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SIMULATED FORMATION CONTROL MANEUVERS FOR THE RANGE CUBESAT MISSION

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The Ranging And Nanosatellite Guidance Experiment (RANGE) mission will fly two 1.5U CubeSats in a leader-follower formation, using only differential drag to control their relative separation distance. To prepare for mission operations, a simulation was developed that involved the creation of a high-precision orbit propagation (HPOP) plugin for AGI's Systems Tool Kit (STK) that accounts for rarefied flow characteristics and incorporates a maneuver control system. To evaluate the impact of using the rarefied flow model, various scenarios were run in high and low drag modes using the HPOP propagator, with and without the plugin activated. The difference was significant, showing differences at the kilometer level after several days of simulation. This analysis was compared to real mission positioning data from similar missions by Planet and Aerospace Corp. These comparisons allowed for the determination of an upper and lower bound of expected separation rates for RANGE. This enabled the creation of a series of control maneuvers that will be used to maintain a stable (oscillating) orbit configuration for RANGE, as well as for increasing or decreasing the satellites relative distance within a fixed timeframe.

INTRODUCTION

The control and utilization of small-satellite constellations and formations is particularly challenging due to the resource constraints involved, especially for CubeSats. One example of this is the Ranging And Nanosatellite Guidance Experiment (RANGE) mission, which is a two-satellite CubeSat mission consisting of two 1.5U (1U = 10 x 10 x 10 cm) satellites that will follow each other in a leader-follower formation, with the goal of improving the absolute and relative positioning capabilities of CubeSats. The satellites will have no on-board propulsion system, and will rely on differential drag techniques to control their relative position. While the topic of differential drag has been described in literature [1], there are few missions that have relied upon this method as their sole means of formation control. The RANGE mission hopes to contribute new insights into this topic and prove differential drag for close-range orbital formation-keeping.

To prepare for mission operations, a high-precision orbit propagation (HPOP) plugin for AGI's Systems Tool Kit (STK)[‡] was developed to account for rarefied flow characteristics and a maneuver control system. This paper lays out the design process and verification of the STK plugin and predicts the separation rates of the RANGE satellites based on their orientation. By changing the attitude of the satellites into various drag profiles, different ballistic coefficients are created. These changes in orientation can lead to drastic changes in the separation distance of the two satellites

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over a short time interval. When these satellites are in low Earth orbit the potential drag is small, but accumulates over time. This drag effect can be useful for small satellites at low altitudes, such as the RANGE satellites, because it enables mission planning maneuvers without an onboard propellant system. By making changes in the orientations of the two satellites, the linear distance between them can either be expanded or decreased depending on what the mission requires.

The plugin uses a rarefied aerodynamics database for the RANGE satellites that was generated using the Direct Simulation Monte Carlos (DSMC) method to calculate the expected forces that the RANGE satellite would experience in any orientation [2]. Currently, STK is unable to perform the types of calculations that would enable small satellites to be able to reliably plan for maneuvers using differential drag due to changes in orientation. For the HPOP propagator, STK relies heavily on the user inputs for mass, M, cross-sectional area, A, and the drag coefficient, C_D, to calculate a ballistic coefficient, BC. The primary problem with this model is that the cross-sectional area and the drag coefficient will be constantly fluctuating. STK currently accepts these inputs only as constants which does not account for varying attitude. STK does have a built-in attitude control menu where the changes in orientation could be implemented, but this control method is currently not linked to the force propagator section, so any modifications would not actually affect the required transformation of the ballistic coefficient that would be required. For these reasons, this plugin is an accurate way to predict the behavior of a small satellite without onboard propulsion that will be performing maneuvers by just shifting orientation.

The initial sections of this paper will discuss the creation of the plugin and its integration into STK, to include the comparison of the simulated data from the plugin to positioning data available for Planet's Flock 1c satellites and the Aerospace Corporation's AeroCube-6 satellites. model comparison was done analyzing the results of the plugin against data generated using STK without the plugin while using a constant ballistic coefficient. These results are then used to estimate various differential drag scenarios for the RANGE mission, and the corresponding maneuvers required to increase or decrease the satellite's relative distance. Lastly, the potential for using these data to explore autonomous formation control operations is explored. The experiments were useful in generating reliable upper and lower bounds for separate rates that will be incorporated into RANGE's mission operations plan, to include autonomous control experiments.

