, is used to determine partner range and bearing to centimeters and arcminutes.

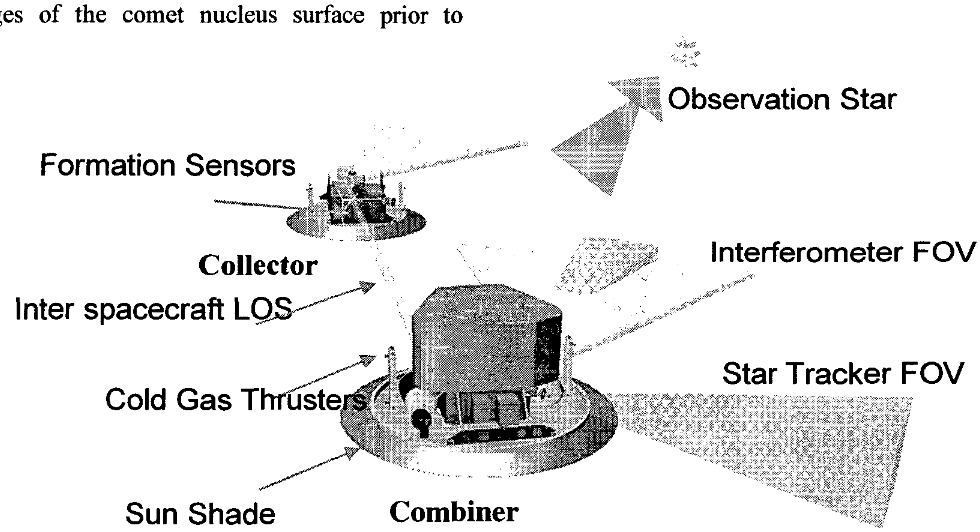

Figure 4. StarLight Separated Spacecraft Interferometry Demonstration.

Propulsion Analysis and Needs -- More specifically, the StarLight Project was being developed to perform roughly 2500 formation maneuvers during a nominal one-year mission. Formation translation maneuvers use reaction control thrusters (RCS) thrusters for changing the interferometer baseline and acquiring different target stars. Propellant consumption for formation translation maneuvers is derived from a prescribed set of nominal maneuver distances and allocated maneuver completion times. The cold gas propulsion system requires 51.6 kg of GN_2 distributed across both space vehicles to perform the nominal one-year mission [7,8,12]. The entire prescribed formation translation maneuver set (~2500) is completed in roughly 3039 hours and consumes 89% of the total required propellant. A sensitivity analysis, shown in Fig. 5 [12], indicates that extending maneuver completion time by 25% reduces propellant consumption by 20%, whereas requiring 25% shorter maneuver completion times increases propellant consumption by 37%.

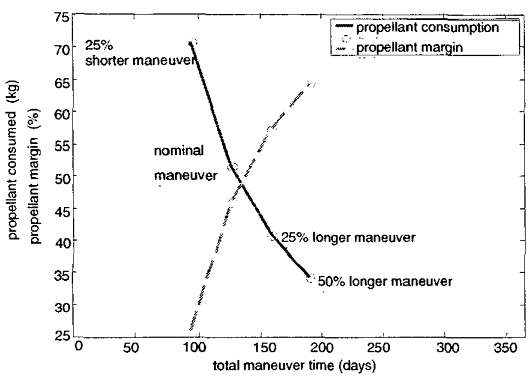

Figure 5. StarLight Propellant Consumption and Margin as a Function of Maneuver Time [12].

An order of magnitude propellant reduction can be realized if one replaces the 7.5-mN, 60-s specific impulse, cold-gas propulsion system with a 3.0-mN, 600-s specific impulse pulsed plasma thruster (PPT) system. The nominal mission's required propellant mass decreases from 51.6 kg to 5.6 kg. Moreover, the entire prescribed set of formation translation maneuvers can be executed in the same 3039-hour time allotment. By trading longer maneuver times (lower thrust) for lower propellant mass (higher specific impulse), the propulsion analysis can specify PPT performance requirements that optimize spacecraft design.

Move analysis and fuel use for formation interferometry - Once the formation flying demonstration is complete, StarLight was to examine various "science" targets (stars). In order to measure the brightness distribution (angular size and structure) of a science target, StarLight planned to make fringe visibility measurements at a number of *u-v* points.* Moving between *u-v* points entails physical motion of the two spacecraft - changing their vector separation. The number of such moves (and therefore the number of science targets which could be observed) would then be limited by the mission duration and schedule, and by the available fuel. Tables 2 and 3 summarize the parameters and assumptions used in these calculations to determine how many science targets could be observed.

Table 2. StarLight Spacecraft Parameters.

Item	Value
Spacecraft masses	234 kg collector; 350 kg combiner
Available thrust per spacecraft	9 mN (two 4.5 mN thrusters, working as a pair)
Cold Gas Isp	65 s

Cold gas thrusters have a fairly low specific impulse, so that there would not be sufficient fuel to operate at full thrust during all the moves without large tanks. Fuel can be conserved by utilizing a combination of thrusting and coasting. The most efficient use of fuel is to use maximum thrust for the initial acceleration and final deceleration, with coasting in between. The free parameter in the optimization is the fraction of each move in which the acceleration and deceleration phases take place.

During the separated spacecraft interferometry phase of the mission, with constraints of time (t) and total fuel (f), the number of targets (N) that can be observed is:

$$N = 0.77\, f_{kg} \sqrt{\frac{4.9\, t_{days}}{f_{kg} - 1}}$$

After accounting for time allowances for initial checkout, communications with the ground, the formation flying demonstration tests, and fault recovery, ~1400 hours would be available for spacecraft moves and observations. The available fuel, after allocation of margin, for the formation interferometry phase, was 20 kg. We can therefore derive a total of 43 sources that could be observed during the interferometry portion of the mission.

