LOW-COST, MINIMUM-SIZE SATELLITES FOR DEMONSTRATION OF FORMATION FLYING MODES AT SMALL, KILOMETER-SIZE DISTANCES[1]

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Abstract. In-flight demonstration of close-range formation flying modes is discussed with emphasis on low cost, based on spacecraft design simplicity and minimum size, and hence minimum launch cost. Piggy-back launch of three identical demonstration spacecraft is the preferred option. This demonstration includes comprehensive collection and analysis, on the ground, of relative motion data derived from GPS signals that are received by the participating satellites. In addition to realistically reflecting the known Keplerian and non-Keplerian characteristics as part of the flight dynamics analysis, an important concern is to demonstrate the less well-known effects of differential drag perturbations. These can be deliberately induced by small aerodynamic cross section variations on the satellites. Results of the demonstration mission will benefit currently projected and other future missions that require close-formation flying, e.g., radar and visual observation missions, by proving feasibility, assisting detailed and comprehensive operations planning, and helping diminish their potential risks.

1. Introduction

Formation flying at very small distances, of the order of kilometers or less, is currently being projected for military and non-military observation purposes (References 1 to 11). This novel mission class necessitates specific flight control techniques and safeguards to achieve the intended observation objectives without posing unacceptable risks.

This paper considers examples of representative, close-proximity formation-flying modes and their implementation. Of these mission alternatives, one particular close-proximity relative motion type is selected to define a low-cost demonstration flight program that should precede the first actual "full-up," operational close-formation mission. This demonstration program is intended to show operational procedures and flight control techniques to be used for achieving specified mission objectives. Even without carrying any of the payloads that would be part of actual close-formation flights, the demonstration mission is considered essential to exhibit realistic characteristics, but also potential implementation problems, inherent in this new technique of making the intended joint observation at an unprecedented close proximity between the satellites involved.

Emphasis is placed, therefore, on demonstrating that these objectives can be met without risking a collision, due to imposing unsafe relative motion criteria or experiencing trajectory perturbations that were neglected or not sufficiently well understood. Of primary concern are perturbations caused by differential drag acting on the satellites flying in very close proximity. Also of concern are perturbations due to orbital motion around a non-spherical Earth, such as J_2— effects superimposed on Keplerian orbital dynamics. In Section 2 some typical formation flying patterns and their different characteristics will be discussed.

An important consideration in the proposed demonstration program is to minimize cost by selecting a small-size, simple spacecraft design based on earlier design concepts of similar configuration and operating characteristics. Reduced size, design simplicity and high reliability are keys to a low-cost demonstration flight program. As will be discussed in Section 3, a configuration reminiscent of some spin-stabilized, early spacecraft designs is selected for each of the formation-flying satellites, with a spin axis orientation normal to the orbit plane.

The preferred demonstration spacecraft design is, of course, much smaller in overall size, than earlier spacecraft designs it resembles, such as the interplanetary

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Pioneer 6 to 9 series flown in the 1960's, and correspondingly of much smaller mass (approximately 50 kg). For example, a satellite with one half of the reference spacecraft dimensions would probably have much less than a quarter of its mass. Its power requirements also are very much smaller, considering the relatively small communication distances that are involved (i.e., LEO and Earth, rather than interplanetary space and Earth).

The small size and mass of the selected demonstration satellites will permit their launch, in piggy-back fashion, by an expendable launch vehicle, together with other primary payload spacecraft, or on-board the Shuttle Orbiter, with low-Earth orbital characteristics that will be compatible with those of the primary payloads being carried. It would, of course, be best, from a launch cost point-of-view, to make these miniature demonstration spacecraft small enough for several of them (perhaps two or three) to fit within the total secondary payload mass and size capacity of the selected launch vehicle, in addition to its primary payload.

The formation-flying demonstration, and the simultaneous evaluation of flight data on the ground, are likely to exhibit discrepancies due to incomplete modeling of the actual flight characteristics of the satellites in their closely spaced orbits. The planning and execution of the intended demonstration sequence, as well as the determination of the magnitude and nature of the discrepancies to be observed are of principal interest. The results are of critical importance to the design, planning and execution of actual future formation-flying missions.

The evaluation program will consist of obtaining precise relative-motion information based on Global Positioning System (GPS) signals received by each of the participating spacecraft. Individual position and orbit determination, based on GPS signals received by each demonstration satellite in the formation, yield the desired relative motion information in real time. A basic question concerns the degree of operational and control autonomy that the satellites should have. At one end of the spectrum is complete lack of autonomy, where all relevant flight data of the formation are determined by ground data processing and evaluation, thereby relieving the satellites of inter-satellite communication to determine their respective relative positions. A more complex alternative, with partial spacecraft autonomy, consists in letting each satellite determine its respective drift relative to its assigned position in the constellation, but without obtaining its

position relative to the other constellation members. In this case, the ground station(s) are not required to transmit every corrective maneuver command to each constellation member. A key issue is the lack of frequent ground station contacts. At the low altitude envisioned for this demonstration mission there are only a few, say 4 to 6, contacts per day, depending on the satellite altitude and orbit inclination and the ground station latitude, plus additional contacts if more than one ground station is involved. This fact alone necessitates some degree of corrective maneuver—or station-keeping—autonomy. These questions will be addressed in Section 3.

Also of concern is the ability of predicting a potential collision hazard between the demonstration spacecraft, and accordingly determine the need for evasive maneuvers, and their direction and magnitude. Again, continuous data monitoring on the ground, determination of any evasive maneuver requirements and transmission of command signals to one or several of the participating demonstration satellites, is simpler and less costly for this demo mission than letting the satellites handle the task autonomously. On the other hand, the more complex autonomous data-handling and maneuver-determination tasks by the satellites, themselves, may be of sufficient interest to future formation flying missions to warrant their implementation, even at the higher complexity and cost that would be involved. Also, considering that there may be at times several hours between successive ground station overflights, autonomous collision avoidance may become a necessity for close formations. To further illustrate the activities that have to be conducted by the satellites and the ground station(s), a typical scenario and operating sequence that may be used in planning the demonstration mission is described in Section 4.