METHODOLOGY

Plugin Development

For this plugin, the first step was to adapt the tables of data relating to the force that would be expected on the satellite for any given orientation and altitude. For the function to work, it requires two types of inputs, the direction of the freestream flow in vector format and the atmospheric density at the specific instant of propagation. Obtaining the atmospheric density is simple to do as STK has provided some built-in functions for creating plugins, one of which is the ability to call the atmospheric density that STK is reading at the point of propagation for whichever atmospheric density model that is being used. In this paper, the Jacchia-Roberts model was used as it is among the more accurate models that comes standard with STK.

Getting the direction of the freestream velocity in the body-fixed frame was more difficult. Among the functions that STK provides is one that will acquire the velocity of the spacecraft in an Earth-Centered Inertial, ECI, frame. Since the spacecraft is small enough and has a high enough altitude, the freestream velocity was comfortably assumed to be in the direct opposite direction of travel, i.e. no wind. This vector was then converted from the ECI frame to a local vertical local horizontal, LVLH, frame. Changes in orientation would be modified to have an inverse effect in the change in direction of the freestream flow. With the velocity vector known, it could be incorporated into the data tables function which, when combined with the atmospheric density, would create the aerodynamic force vector. The force would then be returned to the plugin where it would be transformed back into an ECI frame and modified to be expressed as a force on the satellite. STK would then need this converted into an acceleration vector, so the vector would be divided by the mass of the spacecraft.

The primary obstacle to this approach is the control of the spacecraft itself for maneuvers. It has already been mentioned that STK is unable to use attitude controls to direct the HPOP propagator. To work around this problem, a separate function with the plugin was created. Since the plugin was run as an open-loop system for each iteration of the propagator the attitude controls would have to be partially independent. The solution was to have the plugin call a function that would grab all the date and time fields for this iteration of the propagator and would converted into a usable timestamp. A maneuver schedule could then be designed and would have timestamps for when the maneuver was intended to begin. Because the propagator is optimized, the time steps would not always be of the exact same increments. The attitude function that was designed to check the current iteration time against the maneuver list and if the time fell between two specific maneuver times it would perform the maneuver in the earliest time slot. Table 1 is a sample schedule that was created. A maneuver number was created to keep track of the sequence and the time that a maneuver was to be initiated came in the next column. This timestamp was created to be the last two digits of the year, the day of the year, and the hours and minutes of the day in military time. The first entry always needed to be at or before the scenario in STK was started to avoid errors. This would be the default starting orientation of the spacecraft. The last column is for the new orientation angle. The zero-degree orientation angle is set to be in the highest drag mode, where the spacecraft has the largest cross-sectional area normal to the freestream flow. This angle also has the spacecraft in an orientation where the antennae would be nadir facing. An orientation angle of 90° would place the spacecraft in its lowest drag profile, where the solar arrays would be facing away from Earth and the antennae would be on the opposite side of the spacecraft from the freestream flow. The last timestamp maneuver should also be beyond the expected lifespan of the satellite to avoid errors.

When creating this version that the plugin would utilize other factors had to be considered. The biggest change was the slew rate of the spacecraft. Since 90° change in orientations could not be assumed to be instantaneous, the change in orientation had to be implemented incrementally. A drawback of the timestamp design was that it was limited to one minute time step increments. For the case of the RANGE satellites, this fidelity was determined to be adequate and no finer elements were added. The C++ code was completely generated through MathWorks Matlab[§] from the data that would be provided in Table 1. An assumed slew rate of 10° per minute was implemented and Table 2 is the table of data that the plugin would use, once converted to the appropriate C++ code.

Table 1. Sample Maneuver Schedule with a Timestamp and the Desired Orientation in Degrees.

Maneuver Number	Time (YYDDDHHMM)	New Angle (°)
1	171211200	0
2	171211205	90
3	202891200	90

[§] https://www.mathworks.com/products/matlab.html

Table 2. Maneuver Schedule that the Plugin Reads, Taking into Account Slew Rates.

Maneuver Number	Time (YYDDDHHMM)	New Angle (°)
1	171211200	0
2	171211205	10
2	171211206	20
2	171211207	30
2	171211208	40
2	171211209	50
2	171211210	60
2	171211211	70
2	171211212	80
2	171211213	90
3	202891200	90

A limitation of this method was that all the attitude controls needed to be generated in C++ and each maneuver schedule needed to be implemented directly into the hard code of the plugin for each planned scenario. Measures were implemented so that up to two satellite maneuver schedules could be implemented, one for each satellite. The problem currently is that each maneuver schedule needs to be modified in the hard code for each planned scenario. This situation is inefficient and requires each maneuver schedule to be stored for later use, otherwise it will be overwritten.