Tiered formation acquisition - Formation acquisition for interferometry uses a tiered approach – from very coarse global performance to very high precision performance. Once the AFF is turned on, distances between RF transmit and receive antennas are automatically computed [9]. If not pointed face-to-face, signals from posterior antennas provide coarse bearing sufficient to reorient the vehicles. Once face-to-face, the single transmit and three receive antennas on each vehicle provide accurate range and bearing

* The *u-v* plane concept is useful for interferometry. Positions *u* and *v* are given by the baseline vector (from one aperture to the other) projected into the plane perpendicular to the line of sight to the star.

Table 3. Interferometry *u-v* Coverage Types.

Coverage Type	Percent of Observation Time	Moves
Simple 1D	20%	39, 68, 87, 136, 168, 199, and 279 m
Enhanced 1D	20%	Simple 1D coverage + two moves of 70 m
Simple 2D	40%	Enhanced 1D coverage at two epochs
Full 2D	20%	Two epochs, each with two moves of (39, 68, 87, 136, 168, 199, and 279 m) & three moves of 52 m
Calibrator stars are only observed at the beginning and end of the *u-v* tracks. The typical angular separation between science targets is 20°.		

measurements. The system is calibrated and has the highest accuracy near the face-to-face orientation used during interferometry. Once aligned to a fraction of a degree using the AFF sensor, laser metrology [10] is engaged to obtain even more precise formation knowledge. Both relative distance and absolute bearing angles are obtained at this phase. The final stage of formation determination and control uses the observation starlight and the interferometer to scan back and forth to search for fringes. During maneuvers, between science observations, when face-to-face operations are temporarily lost, an onboard Kalman filter is used to propagate and maximize the quality of formation knowledge using all available measurements

Formation Flying Sensors - StarLight's baseline relative navigation sensor is the JPL-developed AFF sensor [9], which measures range and bearing angles between the formation's two vehicles. BATC investigated the viability of supplementing StarLight's AFF sensor with an alternative means of measuring bearing angles by using a BATC star tracker to detect an LED on the companion vehicle. The feasibility study concluded that a slightly modified Ball CT-633 star tracker could be used to sense an LED (of a 625-nm wavelength) located on the companion vehicle and derive bearing-to-target angles from the measurements. Necessary modifications to the star tracker include incorporation of a narrow-band optical filter tuned to the LED frequency and decreasing the CCD integration time. In addition, the LED power must be reduced by a factor of ten when the vehicles are within a 150-meter range, Fig. 6.

RF coarse range and bearing - Coarse range and bearing determination can be made in a formation flying mission through a multi-frequency inter-spacecraft communications (ISC) link to calculate range to an error of less than 2% at any separation and bearing error to less than 6 degrees off angle. Utilizing simple telemetry (such as temperature, detected transmitter power level and received signal strength) along with knowledge of system losses and antenna off-boresight gain, the techniques for calculating space loss and equivalent isotropic radiated power (EIRP) can be used to calculate the spacecraft separation. This technique may be particularly useful in the fault protection hierarchy.

The StarLight UHF ISC link can easily be implemented as a coarse RF range and bearing sensor for any spacecraft geometry by the simple use of the received signal strength intensity telemetry. For example, consider the scenario shown in Fig. 7. This is the simple case of a stationary Combiner spacecraft with the Collector spacecraft rotating in a counter-clockwise direction. Assuming one spacecraft is held constant and the second spacecraft is moving, the specific geometry between the two spacecraft remains constant. Here, spacecraft separation, or perpendicular range, remains constant. The only true variable in this case, or in most any case of spacecraft separation and orientation, is the antenna off-boresight angle. The UHF transceivers on the Combiner and Collector operate at slightly different frequencies, which is an advantage for range determination. The antennas on both spacecraft are identical in type, but have slightly different patterns, as shown in Figure 8, when operating at the different frequencies, which help in the bearing determination.

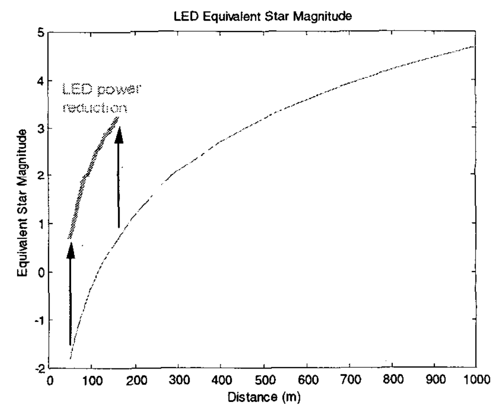

Figure 6. LED's Equivalent Stellar Magnitude Versus Formation Range.

Received signal strength telemetry is the summation of transmit spacecraft EIRP, space loss, receive antenna gain (with pointing loss) and circuit loss to the receiver. Under static or motionless spacecraft conditions, all of the previously mentioned parameters are constant. If the parameters of EIRP, receive antenna gain, and circuit loss to the receiver are removed from the value of the received signal strength intensity telemetry, space loss is determined, from which range can be determined. With two different frequencies, there are two values for space loss, but only one value for range.

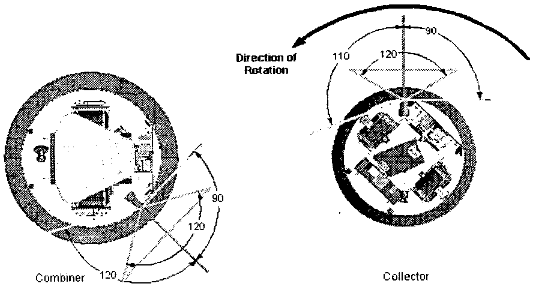

Figure 7. Initial Position Boresight Angles.

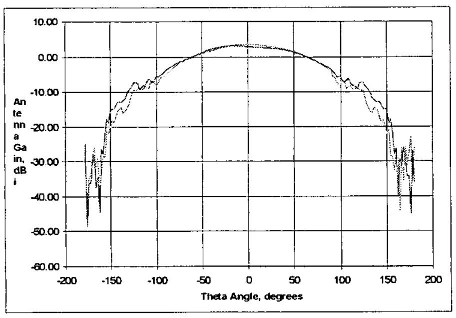

Figure 8. UHF Antenna Patterns.