The demonstration flight program proposed here is intended to permit rapid development at low cost, based on the small size and the design and operation simplicity of the mini-spacecraft that will be used for this purpose. Clearly, it is essential to obtain these results as early as possible, say, on a two-to-three-year schedule, so as to provide the needed input data for effective, realistic and minimum-risk planning of actual, close-distance formation-flying missions that are being considered in the near future. Cost and development schedule estimates are presented in Section 5 to support our projection that the demonstration mission can be performed in a timely manner to provide these benefits.
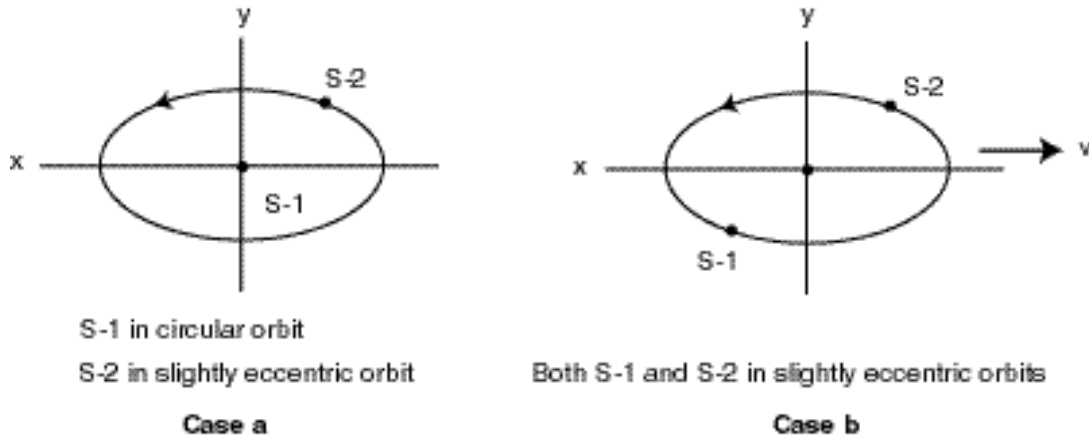
Figure 1. In-plane Formation Flying Patterns of Satellites S-1, S-2.

2. Close Formation Flying Examples.

Examples of close formation flying modes are described in the literature on the subject (References 1 to 11). They include formations with "in-plane" relative motion, such as two spacecraft flying in adjacent orbits, either one being circular and the other very slightly eccentric, or with both spacecraft in orbits of equal small eccentricity, with their respective apogees or perigees 180 degrees apart. The resulting relative motion is illustrated in Figure 1. In Case a, the first spacecraft (S-1) moves in a circular orbit, the second one (S-2) in an adjacent, co-planar slightly eccentric orbit, with the semimajor axis being equal to the radius of the circular S-1 reference orbit. In this case the relative motion of S-2 with respect to S-1 is a small ellipse with the ratio of its major to minor axis being 2:1, with S-1 being located at the center of this ellipse. The relative motion is described by Hill's equations (Refs. 2 and 10), also sometimes referred to as Clohessy-Wiltshire equations, with high accuracy for the very small excursions involved (see Appendix). In Case b, both spacecraft move in eccentric orbits of equal dimensions, such that their relative motions with respect to a common circular reference orbit are on the same 2:1 elliptic trajectory, with one spacecraft passing apogee at the time when the other one passes perigee.

One can place more than two satellites into a close, co-planar formation pattern comparable to that in Figure 1, such that they remain at a fixed separation (120-degrees if 3 satellites or 90-degrees separation if 4 satellites). A corresponding, symmetrical relative motion pattern would apply for a greater number of satellites flying in this kind of formation.

The above description omits the apsidal advance rate (d /dt) which is caused by the J_2-term, due to the non-

spherical Earth, since Hill's equations are derived with the assumption of Keplerian orbital mechanics for a strictly spherical Earth. However, since this advance rate is very nearly equal for the assumed extremely small eccentricity of the orbits involved, the effect of this eccentricity difference has been neglected. Thus, the relative motion pattern remains invariant, although it moves in the forward direction. For a reference circular or near-circular orbit of 600 km altitude and 30 degree inclination, this common apsidal advance rate is about 0.74 deg per orbital revolution, or 11 deg per day.

Figure 2 shows another relative motion pattern for close-distance formation flying, with S-2 moving normal to the orbit plane of S-1, having small sinusoidal excursions with a period equal to that of the circular reference orbit (Case c). The orbit of S-2 has the same inclination angle, but a very slight difference in its longitude of ascending node on the equator, i.e., the
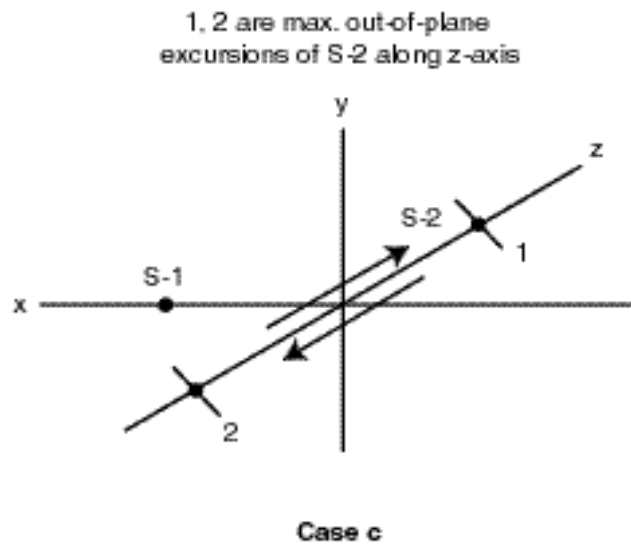

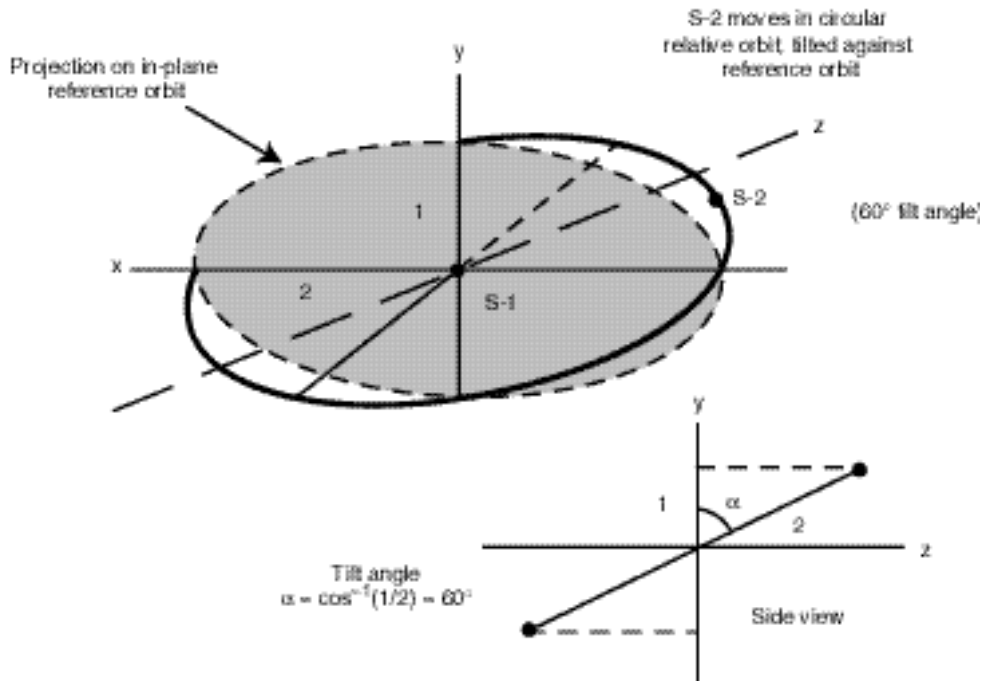**Figure 2. Out-of-Plane Formation Flying Pattern of Satellites S-1, S-2.**

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Figure 3. Out-of-Plane Formation Flying Pattern with S-2 at Constant Distance from S-1 at Center.

two orbits are slightly tilted with respect to each other. (This relative motion also is described in the Appendix).