RANGE SATELLITE MODEL COMPARISON ANALYSIS

A model comparison contrasting the results of the plugin against the outputs of the same satellites using only the HPOP propagator with no plugin and constant C_D, A, and M values was performed. This was done to see how much of a difference in position, drag, and drift rates would be observed between the predictive model using rarefied flow characteristics and the standard STK HPOP model. To perform the model comparison analysis, two RANGE satellites were put into an STK scenario with identical orbital elements. One satellite was placed in the maximum high drag mode, labeled RANGE HD, and one was placed in the maximum low drag mode, labeled RANGE LD. These satellites were propagated using the plugin that had been created. A second set of satellites was added that would have all the exact same characteristics of the first two satellites, but these two would operate without the plugin. To match the characteristics for the orbits, the two satellites that were not using the plugin had to have extra data entered into the force model. The satellite mass was the same as entered before, but the drag coefficient and area to mass ratio were two numbers that each satellite needed to be manually entered into STK when the plugin was not in use. For the plugin, these values are set to zero since all calculations relating to these values is handled within the C++ code.

While it was important to see how far the satellites separated depending on the type of model used, it was more necessary to evaluate the separation rate between the high and low drag satellites in each model and to see if there was any difference. Figure 1 is the linear separation distance between the high and low drag satellites as measured using the left axis over a 48-hour period. While the two plots appear to be very similar, the difference between the two grows to about 1000 meters by the end of the simulation period as observed on the right axis. This means that in a short order of time the distance between the satellites could be very different than the expected result.

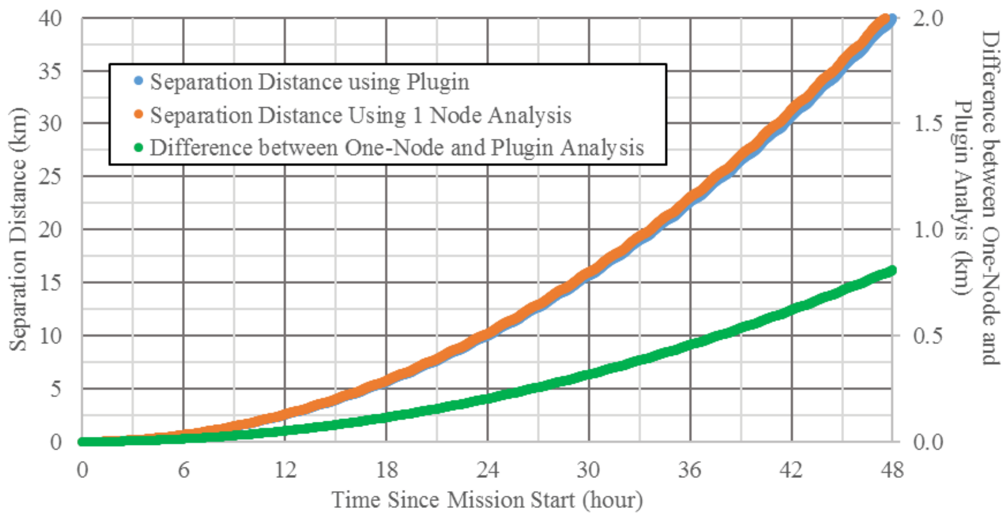

Figure 1. The Left Axis is the Linear Distance between Satellites for Each Mode and the Right Axis is Difference in Distance between these Two Models, Represented by the Green Plot.

RANGE MANEUVER ANALYSIS

To analyze different STK maneuvers, various base orientations were input into the C++ code to find the separation rates under initial conditions. For each scenario, no kickoff force was applied, leaving the increase in linear separation distance to be reliant on only the change in orientation. Figure 2 is a computer-generated image of one of the RANGE satellites with the axis attached. For the base orientation, the x axis is the in-track direction, the y axis is the cross-track direction and the z axis is the nadir pointing direction. This orientation allows the satellites' antennae to be pointing at the earth for maximum earth-bound communication. The limitation of this orientation is that this is the highest drag mode that the satellite will observe. For the purposes of this mission, this orientation is considered the 0° orientation with all attitude rotation being about the y axis.