When one spacecraft is in motion, the same philosophy can be used for removing system constants. The only difference is that an iteration of antenna loss over off-boresight angle must be performed, and this must be performed for both spacecraft. The beauty of this system is that while iterating through the off-boresight angle, or antenna pattern of both spacecraft, range calculations on both spacecraft will not agree until the proper off-boresight angle is found. If a standard 2° x 2° antenna radiation distribution pattern (RDP) is measured, spacecraft separation can be determined to within 1.8% error at any range and bearing to within 6° using the ISC.

Fault Protection / Formation Safing - Starlight uses a mixture of fault avoidance, fault tolerance and hierarchical fault protection to guarantee functionality in the formation flying system. This is done autonomously given the potentially infrequent communications of a deep space mission. Fuel, thermal margins and power are guarded closely by the fault protection system.

First, the system is designed to operate within regions where a fault would have minimal impact on the system. For example, spacecraft are never given relative velocities which, in the event of a processor reset, could result in a collision. Second, the system is designed to be robust. Components with credible failure scenarios are made redundant. Attitude and formation estimation algorithms are designed to operate through sensor outages without loss of required performance. The estimators propagate high accuracy system state values based on the most recent measurements, spacecraft and thruster dynamics models and solar disturbance models, when the AFF or star trackers are temporarily offline.

The autonomous fault protection system attempts to maintain the highest level of function possible at all times, and not to just transition to a last-resort individual spacecraft safe mode which waits for the ground to clear the problem. The formation must be safed when possible. It's critical not to collide, and cause damage, or to drift too far apart, which might require ground-tracking intervention and waste valuable mission time and fuel. The first level of fault protection presumes the instrument has a fault and the fault has persisted for a while. The bus, AFF, trackers and ISC are still functioning. The centralized master-slave formation commander brings the formation to a relative stop and continuously performs onboard diagnostics and recovery algorithms. If instrument function is recovered, reorientation of the formation is initiated and science data collection continues. If not the system is safe and waits for ground intervention.

At the same fault protection level, if the system finds itself separated by much greater than 1 km (up to 10 km outside of its normal operating range), fault protection works to bring the two spacecraft back together in a controlled fashion. Both the AFF and ISC are designed to work out to 10 km providing formation knowledge and coordinated activities. Beyond 10 km, ground tracking, orbit estimation and manual control are required.

At the next level of fault protection, only one spacecraft is capable of sensing or maneuvering with respect to the other spacecraft. The other craft is deemed uncooperative. The spacecraft that can, moves to a point near the disabled spacecraft and holds position until the fault clears or the ground gets involved. Finally, when neither spacecraft can intelligently affect the formation, relative motion is brought to a perceived stop based upon propagating range and bearing estimates (state filters). Each spacecraft knows what to do when it loses contact. Independent safe modes are activated to ensure power and thermal margins. Fuel consumption is minimized to momentum dumping and even momentum dumping is reduced to nearly zero through active use of solar torques. Onboard diagnostics and tests are performed until the ground gets involved.

Terrestrial Planet Finder

Terrestrial Planet Finder [13] (TPF) is a NASA mission concept, now in pre-phase A, which will directly detect the light from planets orbiting nearby stars, and analyze that light for signs of biochemistry. Launch is anticipated around 2015. The European Space Agency (ESA) is considering a mission called Darwin [11,14] with the same objective. The nominally five-year TPF mission will perform a search for Earth-size planets in the habitable zone around approximately 150 solar-type stars (spectral types F, G, and

K). The habitable zone is defined as the range of distances from a star where liquid water could be found on a planet's surface. In order to obtain a suitable sample of 150 stars where an Earth-size planet may be found, TPF will need to make observations out to a distance of up to 20 parsecs.

Until recently, NASA has envisioned TPF as a nulling interferometer comprising four or more telescopes in a cryogenic environment (~40K), either on a large deployable structure or on a set of formation-flying spacecraft. ESA has considered several arrangements of the telescopes for Darwin, all based on formation flying. Recent NASA studies (including one by a Ball-lead team) have prompted earnest study of single-telescope coronagraphs as well as smaller versions of the structurally-connected interferometer (with reduced science capability). While acknowledging this new uncertainty about formation flying for TPF, we will describe the features of the reference formation flying design, Fig 9 [13].

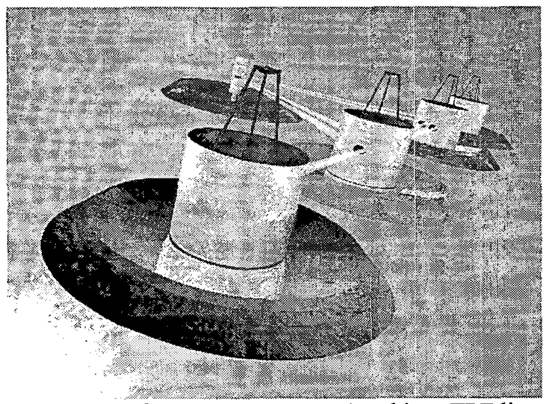

Figure 9. The four-Collector, one-Combiner TPF linear array concept in operation [13].

All formation flying-based designs for TPF and DARWIN are confined to a plane because of thermal constraints: the cryogenic optics of one spacecraft must not be exposed directly to the warm sunlit surfaces of another. Still, the formation of telescopes may be arranged either in a line or in a two-dimensional pattern. For each star, the telescopes must rotate through some range of angles around the line of sight and observe at many orientations along the way; for a linear array, the range must be about 180°, whereas for a hexagonal array, the range is only 60°. For the baseline interferometric TPF architecture [13], these observations will use maximum baseline lengths of 75-200 m. In addition, astrophysical observations of 750-1000 objects will occupy approximately 50% of the total observing time, and will use baseline lengths up to 1000 m.