This "out-of-plane" formation flying pattern is distinct from one where the two spacecraft orbits would have *slightly different orbit inclinations*. The relative motion of S-2 with respect to S-1 again would be sinusoidal. However, in that case the two orbits would have slightly different nodal-regression and apsidal-advance rates, due to the distinct J_2-effects, a condition not suitable for maintaining the desired (nominally) invariant relative motion pattern between the two spacecraft.

A combination of Case a and Case c is of potential interest. Here (see Figure 3), the out-of-plane motion of S-2 is chosen to have a maximum excursion from the centrally located S-1 with exactly the same maximum distance as that of the 2:1 relative motion ellipse (i.e., the semimajor axis) of Case a. This means the effective relative motion of S-2—in an orbit that is slightly tilted with respect to the reference orbit plane—always remains at a constant distance from S-1, a fact that can be derived from the relative motion equations in the Appendix.

The simple formation-flying pattern, Case a, depicted in Figure 1, appears most suitable for use in the proposed demonstration mission, with co-planar relative motions that can be readily verified from the respective GPS data received by the two or more satel-

lites. The demonstration can be varied by successive eccentricity changes of the orbit of S-2. Other advantages of selecting this formation flying pattern will be pointed out in the next section.

3. Spacecraft Design, Operating Modes, and System Engineering Issues for Formation-Flying Demonstration

As mentioned before, the simple design concept of a spin-stabilized cylindrical spacecraft body with a protruding axial antenna boom is suitable for the demonstration mission. The two or more spacecraft flying in close formation all have identical configurations. The spacecraft gross mass is expected to be on the order of 50 kg. As a preliminary estimate, the cylindrical body dimensions are 1 m diameter and 0.6 m length. The body-mounted solar array produces approximately 50 W of power, which will be sufficient to support:

- the communication functions;
- the extensive on-board computing and data-handling tasks;
- occasional reorientation and orbit change maneuvers (including any fast evasive maneuvers that may be required to prevent collisions);
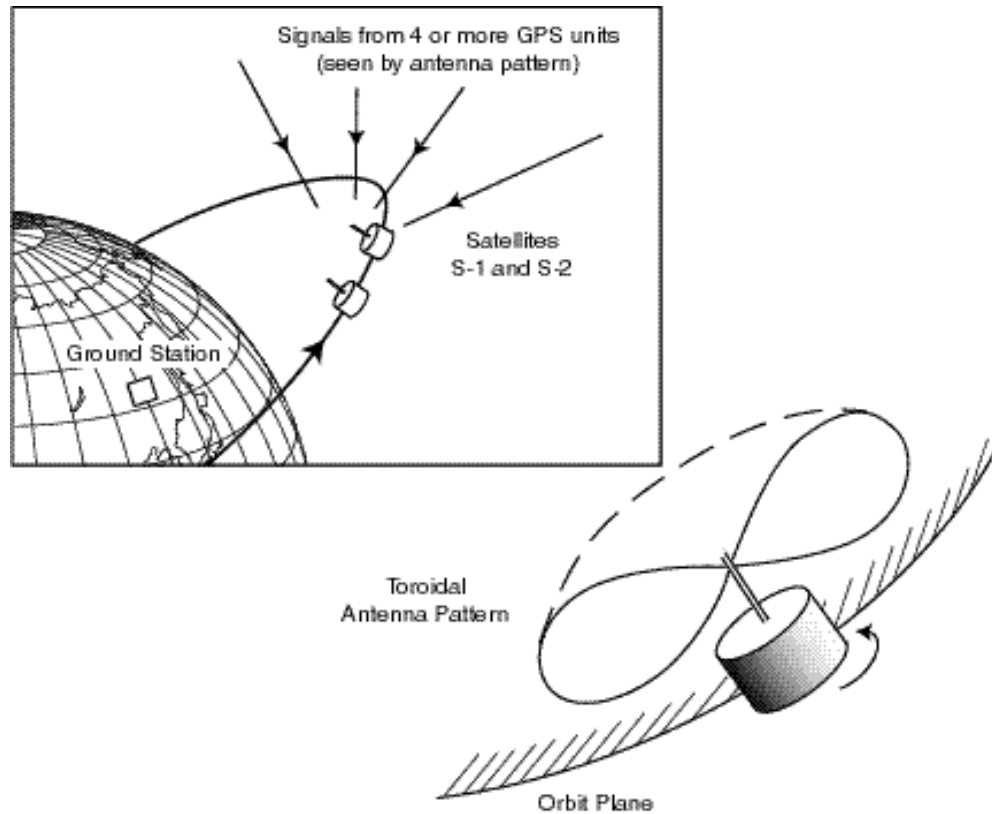
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Figure 4. Satellite Orientation with Spin Axis Normal to Orbit Plane.

• and battery recharging, with a maximum of 36 percent eclipse duration in low Earth orbit.

The spacecraft spin axis orientation is always normal to the orbit plane except at times when tangential orbit change maneuvers are performed. Therefore, the axial antenna boom with its toroidal rf transmit/receive gain pattern (see Figure 4) is continuously pointing in the direction that is most favorable for communication (a) with ground stations, and (b) with the other member(s) of the formation that move in, or very nearly in the same orbit plane, with their spin axes always oriented essentially in the same direction. Of particular importance is the additional property, (c), of having line-of-sight directions between the spacecraft and at least four or five GPS satellites that permit the satellites continuously receiving data for position determination. This factor is essential for achieving the demonstration mission objectives, by having up-to-date, accurate, individual (i.e., absolute) position data. By differencing these data, the relative orbital positions of the formation's members can always be established with high accuracy, typically of the order of tens of meters.

The diverse communication and data handling tasks to be performed by the participants of the mission, and the control functions essential to continue the mission successfully over the intended mission duration are illustrated schematically in a (simplified) functional flow diagram, as shown in Figure 5. The diagram centers on the functions of two of the satellites, receiving position and velocity data from several GPS satellites that are simultaneously in view, storing the data and transmitting them to the ground control station when in view.

Perturbing influences from non-Keplerian and non-gravitational effects also are indicated. They are generally well understood and being modeled and assessed by the ground control system, with the exception of drag effects that depend on many factors which are not as precisely known.