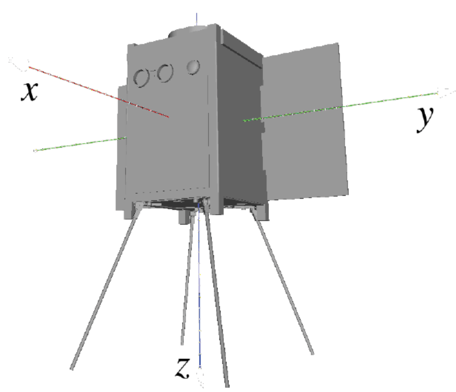

Figure 2. Computer Image of the RANGE Satellite Orientation where x is the In-Track Direction, y is the Cross-Track Direction and z is Nadir Pointing.

To test for acceleration rates one satellite was in a fixed high drag mode orientation and the other satellite's orientation was varied between -90° to +90°. These scenarios were run at 15° increments. Additionally, between -15° and +15° scenarios were run at 3° increments to get a better understanding of the relative acceleration rates between the satellites if only small adjustments were needed. Figure 3 is a plot of the linear distance between the satellites as a function of time for a few of the scenarios run. The data was processed at 1 minute increments since this was the highest fidelity that the plugin program had and was computed using STK's chain tool. The results were as expected, with the greatest change in orientation angle having the greatest separation distance over time. It should be noted that the change in distance can be very fast with the higher change in degrees, seeing a distance of 200 kilometers in just a few days.

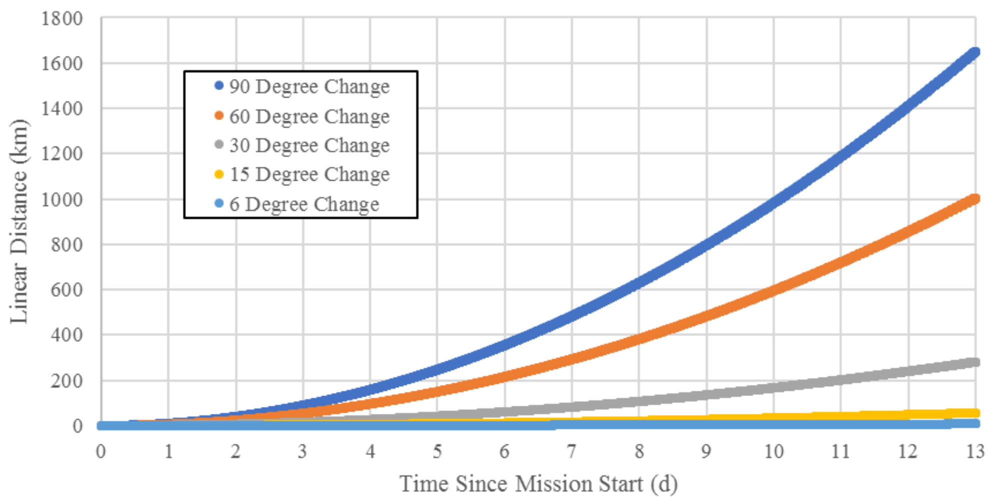

Figure 3. Linear Distance between the Two Satellites as a Function of Time with One Satellite in a Fixed, Maximum High Drag Orientation and the Other Satellite in Varying Degree Changes.

To compute the acceleration rates each scenario was plotted and fitted with a trendline. For each scenario, a second order polynomial fit was nearly exactly matched. The only problem was with the -9° change in orientation, where the high drag satellite was very early on overtaken by the trailing satellite. This cannot be fully explained at this time, though it is theorized that the combination of the drag force and a small downward force due to the angle of attack may cause a slightly faster descent in altitude than the high drag satellite sees purely due to drag. Figure 4 is a plot with the linear separation distance between the high drag satellite and the second satellite in a pure low drag orientation of +90°.

As can be observed in Figure 4, the second order polynomial fit is a very close match to the curve. With this test performed, a second order polynomial curve was generated through linear least-squares regression. Once the equation was known, the second derivative was taken to find the acceleration. This was repeated for each scenario and Table 3 is the relative acceleration rate that was computed for each change in degree. Not included in the table is a 0° change since this output showed no change in distance, as was expected. The maximum relative acceleration rate was at +90° and -90°, with a rate of approximately 2.60 x 10^{-6} m/s^2.

The results of the relative accelerations in Table 3 were then plotted in Figure 5. The relative acceleration rates were found to have greater affect the greater the change in orientation angle with the rates between -15° and +15° having very minimal change in acceleration rate.