Move analysis and fuel requirements -- For planet search observing mode, the baseline lengths will be kept constant, and the array rotated about the line of sight to the target star. Each spacecraft will follow circular or polygonal paths enabling the formation geometry to remain constant through the formation size changes. For circular motion with a fixed radius of curvature, the acceleration varies as the inverse square of the rotation period, with the total velocity change varying inversely with the period. The nominal plan for TPF is a rotation period of eight hours, a compromise between fuel use and observing efficiency (for stars at a distance of 10 pc, the required total integration time may be only two hours). For an eight hour period and a typical maximum baseline of 150 m, the acceleration is 3.6×10^{-6} m/s^2 for the outer two collector spacecraft (note that each of the outer collectors is 75 m from the center of the circular trajectory). The acceleration for the inner two collectors is each one third as large. The actual trajectory may be approximated by a many-sided regular polygon, with relatively brief thruster firings interlaced with ballistic coasts, in order to give quiet periods for observing. However, the average acceleration will be nearly the same as for a circular trajectory.

The science program in Ref. 13 included multiple observations of 150 stars. In order to confirm detections, and to account for phases in each planet's orbit when it may be too close to the star to be detected, we assume multiple observations per star. Each of these will involve a 180°-baseline rotation in 2 to 6 hours. There will be some follow-up observations, involving spectroscopic measurements. However, these observations will be conducted only for a relatively few stars, and will involve such long integration times so that the fuel requirements will be minor. There will also be much longer baseline maneuvers for astrophysical observations. This maneuver set can be broken up many ways. Here, we base our calculations on the models originally developed for our Starlight work [12] and a derived total impulse requirement of approximately 80,000 Ns.

If we assume a nitrogen cold-gas propulsion system, with a specific impulse of 60 s, the propellant mass to execute this maneuver set would be approximately 146 Kg per spacecraft (1000 Kg) in the formation. This gas would require a very large volume for storage. Further, leakage concerns over a five year mission duration would drive the mass and volume even higher. Therefore, cold gas appears impractical and electric systems (PPT, FEEP or colloid thrusters) with much higher specific impulses must be considered. For example, the propellant needed by a coaxial PPT-based system operating at 600 s specific impulse (derived from the PPT system described in Ref [12]) for each 1000 Kg TPF spacecraft is ~14 Kg.

Precision formation requirements for interferometry -- Broadband optical/infrared interferometry requires that the optical paths from the star to the interferometric combiner via each telescope (optical path delay, (OPD)) be equal to within a few wavelengths, some tens of microns. This is achieved coarsely by the formation flying arrangement of telescopes, routing of the stellar beams, and placement of the combiner instrument. For high-sensitivity nulling interferometry, the path matching must be accurate and jitter free at the nanometer level. This is achieved by closed-loop control of the optics and very low disturbances on each spacecraft. Nulling interferometry cannot begin until this equal-OPD condition has been achieved. This so-called "fringe acquisition" step begins after the instrument pointing control loops have been acquired. With careful measurement of angles and distances within the instrument and angles with respect to the target star, one can estimate the relative OPD and OPD rate-of-change. As with StarLight [8], a complement of instrument-based sensors and perhaps absolute ranging from the formation-flying sensors is sufficient for these measurements. Extension of this process for longer baseline (for astrophysical imaging) can begin to drive the requirements of this fringe-acquisition sensor suite.

The ideal spacing of telescopes in the array, Fig. 9, is mainly determined by the typical planet-star angular separation and the instrument's wavelength range. For the linear array interferometer designs baselined by NASA, the optimum end-to-end array length is 30 – 50 m. However, with 14 m-diameter sun shields (similar to those used for James Webb Space Telescope (JWST) [15]), the closest together one could reasonably bring the telescopes to each other is 15 - 20 m, i.e. 1 - 6 m "wingtip-to-wingtip", for a full linear array length of 45 - 60 m. Reducing the shield diameter would reduce the sky coverage and hence the depth of search for planets. Thus there is a strong motivation to provide formation flying at very close distances. This will require high reliability and robustness in both the software and the hardware.

Formation sensor requirements and options -- The coarse formation flying challenge encompasses all conditions from post-launch deployment or "lost-in-space" reacquisition to the acquisition of instrument pointing loops. Bearing angles to any spacecraft could be anywhere in the celestial sphere, and the inter-spacecraft distance could range from 15 m up to a few kilometers. The coarse formation sensor should behave gracefully throughout these ranges, both for safety and for simplicity of the software. To avoid solar glare problems, the coarsest sensor, covering the full 4π-steradian sphere, will almost certainly use RF beacons and sensors. The AFF sensor developed for StarLight is an excellent candidate for this application [9]. Reflections from metallic coatings on the thermal shields are expected to limit the accuracy of the RF sensor; thus an intermediate-resolution sensor will probably be needed to bridge the gap from the coarse sensor accuracy to the level needed for robust acquisition of the instrument pointing loops. Candidates for this intermediate sensor include an optical beacon-and-camera system and a narrow-beam version of the AFF sensor. These sensors, together with the formation flying software, must be robust enough to maintain the formation without unrecoverable mishaps throughout a 5-10 year mission.

The specific range and bearing requirements will depend on the details of implementation of TPF. Using the design of StarLight as a guide, we can give approximate requirements. For coarse formation flying, the distances must be controlled to a few centimeters, and angles to approximately one arcminute to allow acquisition of the pointing control loops for laser and starlight beams. Then data from the instrument can be used to provide finer knowledge of the formation, adequate to conduct a search for the stellar interference fringe. Because this interference signal is evident only for a few microns' range around the zero-OPD condition, this is a challenging step. For the planet-finding case, the stars will be bright, and the distances short. Thus the requirements can be fairly loose: angle knowledge to about 2 arcsec and 100 mas/sec, distance knowledge to about 1 mm and 50 micron/sec. For astrophysics observations, many targets will be much fainter, and the baselines may be 10-20 times longer. For the dimmest targets, the angle knowledge must be about 0.2 arcsec and 0.04 mas/s, and the distance knowledge about 0.1 mm and 20 nm/s. These assumptions are worst-case; there are several approaches under consideration to relieve these difficult requirements for dim-star astrophysics.