The functions of the ground control station, or stations, are listed in the figure, including tracking, command and control, and evaluation of the data transmitted individually by each of the participating satellites. This assumes that there is little or no inter-communication between the satellites themselves, except perhaps to signal an alert if a possible collision is threatening and evasive maneuvers are needed. Detec-
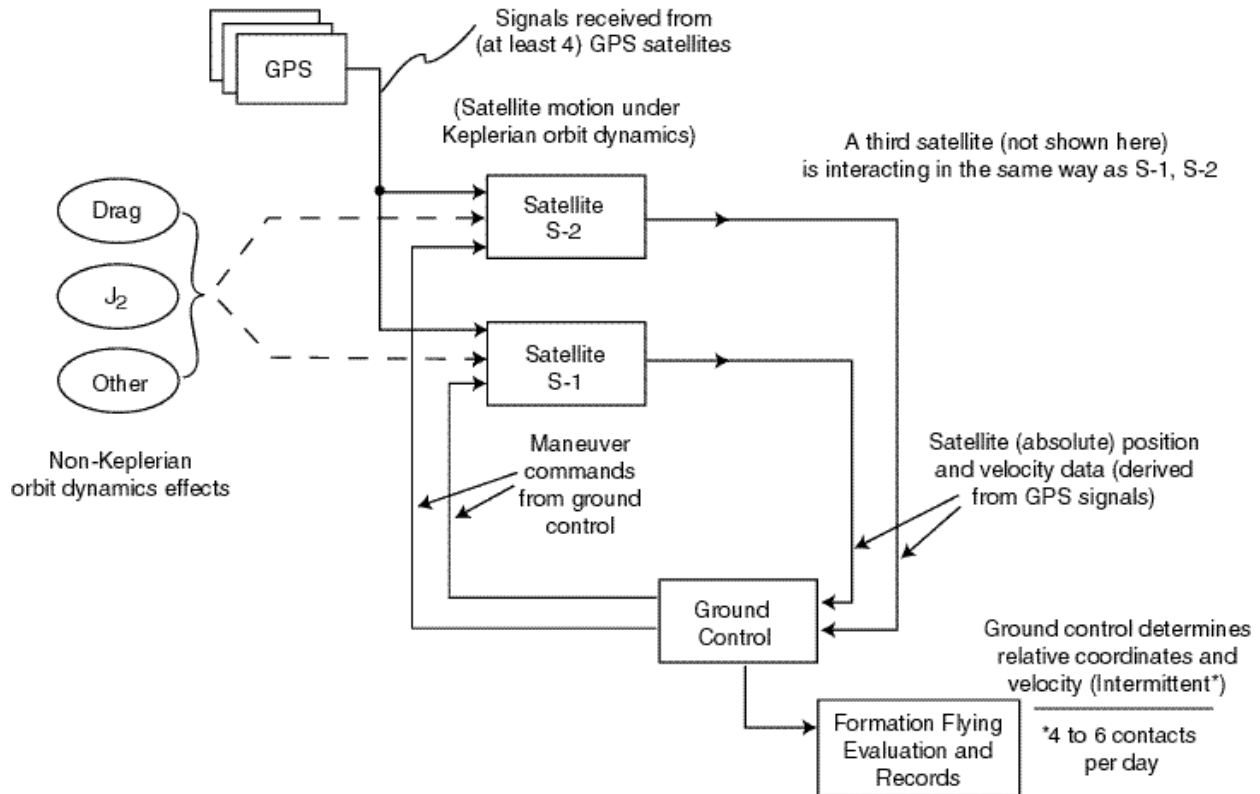
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Figure 5. Functional Flow Diagram

tion of such an emergency by a simple optical sensing instrument carried by each satellite, or a beacon, or by employing autonomous differential-GPS relative navigation and subsequent response by transmitting an alarm signal to the constellation member in question. An important factor to be considered is the intermittent radio contact with the ground station(s), only four to six times per day during northbound and southbound passes. The number of contacts per day depends on the ground stations' latitude and the satellites' orbit inclination and altitude.

Figure 5 also indicates orbit dynamics effects of Keplerian and non-Keplerian type that must be considered in establishing precise absolute and relative flight parameters. A key factor is the aerodynamic drag which is of concern even at the altitudes of 400 to 600 km, where the formation flying performance is likely to be demonstrated. Differential drag effects can be significant, since in actual formation-flying missions the participating satellites will not necessarily have an identical configuration, and some deployed appendages, like antennas, may be pointed in different directions relative to the velocity vector.

In this demonstration, even with otherwise identical designs and orientations of the formation members, differential drag effects can be created by small deploy-

able panels, to increase the aerodynamic cross section area on one of the satellites. With the assumed spinning configuration such panels can be deployed passively to various lengths from a rolled-up, stowed position. As shown in Figure 6, two rectangular panels, or wings, of a light-weight, transparent material, such as Kapton, on opposite sides of the spacecraft hull would serve this purpose. The power output of the body-mounted solar array will only be slightly degraded because of the transparency of the panel material. Centrifugal deployment to the desired length is effected by metal rods carried at the end of each panel. The resulting differential drag effect on the formation pattern, and its analysis on the ground, from the observed satellite relative position changes, would enhance the scope of the demonstration program.

Reorientation and transfer maneuvers of the spacecraft are performed by small monopropellant thrusters that are located on its hull, pointing in the axial direction. The principle of this simple thruster utilization was used on the earliest spinning satellites of the 1960s, such as the ones built by Hughes and TRW. Pulsed thrusting of a single such thruster, once per revolution, causes precession to the desired orientation. Continued thrusting by a pair of thrusters on opposite sides of the hull produces a velocity increment in the
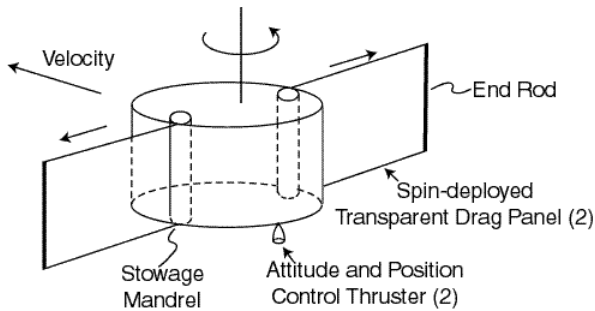
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Figure 6. Spin-Deployed Drag-Variation Panels (variable length).

axial direction, i.e., the direction to which the spacecraft spin axis has been precessed. Typically, the 90-degree reorientation, from the standard spin-axis orientation, normal to the orbit plane, to the tangential one required for orbital maneuvers, can be performed in several minutes. A similar time interval is required for the velocity change of a few meters per sec, by continuous thrusting in the axial direction for several minutes, using a pair of the circumferential small thrusters. Details will be determined when the spacecraft mass, moments of inertia and thrust magnitude have been established. With a reasonably high spin rate of up to 10 rpm and thrust forces of 10 to 20 Newtons, the spin-deployed drag enhancement panels will undergo only small deformations. These will disappear again soon after the propulsive maneuvers are terminated.

Emergency maneuvers to prevent threatening collisions can be performed much more quickly, since reorientation is unnecessary, and the evasion is in the direction normal to the common orbit plane of the two satellites approaching each other. For example, with a thrust acceleration of 0.4 m/sec^2 (assuming two thrusters of 10 Newton thrust each, acting on a 50-kg spacecraft) a 20-sec thrust impulse would cause an 80 meter excursion. The question remains of what kind of warning signal is used to trigger this maneuver. It may have to be generated by spacecraft-to-spacecraft communication after an on-board sensor on one or both vehicles recognizes the collision hazard. Ground communication usually will not be available in time to provide a safe alternative. A second concern is the sinusoidal nature of the relative out-of-plane motion. With the above maneuver characteristics, the maximum excursion after a quarter of an orbital revolution would be about 8 km. After half a revolution the evading spacecraft will return close to the relative position with respect to the other spacecraft that it had prior to the maneuver initiation. Thus, a follow-up maneuver also is required, however, with more time available for its execution.