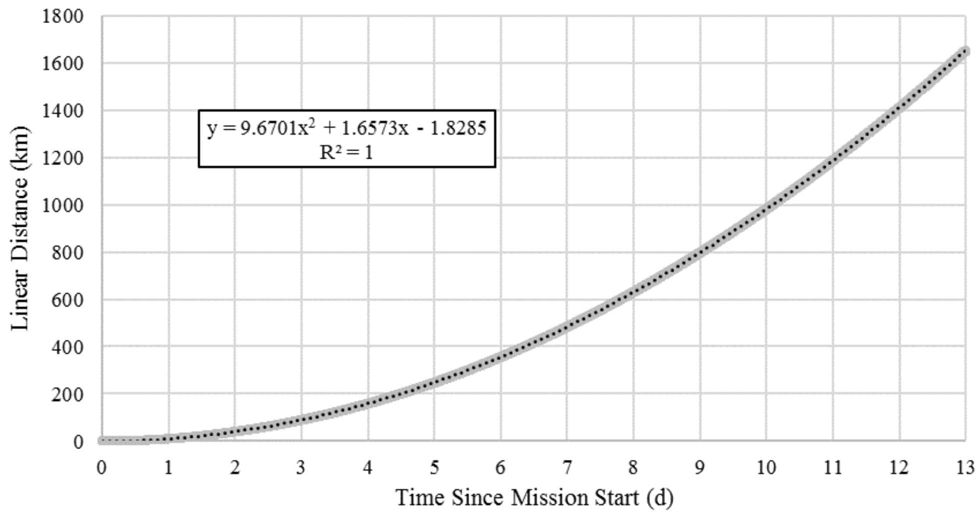

Figure 4. Linear Distance between Two Satellites as a Function of Time where One Satellite is in the Maximum High Drag Orientation and the Other Satellite is in the Maximum Low Drag Orientation.

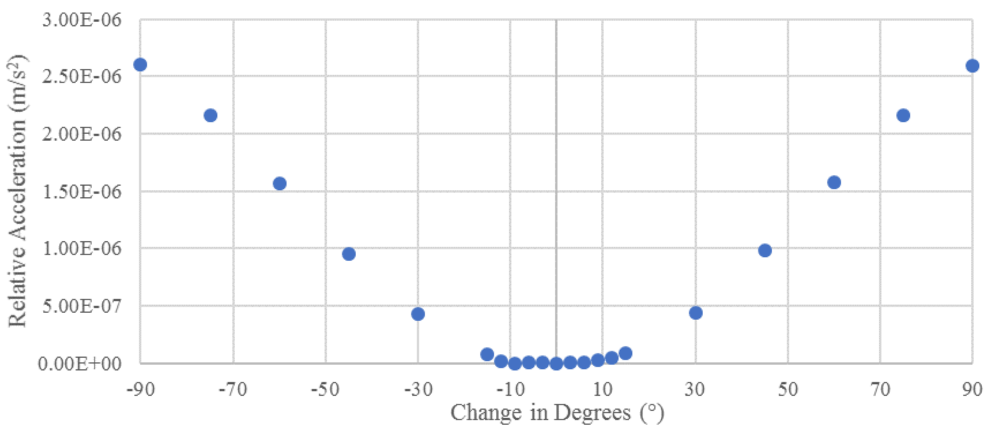

Figure 5. Relative Acceleration Rates as a Function of Change in Degrees from the Maximum High Drag Orientation Angle.

Table 3. Relative Acceleration Rates as a Function of the Change in Degrees from the Fixed High Drag Orientation Satellite.

Change in Degrees (°)	Relative Acceleration Rate (m/s^2)	Change in Degrees (°)	Relative Acceleration Rate (m/s^2)
3	3.12E-09	-3	5.17E-09
6	1.19E-08	-6	4.80E-09
9	2.61E-08	-9	1.04E-09
12	4.91E-08	-12	2.13E-08
15	8.88E-08	-15	7.48E-08
30	4.43E-07	-30	4.29E-07
45	9.78E-07	-45	9.52E-07
60	1.58E-06	-60	1.56E-06
75	2.17E-06	-75	2.16E-06
90	2.59E-06	-90	2.60E-06

COMPARISON OF RANGE DATA WITH OTHER CUBESAT MISSIONS

For verification purposes three different scenarios were evaluated. The first was to examine the expected separation rates between Planet's Flock satellites, which operate at a higher, circular, orbit than the RANGE satellites and used higher ballistic coefficient ratios for their maneuvers. The second scenario was the evaluation of Aerospace Corporation's AeroCube-6 which operated at a higher, slightly oblong, orbit with much smaller ballistic coefficient ratios. By analyzing these two scenarios, an approximation of separation rates could be observed with the theory that RANGE's separation rates should be relatively close.