3. Modeling Work

In the modeling area, BATC has several ongoing activities including system modeling and proximate operations algorithm development and implementation.

High Precision Formations (HPF) Study

The NASA-funded HPF technology development effort is aimed at building and testing a formation geometry sensor that enables high precision formation knowledge and control for multi-spacecraft missions. We derive formation sensor requirements from integrated modeling and analysis of proposed space systems to trace mission requirements down to the sensor requirements. One of these missions is MAXIM Pathfinder, an X-ray interferometer that uses grazing incidence mirrors in one or several optics spacecraft and a distant detector spacecraft to form the entire interferometer. The high resolution mode of operation requires that several optics spacecraft be held in precise formation with each other to give a stable image at the detector spacecraft.

The geometry of the formation and some of the required control and knowledge requirements are shown in Figure 10. The sensor under study is that needed to maintain the formation of the optics spacecraft (hub and freeflyers). In addition to the distance stability requirement and the out of plane requirement, there is a roll requirement on the freeflyer spacecraft. These requirements are being refined through modeling of system misalignments and analysis of their effects on the image quality.

A possible sensor concept for monitoring the optics spacecraft formation geometry is based on combining inter-spacecraft beacons using a beamsplitter and measuring deviations from overlap to determine departures from

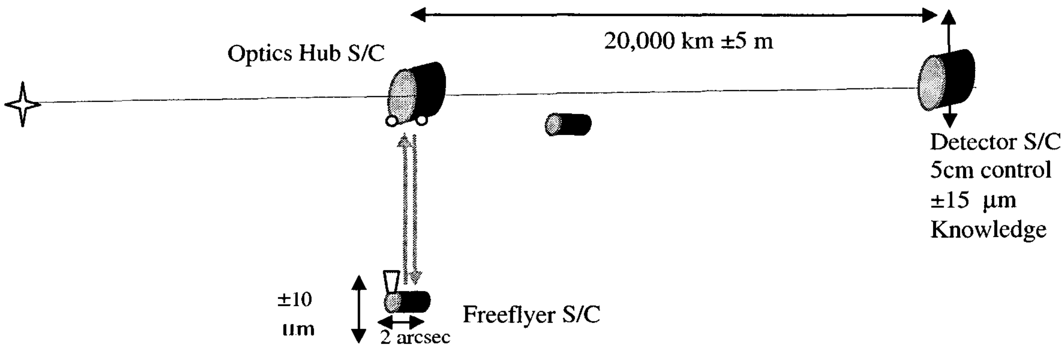

Figure 10. Proposed formation geometry for the high resolution mode (microarcsecond resolution) for the Maxim Pathfinder. Figure courtesy Keith Gendreau, NASA/GSFC.

nominal formation geometry. Such a system could determine all relative bearing and attitude parameters, leaving only range to be determined to set the scale of the formation. We note that the formation sensor requirements for the planet search mode of a multi-spacecraft interferometer version of the Terrestrial Planet Finder (TPF) are close to the required formation knowledge/control requirements for the optics spacecraft relative positioning on MAXIM Pathfinder. A similar approach to formation sensing may be applicable to both systems.

ITM use with formation flying executives

To solidify our understanding of formation flying technology, BATC is building a modeling software environment to facilitate analysis and trade studies. We are developing a computer laboratory that will combine several commercially available software packages into an engine for generating spacecraft trajectories of all types. The laboratory is being used to study a formation flying mission wherein two satellites with complementary sensor suites will be used to study weather and other events from a moderately high inclination, low earth orbit. This lab is planned for use in formation flying and proximity operations studies of all types including in planetary orbits or deep space. It will also handle any combination of sensors and actuators to accomplish the formation flying mission.

BATC has supported development of an integrated modeling environment for telescope performance modeling and analysis. The Integrated Telescope Model (ITM) [16-18], the realization of this effort, has been used on several current and future large telescope programs such as the VLT, NGST, TPF and MAXIM. The ITM is a new project tool for cradle-to-grave support for the system engineering function. It permits the user to do both time simulations and analytical work in the spatial/temporal frequency domains. The individual discipline models (structural dynamics, optics, controls, signal processing, detector physics and disturbance modeling) are seamlessly integrated into one cohesive model to efficiently support system level trades and analysis. The core of the model is formed by the 'optical toolbox' implemented within a commercially available mathematics package and realized in an object-oriented environment. Both geometric and physical optical models can be constructed and interfaced to disturbance and detection models. The geometric approach includes ray tracing for exact modeling or sensitivity matrices for rapid execution. The BATC-developed ITM will be enhanced to permit interface with formation flying executives. This will allow total end-to-end system performance evaluation of formation flying and the interplay with the metrology system.

Figure 11 shows the overall structure of the ITM with MAXIM Pathfinder as an example system [18]. The model shown in the lower left pane represents one of several spacecraft in the formation. It receives two inputs from outside sources – metrology beams from other spacecraft and commands from an overall formation flying executive controller. For the MAXIM Pathfinder concept, multiple spacecraft fly in formation at L2. Internal metrology maintains alignment of the flat, segmented optics shown in the upper right block. All structural motion, rigid body and flexible, operates directly on the optics allowing true analysis of dynamic effects on the resulting optical image. This information can then be directed back to the ACS and the formation flying executive control systems.