In summary, the ground station(s), even with only intermittent contacts with the formation-flying satellites, will have a complete record of events and relative motion parameters transmitted to them from stored data onboard the satellites. Assessment of the satellites' formation flying record, both of the reasonably well-known cyclic and secular elements, and of the unmodeled drag influences affecting the formation's absolute and relative motion patterns will be a principal result obtained from the demonstration mission. This, in turn, will help in defining subsequent formation flying missions and performance requirements.

System Engineering Considerations

Formation flying is conceptually simple, but can pose significant implementation problems, because the interrelationship between the principal characteristics are often not fully understood and the objectives not well defined. Most formations are inherently unstable and will drift apart with time. On the other hand, the forces pulling a formation apart are small, such that nearly any formation can be maintained for a few hours with continuous thruster firing.

Because of the perturbative forces and lack of any long-term stability, formation flying necessarily represents a compromise between formation structure, accuracy, long-term maintenance, and propellant utilization. This trade is complex and involves a number of principal elements, specifically: the formation's size and shape (configuration), configuration properties over time, control vs. determination, configuration lifetime, propellant requirements, orbits, spacecraft properties, and attitude coupling.

- Formation size and shape (configuration)—The formation's size and shape determines to first order the relative natural motion of all the satellites.

- Configuration properties over time—Except for satellites directly in front of or behind a base satellite at the center of the formation, satellites undergo a natural oscillation with respect to the center. This oscillation can either be nulled out (excessive propellant), controlled (moderate propellant), or left uncontrolled (no propellant).

- Control vs. determination—Loose control with high accuracy relative position determination is far more economical than maintaining tight control and may satisfy mission needs equally well.

- Configuration lifetime—Nearly any configuration can be maintained for a short period. Longer lifetimes require an efficient formation and control process that minimizes propellant usage.

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- Propellant requirements—Lifetime and the fraction of the spacecraft mass allocated to propellant are directly related.

- Orbits—Orbits far from the Earth or other perturbing bodies are far more stable locations for formations than are low Earth orbits.

- Spacecraft properties—The non-gravitational perturbative forces (i.e., aerodynamic drag and solar pressure) will depend on both the cross sectional areas and surface properties of the spacecraft. For example cross-link communications may require different antenna pointing on each spacecraft, which changes the cross sectional area.

- Attitude coupling—Some spacecraft formations require knowing or controlling only the relative positions of the spacecraft. Other applications will also require high accuracy, correlated attitude control.

Each of these issues needs to be addressed in terms of the specific mission application. Just as there are many different solutions to spacecraft attitude control, there are as many approaches to solving formation control, depending upon the mission requirements.

The formation flying accuracy that can be achieved will depend on the specific implementation. The principal elements that determine this accuracy are the type of accuracy required (control vs. knowledge), the dynamic model, the control laws used, propellant budget, actuators, and sensors. Each of these must be considered as part of the formation flying systems engineering effort.

- Accuracy required (control vs. knowledge)—Is knowledge or control critical? Is the formation static or dynamic and what is meant by control accuracy in the case of a dynamic formation.

- Dynamic model—In most operational systems, the limiting accuracy is determined by the accuracy of the dynamic model; i.e., how accurately does the real spacecraft motion match the shape of the analytic model to which the observation data is being fit?

- Control laws used—How complex are the control laws and what are the limitations? Are they intended to optimize performance or minimize propellant utilization?

- Propellant budget—This is strongly related to how the spacecraft are controlled, the control law itself, and what is being optimized.

- Actuators—The key issues are granularity of thruster firings and the accuracy with which both

the magnitude and direction of the firings is known.

- Sensors—3 components of position and velocity for each satellite relative to the defined formation reference frame are required (either relative, or absolute).

4. A Typical Demonstration Mission Scenario.

Based on the functional aspects and the flight profile of the demonstration mission discussed in the preceding section, a typical mission scenario is described by listing the successive steps of the satellites' in-flight activities and those of the other participating elements. (Actual mission characteristics and operating modes and sequences that are to be selected, however, will be subject to performance and cost trades, that are beyond the scope of this discussion).

a. The satellites are launched and placed in orbit in close proximity of each other. Simultaneous launch is projected as a major cost-saving factor, considering the small size of the satellites. The initial demonstration altitude is 600 km.

b. Aided by ground control, an initial checkout of satellite system status is performed. Required functional corrections are commanded and performed by the satellites.

c. The satellites perform initial positioning maneuvers in accordance with the intended initial relative motion pattern, moving in a common orbit plane, as discussed in Section 2. This initiates Phase 1 of the demonstration.

d. GPS signals are continuously received by the satellites, and position and velocity data are transmitted to the ground intermittently during available contact periods, several times per day.

e. The ground station controlling the mission verifies and, if necessary, sends maneuver commands to correct the initial formation pattern.

f. Ground command continuously monitors the formation pattern and its cyclic and secular variations. It intermittently transmits corrective maneuver commands to the satellites in the formation to restore the intended relative motion pattern.

Note that certain secular motion components such as the common nodal regression and apsidal advance, which leave the formation pattern unchanged, do not require corrective maneuvers. Other secular motion components, particularly those due to differential drag effects must be corrected repeatedly. (Deliberate changes in differential drag effects are introduced by increasing

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the size if the deployed drag panels on one satellite). Cyclic variation components generally can remain uncorrected.

g. After several weeks of successfully operating the first phase, at a minimum satellite spacing of about 1 km, Phase 2 is initiated by reducing minimum spacing to about 200 m, i.e., at a more demanding station-keeping accuracy requirement. The satellites execute the commanded repositioning into the revised formation. This spacing will demand more frequent corrections of the secular perturbations. Otherwise the interaction between the satellites and ground command continue as before.

h. A third phase will be initiated with a still smaller minimum spacing of about 50 m. This close proximity demands more frequent correction maneuvers and very careful ground command supervision, considering the much greater risk of unacceptably close approaches between the satellites.

i. The next phase (Phase 4) is intended to operate under more severe secular and cyclic perturbations, at lower altitude, say 400 km. The satellites must maneuver to this altitude, expending a total of delta-V of 110 m/sec. (The required propellant mass is about 5 percent of the satellite dry mass, an acceptably small extra amount of onboard propellant).

j. All activities described under items (c) through (g) will be repeated in this mission phase, to the extent that unacceptably severe risks of collision due to rapid degradation of the formation pattern can be avoided.

The mission can be extended in Phase 4 if program schedules and available funding allow extra time, and provided the demonstration satellites continue to function successfully. This will permit repeating or extending some steps in the scenario that tend to produce data of highest interest. It also will allow pursuing specific demonstration aspects that either yield unexpected results or promise a clarification of unresolved performance issues.

A trade between increased demonstration benefits—and experience gained—versus the added cost due to extended mission duration should be carefully assessed in this connection.