Proximate operations algorithms

BATC has been examining proximate operations algorithm modeling and simulation. Our work to date, has focused on four different types of algorithms; Clohessy-Wiltshire (Hills)[19-24], Lambert (in-plane and out-of-plane) [19,20], closed loop controller [25], and the genetic algorithm [19,26]. The following discussion presents the benefits and disadvantages of each of these algorithms. In this discussion, the terms "proximity operation" and "formation

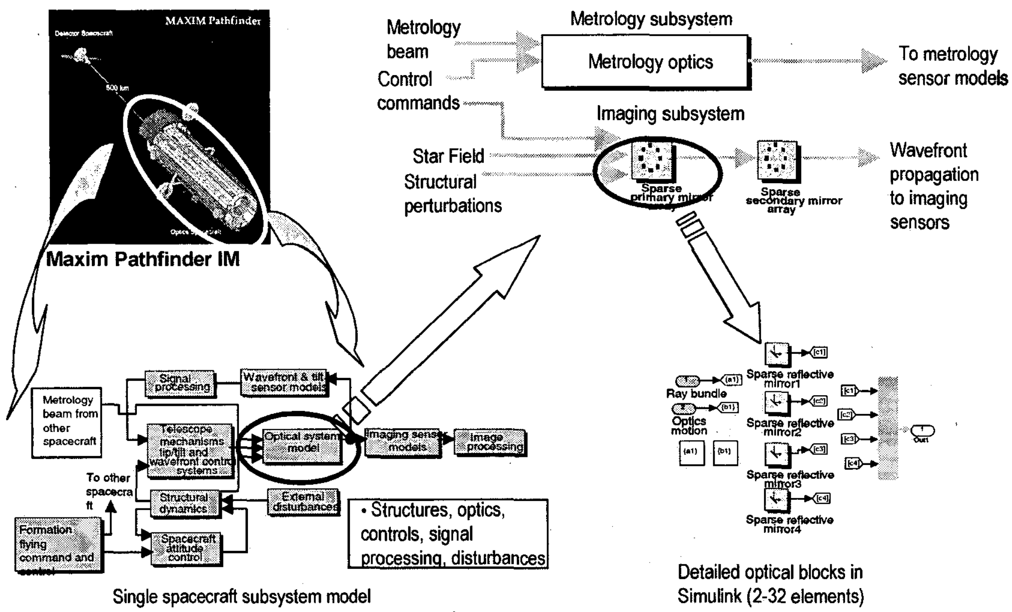

Figure 11. ITM based integrated model of MAXIM – model layering of MAXIM system, optical subsystem, and individual optical elements [18].

flying" are synonymous. Each term can be used to represent a spacecraft that maneuvers with respect to a second spacecraft. The maneuvers can be station-keeping maneuvers (as in a long-baseline interferometer mission) or they can be GN&C maneuvers (as in circumnavigation of the second spacecraft). Also, a spacecraft need not rendezvous with a second spacecraft. Often the term is used to denote a maneuverable spacecraft moving with respect to a location in space, thus it can *rendezvous* with a point. Figure 12 shows typical formation flying geometry.

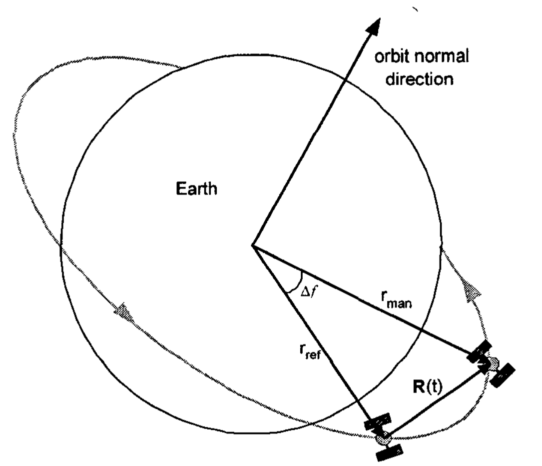

Figure 12. General LEO Formation Flying Geometry.

Clohessy-Wiltshire (Hills) Algorithm-- The Clohessy-Wiltshire (CW) algorithm has one distinct advantage over most orbital dynamics algorithms. It exists in a closed form solution that can be used onboard a spacecraft to calculate the next ΔV maneuver to achieve a desired proximity state. However, to derive the CW equations one must make several simplifying assumptions (e.g., the second spacecraft must be in a circular orbit and both vehicles must be modeled in an Earth point-mass gravity field) [19-24]. These simplifying assumptions have the potential to lead to significant position errors if care is not taken to use the CW equations correctly.

Lambert Equations-- If a good Earth gravity model is available to the mission planner then the Lambert equations can be used to calculate the ΔV expenditure at t_0 that is required to achieve a specific position at some later time, t_1 [19,20]. For proximity operations, the Lambert equations would be applied as follows: 1) Given the position and velocity of each vehicle at t_0, 2) propagate the reference vehicle forward to t_1, 3) locate the desired position of the maneuvering vehicle with respect to the reference vehicle at t_1, 4) input the position from step 3 into the Lambert equations and solve for ΔV at t_0. The Lambert equations are well understood and they are outlined in many textbooks. However, they require an iterative scheme to solve, so they may not be appropriate for every on-board processor.

Closed Loop Controller -- The Closed Loop (CL) Controller is useful for very close proximity operations or for precision maneuvering [25]. Typically, a CL controller will neglect orbital dynamics and treat the maneuver problem as an inertial dynamics problem. The CL controller is set up as a typical proportional-integral-derivative (PID) controller and it calculates the ΔV at every calculation step. Of course, for the most accurate controller, one will need inputs from a position sensor, a velocity sensor, and an acceleration sensor to drive the CL controller. Actuators must have a small minimum impulse bit to achieve optimal precision.

Genetic Algorithm -- The Genetic Algorithm (GA) is used to improve upon any of the other algorithms [19,26]. For example, the CW equations could be evaluated for an initial guess of ΔV at t_0. If the ΔV is applied to a satellite that is propagated in a high fidelity gravity model (say, J_2, J_4, or higher) then the satellite will not gain the final desired position. The GA is used to improve the CW initial guess by creating a population of random ΔV vectors. Then it evaluates the population and chooses the best individual from the population. The algorithm has the ability to randomly mutate and select some of the individuals from the population. Then it re-evaluates the population and chooses the best individual. After this procedure has been completed many times, the GA converges to a solution that performs much "better" than the CW solution. The term "better" is a relative term and can be applied to many variables such as better range error, better velocity expenditure, etc. See [26] for a detailed discussion of the GA applied to spacecraft rendezvous.