5. Estimated Development Schedule and Cost of the Demonstration Spacecraft.

Emphasis is to be placed on a spacecraft and mission designed for minimum cost and a short development schedule. The simple design concept, avoidance of unessential subsystem features, and the projected simple operating mode are key elements to making this demonstration mission attractively inexpensive. The miniature spacecraft size generally also will permit a low-cost, piggy-back launch option including several (preferably three) demonstration spacecraft. The discussion includes preliminary cost estimates and a development schedule.

As envisioned at this time, each spacecraft has a wet (launch) mass of 50 kg, 5 kg of which is propellant. The spacecraft mass breakdown is given in Table 1. It is clear that power and structure dominate the spacecraft subsystems, comprising a combined total of 65% of the total dry mass of the vehicle. There is no "payload" in the traditional sense. The key objective is to demonstrate various close formation flying modes and transmit the data back down to the ground for analysis and verification.

Table 1. Spacecraft Mass Breakdown by Subsystem (Ref. 13)

Subsystem	Mass (kg)
Structure	12
Power	17.2
Thermal	2.6
TT&C/C&DH	3.2
ACS	3.5
Propulsion (dry)	3.5
Margin	3
Total S/C Dry	45
Propellant*	5
Total S/C Wet (Launch)	50

* V = 237.64 m/sec, assuming Isp = 230 sec (Mono H propellant)

The spacecraft development cost, 1st unit, 2nd unit, and 3rd unit costs will be calculated from this mass breakdown, using a space mission cost model presented in Chapter 20 of Reference 13.

Table 2. Total System Cost Estimate Less Launch Cost

Program Element	Cost (FY00$K)
3 Spacecraft	10,892
Non-recurring Development	3,606
First Unit*	2,634
Second Unit*	2,371
Third Unit*	2,281
Flight Software (10,000 lines of C)	2,175
Integration, Assembly, & Test	2,399
Program Level Costs	2,691
Ground Support Equipment	412
Launch & Orbital Operations Support	1,053
Total Cost Not Including Contractor Fee or Launch	19,622
Contractor Fee (10%)	1,962
Total Cost Not Including Launch	21,584

*Assumes 95% Learning Curve

We propose a short development schedule, which has the three spacecraft being built and ready to ship to the launch site within 23 months of the program start date (see Figure 7). Based on several small satellite programs conducted in the last 10 years, we feel that this schedule is quite reasonable, especially in light of the fact that no specialized payload element is required. The key cost-driver of the non-recurring spacecraft development cost will be creating formation flying control laws for the various modes to be investigated on orbit. The spacecraft hardware is basically off-the-shelf technology which is space-proven.

law formulation for the key modes to be demonstrated and refining the performance/cost/risk trades. Phase B will finalize formation flying control algorithms and initiate formation flying control and general onboard software development. Also in Phase B, hardware- and software-in-the-loop testbed validation will be carried out for major spacecraft subsystems. Phase C/D will primarily involve building the 3 spacecraft, setting up the chosen ground station with appropriate software and any required additional hardware for processing telemetry data. (We assume use of an existing ground facility with minor modifications.) Phase C/D will also include the final integration, assembly, and test of all 3 spacecraft. Phase E comprises launch and orbital operations for the mission, with a nominal mission life of 10 to 15 months. The total program will last 33 to 38 months.

The anticipated early completion of development is an essential factor in planning and carrying out such an urgently needed demonstration that would precede actual operational formation flying missions which are now being projected by various government and civilian agencies. The benefit will be a timely recognition of practical program obstacles, if any, and help resolve such problems, thereby contributing to further, potential major cost savings.

6. Advanced Technology Options.

Having provided a 3-spacecraft testbed for formation flying experiments, additional, more advanced technologies may potentially be included on the mission, dependent on available funding and schedule constraints.
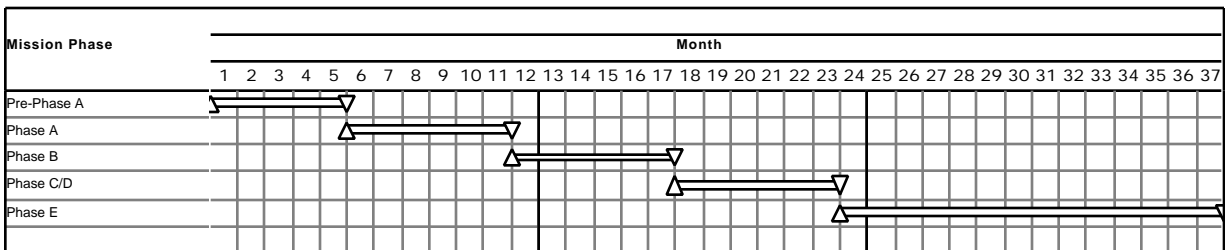

Figure 7. Proposed Mission Schedule for Formation Flying Demonstration Program.

Pre-Phase A will consist of defining the basic mission architecture, laying out the formation-flying performance goals and metrics, proposing a strawman spacecraft design, investigating risk mitigation approaches, evaluating the key performance/cost/risk trade-offs, and selecting a launch opportunity. Phase A will entail detailed spacecraft design, formation flying simulation development/refinement to initiate control

A key problem in the formulation of postulated formation flying experiments is the accuracy of the navigational data, and specifically, the quality of the relative navigation that can be achieved using Commercial Off-The-Shelf (COTS) GPS technology. The key problem to be addressed is that the data must be available in a timely fashion while still retaining suitable

accuracies needed to control formations of sub-kilometer scales.

Before quantifying the current state of the art in relative navigation, it is instructive to first consider what is required of the system. If we take a nominal closest formation spacing of about 100 m separation, from a controls point of view we might require relative position fix accuracies of at least one tenth to one hundredth of this distance, or circa 10 m to 1 m. This also suggests an allowable velocity error level of a few centimeters/second at the very most. This data would need to be made available essentially continuously with an update rate of about 1 Hz. Given this level of state accuracy, a suitable filtering of this data could be expected to improve the raw data by a factor of two to ten. This update rate gives us a about 6000 estimates per orbit, or one full cycle of the relative motion.

A review of the current literature yields insights into a realistic forecast of what may be available for such a dual-differencing GPS system. T. Yunck at JPL [Ref. 14] suggests that two space-based receivers separated by less than 20 km can achieve relative position accuracies of sub-millimeter level routinely and in real time. Yunck goes on to say:

"This is done by tracking the phase of the incoming GPS carrier signal from a single satellite. Each receiver measures the phase of the incoming carriers at the phase center of its receiving antenna. The initial observed phase at the time of signal acquisition will have value between 0 and 2. Thereafter, so long as continuous lock is maintained on the carrier, total observed phase can be continuously accumulated. The L1 phase measurement integrated over 1 second has a typical signal-to-noise ratio of 300. By a standard rule of thumb gives… the resulting phase error of .003 radians, or less than 0.2 mm…"

Yunck suggests that using a dual frequency receiver, carrier phase ambiguity resolution can be achieved "in a matter of seconds" using the pseudorange data that are available by forming the differenced frequency of L1-minus-L2, which has a wavelength of 86 cm. Instrumental noise, ionospheric noise, carrier multi-path, GPS ephemeris error, and dilution of precision all add up to a 3D vector error of around 2 mm. Filtering the data would help remove transient error sources from the data. Thus, with a moderately complex system, we could produce continuous relative navigation data that more than satisfies our 'ballpark' requirements set out above. Simple time differencing of position fixes would more than suffice to produce good velocity estimates of sufficient accuracy.