4. FORMATION FLYING TECHNOLOGIES

Microsat Methodologies

Many customers are displaying increasing interest in missions requiring multiple cooperative microsats flying in formation. Here the term microsat refers to a satellite with a total mass of 100 kg or less. Designing microsats to support formation flying missions requires special consideration, particularly in the areas of sensors, actuators, and processing.

Sensors for formation flying may be generally divided into two classes: spacecraft attitude determination sensors; and proximity operations sensors. Examples of attitude determination sensors are magnetometers, sun sensors, inertial measurement units (IMUs), and star trackers. For microsat applications, it is desirable to reduce the mass of these sensors as much as possible. Off-the-shelf magnetometers and sun sensors are already light weight (~ 0.4 kg for a magnetometer and ~ 0.08 kg for a sun sensor). There are MEMS magnetometers in development that have negligible mass, but they are not readily available for flight. There are commercially available lightweight IMUs that weigh less than 1 kg that would be appropriate for a microsat. There has been much development effort directed at MEMS gyros and accelerometers; the MEMS gyro devices are still developmental, but MEMS accelerometers are currently being incorporated into real devices. Conventional star trackers are probably too heavy for microsats, but lightweight star cameras would certainly be appropriate. Several companies (including BATC) have concepts for miniaturized star cameras that weigh on the order of 1 kg.

Proximity operations sensors would be used for close-in navigation relative to another space vehicle. Acquisition and tracking in sunlight could be done with a visible or near-IR optical sensor, or an RF sensor. Acquisition and tracking in shadow would most likely be accomplished with a mid-wave IR sensor or RF sensor. Ranging could be performed with a laser rangefinder, a millimeter wave radar system, or RF system. For a microsat platform, the mass and power requirements for each of these options is probably a close trade. A complete proximity operations sensor suite consisting of a visible sensor, an IR sensor, a ranging device, and the requisite power processing could probably be implemented in less than 10 kg.

Actuators suitable for microsat attitude control include gravity gradient booms, magnetic actuators, momentum/reaction wheels, and various micropropulsion reaction control options. All of these actuator types have been successfully used on microsats. The choice of actuators for a specific mission depends on the pointing accuracy and agility required, as well as the mass and power available. Further, formation flying microsat propulsion presents a challenge due to the severe mass constraints levied on a microsats, as well as the small impulse bits required to support very fine maneuvering. It is likely that precision formation flying missions would require active three-axis attitude control.

Microsats that support formation flying and/or proximity operations have special onboard processing requirements. The processor must be able to perform the specialized tasks of acquisition, tracking, ranging, relative navigation, and closed-loop control. The processor must also support the other routine spacecraft bus functions of attitude determination, command and telemetry processing, data transfer, fault protection, and mode control. In addition, the processor may be required to perform data processing for the primary payload. There are several new processor architectures emerging that use the RAD-750 CPU. These architectures are generally compact (3U form factor) and have an available throughput of at least 100 MIPS. This robust thru-put capability should be adequate for most microsat formation flying missions.

Formation flying sensors

Establishing a formation requires knowledge and control of relative range and bearing angles. A tiered formation acquisition process can be initiated using relatively coarse RF sensors with wide FOV, allowing the vehicles to acquire each other from distant range and virtually any bearing. The RF system enables adequate formation range and bearing control for transition to more accurate but narrower FOV sensors (e.g.: optical). The sensors, in turn, provide sufficiently accurate formation control to allow acquisition of higher precision angle and range metrology systems that may operate over a very narrow FOV. This layered approach allows progressively finer knowledge (and optionally control) of the formation as needed to meet the requirements of each mission.

GPS for precise relative navigation

The capability exists today for formation flying spacecraft that are equipped with GPS receivers and communication links (either between spacecraft, through a relay satellite, or through ground stations) to formation fly [27-35]. Each spacecraft needs to receive at least four independent GPS signals to estimate their own precise relative position and transmit the position estimate and/or raw GPS data to the other spacecraft, thereby providing information for each spacecraft to determine its own precise relative position from the other. This method can also be used for relative attitude determination between spacecraft.

The principle behind precise GPS relative spacecraft position estimation is called carrier-phase differential GPS (CDGPS). The measured phase of the GPS carrier is compared to the carrier phase measured at the other spacecraft. The range measurements achieve precisions that are a few percent of the carrier wavelength (<0.5 cm). Since the spacecraft (and therefore their GPS antennas) are separated by more than one wavelength (19.2 cm), the position solution is ambiguous since the number of wavelengths in the phase difference is undetermined. The main computational burden using this technique is to solve this carrier phase integer ambiguity. Rapid algorithms for solving the carrier phase integer ambiguity are required for precise relative navigation due to frequent re-initialization of the integers. These reinitializations occur during events such as: initialization of the formation; insertion of new satellites in the constellation; high multipath environments, when GPS satellites move into and out of view; and attitude maneuvers between formation flying spacecraft. Research results indicate that relative spacecraft positions can be estimated using this technique with an accuracy of 2-5 cm at up to a 1 Hz rate.

There are three types of GPS-based relative navigation configurations corresponding to three mission types:

- LEO to low HEO missions have adequate signal coverage (≥ 4 GPS satellites visible) and can use the existing GPS infrastructure with current space-qualified GPS receivers. Absolute position reference for this mission is calculated from the GPS network.
- High HEO missions have inadequate signal coverage (< 4 GPS satellites visible) and require augmentation of the existing GPS network in the form of signal augmentation with pseudolites. Absolute position reference for this mission is also calculated from the GPS network.
- Deep space missions have no signal coverage (0 GPS satellites visible or out of range) and require a stand-alone GPS network consisting of only pseudolites. Absolute position reference for this mission is calculated from DSN Doppler ranging measurements.

The pseudolites are GPS-type transceivers that can transmit and receive GPS-like signals and communicate GPS-like data independent of the NAVSTAR network. All mission types require a transmitter on each spacecraft to broadcast its relative position solution and/or raw GPS data to the other spacecraft. These three mission types and their associated relative navigation configurations are summarized in Table 4.

BATC is equipped to perform detailed GPS software development, integration, and testing and has experience in this area from the QuickBird and ICESat programs. A GPS RF signal generator is part of the standard test equipment in the software test bench lab for testing and evaluating dual-frequency GPS equipment and sensors integrated with flight code. End-to-end modeling with hardware in-the-loop for formation flying spacecraft can be accommodated with the existing set-up.

Interaction between reaction wheels and thrusters

Attitude maneuvers are generally expected to be executed using reaction wheel assemblies. Thrusters are not used while performing attitude maneuvers, conserving propellant for other operations. For example, the reaction wheel assemblies for StarLight were to be at zero spin speed during interferometer data collection periods in order to eliminate a significant source of induced vibration that would disturb instrument line-of-sight pointing and optical phase jitter stability.

Table 4. GPS-Based Relative Navigation Configurations

Mission Type	GPS Satellites in View	Network Type	Required Equipment	
			Relative Position*	Absolute Position
LEO to low GEO	≥ 4	NAVSTAR	GPS receivers	NAVSTAR GPS network
high GEO	< 4	augmented NAVSTAR	GPS receivers, pseudolites	NAVSTAR GPS network
deep space	0	stand-alone	Pseudolites	DSN ranging

* all spacecraft require communication links either as a separate system or built into the pseudolite design.

The thrusters' primary functions are to actuate formation translation maneuvers and de-saturate angular momentum from the reaction wheels (in the absence of a means to perform angular momentum management by actively controlling the vehicle's center-of-pressure to center-of-mass offset). On the StarLight mission, the thrusters would provide attitude and position "deadband" control during interferometer data collection when the reaction wheels were at zero spin speed. Instrument optical surfaces would be counter-steered to correct for the modest attitude and position drift inherent in the "deadband" control strategy.

Propulsion Options

Propulsion trades have been conducted to select the reaction control/propulsion subsystem for various missions. [12,36-40] Options have included conventional liquids, helium cold gas, nitrogen cold gas, pulsed plasma thrusters (PPT), field emission electric propulsion (FEEP), colloidal thrusters, hydrazine warm gas and unconventional options such as MEMS, butane and hydrogen peroxide micro-propulsion. Drivers have included small impulse bits, impulse bit repeatability, propellant efficiency, particulate and electromagnetic contamination, and tight mission cost constraints. For example, nitrogen cold gas was ultimately selected for StarLight [7,12] to maximize the reliability of demonstrating precision formation flying and optical interferometry.

The best choice for a given formation flying mission will, to a first order, be a function of the accuracy of formation flying and attitude control achieved propulsively and the proximity of the satellites in the formation. In general, the finer the formation flying and attitude requirements, the smaller and more repeatable the impulse delivery needs to be. The closer the formation flying satellites are to one another, the cleaner the propulsion system needs to be. Conventional hydrazine monopropellant systems are readily available, but may not be the best choice for close-in maneuvering where minimum impulse bit and plume impingement are of concern. Hybrid hydrazine/warm gas systems are a possible solution, but some of the requisite hardware is developmental (e.g., gas generators). Electric propulsion devices such as PPT and FEEP thrusters provide extremely small impulses, but plume impingement could still be an issue due to the propellant species involved (Teflon® and cesium or indium). The power budget may also not support prolonged operation of electric propulsion devices. Cold gas systems address both problems, but are not particularly efficient (specific impulses of ~65 s). Other exotic propulsion systems have been studied (MEMS micropropulsion), as well as the use of unconventional propellants (e.g., butane), but these are developmental.

5. CONCLUSIONS

Ball Aerospace & Technologies Corp. is involved in many facets of spacecraft formation flying. Work includes mission development, detailed mission studies, formation flying sensor and actuator system and component development, systems modeling and proximate operations algorithm development and implementation. Our work covers both the deep space and Earth-orbital regimes.

BATC is developing the CloudSat spacecraft which will formation fly with four other satellites. CloudSat will have to maintain its orbit within a specified orbital error box with respect to the Calipso spacecraft in Earth orbit. BATC is responsible for developing the NEXTSat spacecraft as part of the DARPA Orbital Express Program. NEXTSat will be the target for on-orbit rendezvous and docking technology demonstration work. As the flight segment provider for the Deep Impact Discovery Program, BATC is developing both the Flyby and Impactor spacecraft and working the encounter scenario where the Impactor and Flyby spacecraft formation fly with respect to Comet Tempel 1. As part of the StarLight definition program, BATC was developing the two spacecraft which were to formation fly to within centimeters in range and arcminutes in bearing. As part of this work, BATC traded propulsion system options and defined propellant needs, assisted in establishing the tiered formation acquisition approach, examined alternative formation flying sensors, and examined fault protection/formation safing requirements. BATC has been examining architecture options for the Terrestrial Planet Finder Mission for several years. For an interferometric TPF architecture, BATC has conducted requirements analysis, propellant utilization assessments, propulsion trades, formation sensor options, and telescope spacings.

In the modeling area, BATC has several ongoing activities. In the High Precision Formation Study, BATC is conducting detailed requirements analysis and modeling in a effort to define a formation sensor system for the grazing-incidence, X-ray mirror segments which formation fly at very high accuracy and make up part of the MAXIM X-ray telescope. BATC is also developing a modeling lab, which combines commercially-available software packages, into an environment to assess various mission scenarios including formation flying missions. Finally, BATC is developing and implementing proximate operations algorithms for stand-alone analyses and/or integration into the modeling efforts described above. The Clohessy-Wiltshire (Hills) algorithm, Lambert equations, closed-loop controller and genetic algorithm are being assessed.

Formation flying technology work has also been addressed at BATC. Activities have included use of the RF inte-satellite link for coarse range and bearing information, star tracker based range and bearing determination, optical system range and bearing determination, GPS use for range and bearing determination, and reaction wheel-thruster interactions.

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