To take advantage of these data sources, however, we need to communicate immediately and continuously between the spacecraft in the formation. Such a link could be formed using a low-power rf link, as postulated for the spacecraft proposed for this demonstration program. Indeed, such a link between any two spacecraft would allow for the removal of a downlink-uplink system from one of the pair of spacecraft since its 'payload' and housekeeping data could be simply relayed to the second spacecraft over the low-power inter-spacecraft link for later downlinking to ground stations by one assigned spacecraft. This would increase the requirements for the sole Earth-to-space link. Alternatively, data on the carrier phase and pseudo-range could be accumulated independently by each spacecraft and processed on the ground. However, this approach necessitates the spacecraft having a secondary 'collision warning sensor' of some type in any case. Similarly, this approach increases the need for timely and robust communications sessions with ground facilities.

Assuming navigational data of sufficient accuracy can be provided onboard in real time at the required frequency, more complex formation-keeping strategies can be tested. Application of control schemes of graduated levels of model accuracy, (that is simple Keplerian–Clohessy/Wiltshire (Ref. 2) (CW), more complex Der-Danchick (Ref. 4) approaches and yet more complex relative motion schemes currently under development at Microcosm that take into account higher order cyclic and secular perturbations) could be then 'flown off' against one another. Specifically, the traditional approach is shown by application of the CW approach to predicting where a spacecraft should find itself in the near future. Such inputs could then be compared with the incoming navigational data, creating an error signal for a controller to work with.

CW formulations are the simplest approach of the above list and hence provide a reference form with which to test more complex formulations. The CW equations of motion are based on a linearization of the equations of motion and are only valid for circular, or very nearly circular *close* orbits. The Der-Danchick formulation is a linearization that is valid for any general pair of orbits, i.e., non-zero eccentricity orbits. Both of these formulations suffer from the fact that they are based only on motions in a two-body Keplerian orbit system. Work currently underway at Microcosm is attempting to extend the formulation to take into account the motions in a perturbed J_2 system in terms of both secular variations and cyclic variations.

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7. Summary and Conclusions.

The novel mission scenario of having a number of satellites flying in close formation for enhanced visual, radar or geophysical observation purposes, including military and non-military mission objectives, is posing new and potentially major feasibility concerns. This includes the critical issue of performing the mission with relative satellite separation at only a small fraction of a kilometer.

Questions of mission and formation design, spacecraft configuration, operational modes and sequences, and a representative sample scenario were addressed in this paper. Emphasis is placed on achieving a low-cost demonstration mission designed to confirm projected characteristics of subsequent actual mission types, and bring to light potential feasibility and safety issues, considering the risk of operating several satellites at such unprecedented, small separation distances. One key aspect is an underlined early demonstration, as proposed here, to benefit the actual missions now being projected for the near future. The intent of the early demonstration flight proposed here places emphasis on keeping the implementation as simple, practical and risk-free as possible, again with keeping low-cost objectives in mind.

A crucial issue is the degree of autonomy in station-keeping and other functions that is expected of the satellites to be flown in this demonstration. Some autonomy is required because of only intermittent ground contact opportunities, of perhaps only 4 to 6 ground communication contacts available per day. Full station-keeping autonomy involves relatively new technology elements and implies increased complexity of on-board orbit determination and control capabilities. Full autonomy is not projected in the demonstration mission technical approach presented here , for reasons of cost economy and the desired short development schedule. The formation satellites continuously receive GPS signal-derived position and velocity data, which are stored on-board and relayed to the ground at every contact opportunity. Generally, each satellite only has data on its own (absolute) position and velocity from its onboard GPS receiver, and can make small maneuvers autonomously to correct the most significant secular orbital perturbations, e.g., those due to drag.

The ground facility evaluates both absolute and relative satellite motion characteristics and models all cyclic and secular perturbations, to obtain as much orbit dynamics information as possible from this demonstration mission, and to assess lower limits on safe minimum separation distances. Only in case of an impending collision will the satellites themselves be expected to perform evasive maneuvers, since waiting for the next ground contact (occurring perhaps many hours later) could be unsafe. For this purpose, they must carry close approach sensors, the design of which still requires further definition. Their spin-stabilized orientation with the spin-axis pointing normal to the (common) orbit plane helps to simplify the sensing and interactive communication tasks that must be carried out which are involved in performing a timely and effective evasive action. Clearly, these are technology and design issues that require further study and early resolution.

Although a demonstration mission of this kind has not been described in the literature to-date, it appears urgent that this problem be addressed at the earliest time to assess in greater detail its practicality, functional priorities, the full demonstration benefits, and implementation schedule and cost.

Appendix. Relative Motions of Satellites Flying in Close Formation.

A. Basic Keplerian Orbital Motion

A simplified solution of the translational equations of motion for nearly circular orbits was first derived by Hill in 1874. It is based on Keplerian orbit dynamics with the relative motion with respect to a point of the reference circular orbit expressed in linearized form, assuming excursions r from the reference orbit are very small, i.e., $r/R_o << 1$ (See Ref 12). For the assumed formation pattern, with a satellite having departed radially from its reference position in circular orbit, the resulting relative motion with respect to the point of departure is a small ellipse with axis ratio 2 : 1, the major axis being aligned with the local tangent of the circular reference orbit. This is shown in Figure 1 (Section 2) as the motion of satellite S-2 relative to S-1 located at the center of the ellipse. The relative coordinates x,y are the tangential and radial components, respectively. They are expressed by the equations,

$$x = 2 (1 - \cos\ ) (dy/dt)_0/ \qquad (1)$$

$$y = \cos\ (dy/dt)_0/ \qquad (2)$$

depending on the initial radial velocity component $(dy/dt)_0$, with $(dx/dt)_0 = 0$. The independent variable $\ = $ t is the central angle of the reference point (Satellite S-1) at time t, where is the circular orbit's angular rate. Several satellites may be placed in this elliptic relative motion, e.g., two satellites with their respective apogees and perigees 180 degrees apart.
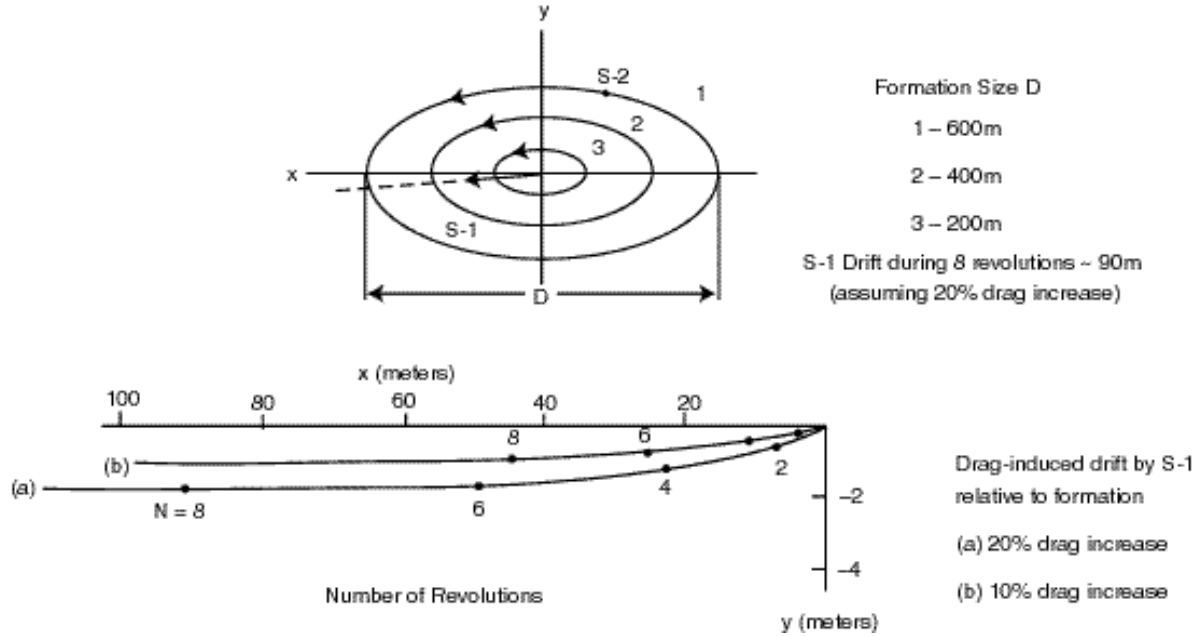
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Figure 8. Differential-Drag Induced Drift of S-2 Relative to Rest of Formation.

A relative motion (z) normal to the reference orbit plane (see Figure 2) is expressed by the equation

$$z = (\sin{}) \, (dz/dt)_0 / \qquad (3)$$

depending on the initial velocity in the z direction, $(dz/dt)_0$. Here the excursion z varies sinusoidally with time, crossing the reference plane twice per orbital revolution. This can be interpreted as a motion that is superimposed on the x, y-motion described by Equations (1) and (2).

In Figure 3 the motion of satellite S-2 is composed of in-plane and out-of-plane excursions relative to the reference satellite S-1 that combine to maintain the fixed distance $(x_{max}/2)$ from it during the entire orbital revolution– a condition that is useful for close inspection of that satellite. This requires the out of plane component z_{max} to be equal to x_{max} $(\sin{})/2$, where $\cos{}$ = 0.5 and therefore, = 60°, as shown in Figure 3.

B. Influence of Non-Keplerian Effects

The principal non-Keplerian influences that modify the basic Keplerian flight dynamics described above include perturbations due the gravitational effects of the non-spherical Earth, luni-solar gravitational influence, solar pressure effects and atmospheric drag. Of these, the perturbations represented by the J_2- term which produces nodal regression (d /dt) and advance of the line of apsides (d /dt) are dominant. They are represented by

$$d /dt = -2.0647 \times 10^{14} \, a^{-3.5} \, (\cos i) \, (1 - e^2)^{-2}$$
$$\text{(deg/day)} \qquad (4)$$

and

$$d /dt = 1.0324 \times 10^{14} \, a^{-3.5} \, (4 - 5 \sin^2 i) \, (1 - e^2)^{-2}$$
$$\text{(deg/day)} \qquad (5)$$

where a is the semi-major axis in km, e is the eccentricity, and i is the orbit inclination. For a 600 km altitude circular orbit of 30 deg inclination, the nodal regression is 6.399 deg/day and the apsidal advance is 10.160 deg/day. (A near-circular orbit of very small eccentricity such as $e < 0.01$ is assumed in the latter case, to interpret the apsidal advance as a motion of apogee and perigee along the circumference).

Neither the nodal regression nor the apsidal advance actually affect the relative positions of satellites in the formation pattern, as long as their respective orbits have identical inclinations. Only the formation as a whole would show the effect of these perturbations. However, if the orbit inclinations are not exactly the same, the small differences in these terms will cause a gradual change in the formation pattern, imposing a secular change in their relative positions.

Other perturbations mentioned above, that also tend to cause such changes, are comparatively much smaller and tend to affect each satellite essentially to the same extent, except any significant differential aerodynamic drag effects. Only if the drag cross sections of all satellites were exactly the same would their relative positions remain unaffected, while each one would experience the same gradual altitude decay. In reality, if

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the satellites have slightly different orientations relative to the air stream, or their appendages are pointing in different directions, at a given time, there will be a differential drag effect causing a secular relative drift.

In the formation flying demonstration, some drag differences are intentionally introduced by extending a pair of small deployable panels on one of the satellites to simulate secular drift, as well as different amounts of this perturbation at a given altitude (see Section 3). The objective is to demonstrate the required correction maneuvers. Changing the altitude from 600 to 400 km leads to an increase of drag by a factor of 15 to 40, depending on the time in the solar cycle. Demonstration of the response to different drag effects will be an important objective of the mission and will increase the value of the data that can be obtained.

The relative drift of one of the satellites with respect to its nearest neighbor in the formation due to an increase in its drag area is derived as an example. It is assumed that the effective drag increase produced by the deployed revolving drag panels (see Figure 8) is 20 percent of that of the nominal satellite's 6-m^2 drag area. The atmospheric density at 600 km altitude is taken as 10^{-13} kg/m^3. The drag deceleration of the nominal configuration is 1.371×10^{-7} m/sec^2, (for a drag coefficient of 4), and the incremental drag a_D due to the deployed panels is set to be 0.274×10^{-7} m/sec^2. Only the incremental drag produces a relative drift with respect to other members of the formation.

To calculate the relative drift an expanded form of the Hill's (or Clohessy-Wiltshire) equations is used which includes an acceleration component a_D in the x-direction. The resulting departure in x and y is given by

$$x_D = a_D \times (8 - 8 \cos - 3 \ ^2)/2 \ ^2 \qquad (6)$$

$$y_D = -2 \ a_D \times (- \sin)/ \ ^2 \qquad (7)$$

where $= 1.083 \times 10^{-3}$ rad/sec is the orbital rate at the formation's 600 km reference altitude. (The drift in x is in the forward direction, that in y is in the downward direction). The resulting drift after 8 revolutions is found to be 89 m in x-direction and 2.4 m in y-direction. This time interval corresponds to a typical non-contact period with the ground command station. Figure 8 shows the drift in x and y-directions for the case of 10 and 20 percent of differential drag. It also shows the drift of the center-satellite S-1 within the formation , previously depicted in Figure 1, for several small formation sizes, the smallest having a 100 m maximum separation in x-direction. This case would be subject to a possible collision of satellites S-1 and S-2, requiring an evasive maneuver.

The results obtained lead to the important conclusion that the drift induced by the assumed differential-drag is sufficiently large to need to be corrected in time. This can be commanded from the ground if the interval between ground contacts is not excessively long, and also depends on the formation size. The alternative of autonomous maneuver control on-board the satellites would require a considerably greater design complexity and cost of the proposed demonstration mission.

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