Planet's Flock Satellite – Analysis

Planet used differential drag by changing Flock satellite orientations to create various drag profiles so that these satellites could achieve different orbital characteristics. Data was gathered primarily from figures Planet has provided for their 1c series satellites. [3] Within "Orbit Determination and Differential-Drag Control of Planet Labs CubeSat Constellations" Figure 8[**] is the theoretical separation distance as a change in Earth-centered relative angle from a reference spacecraft and Figure 9[**] is the actual change with differential drag maneuvers applied. From Figure 8 a baseline of the relative velocity that two spacecraft were making without any drag maneuvers was calculated. Since this would be used to verify the next step, which compares relative velocities after a change in the drag profile, it was important to choose a satellite where these calculations would be the most easily viewed. For this reason, satellite 0905 was evaluated for comparison to the reference satellite 090C. To gauge the separation rate, the slope of the line was calculated. This was done in three sections, each starting when satellite 0905 had a zero-degree difference in relative orbit to satellite 090C. A starting time was approximated and an ending time was chosen to most easily measure a new Earth-centered relative angle. From these approximations, a total time of measurement was taken in seconds. This data is in Table 4.

[**] https://arxiv.org/pdf/1509.03270.pdf, Page 12

Table 4. Calculations of the Change of Degrees from a Set Satellite and Calculations of the Time Period for Satellite 0905.

Flock ID #	Start (°)	End (°)	Total (°)	Start Date	Total Time (Days)	Total Time (s)
0905	0	156.25	156.25	June 20th	43	3715200
0905	0	100	100	Sep 25th	27	2332800
0905	0	100	100	Dec 28th	27	2332800

The next step was to calculate the total distance that satellite 0905 traveled from satellite 090C's frame of reference. In Table 4, the total degree change was noted and this was used to make an approximation for distance. It was given that the Flock satellites would be at an approximate altitude of 620 [4] when their orbit began. This was added to the radius of the Earth and converted to meters and is the radius, R, value used. To find the distance traveled between satellites, the circumference of a circle at that altitude was calculated and was multiplied by the change in degrees over 360. This value was now the total in-track distance covered and was divided by the time previously found to find the relative velocity. The values were then averaged to get closer approximation of the velocity. Table 5 is the total values that were found through these calculations and a relative velocity of this satellite with no orbital maneuvers was found to be approximately 5.20 m/s.

Table 5. Calculation of the Distance Traveled, Relative Change in Degrees and the Relative Velocity of Satellite 0905 to Satellite 090C for the Three Time Periods in the Previous Table.

Flock ID #	Distance Traveled (m)	Relative Change (°/s)	Relative Velocity (Slope) (m/s)
0905	19084084	4.2057E-05	5.1368
0905	12213814	4.2867E-05	5.2357
0905	12213814	4.2867E-05	5.2357
	Average	4.2597E-05	5.2027

The importance of this step was that when the next stage of calculations were made a comparison could be made to the non-maneuver portion to verify that the values were relatively accurate. This would involve a smaller sampling area and the risk of error was increased. Figure 9[††] from the Planet paper shows the orbital spacing of the Flock satellites relative to the 090C satellite. The blue lines are periods where differential drag maneuvers were performed.

Figure 8 and 9[‡‡] were overlaid each other to generate a new image and was then used to best approximate the change in relative distance as a function of time due to an orbital maneuver that changed due to the drag potential. From this image, a change in velocity over a change in time can be used to calculate the acceleration from the differential drag. Again, satellite 0905 was used for this test as it had a long period for its orbital maneuver and, therefore, should decrease the potential for error. It is unfortunately not known to what extent the change in drag was achieved or what the ballistic coefficients during this time were. Table 6 is the calculations that were made to find the distance that would be traveled by a satellite during a time period where both a maneuver was performed and where there was no maneuver performed. From this, the slope of each line was calculated and the difference between the starting and ending Earth-centered relative angles was approximated. This enabled the calculation of a relative velocity for this satellite depending on which mode it was in. Table 7 then shows that value in the difference for the expected distance.

[††] https://arxiv.org/pdf/1509.03270.pdf, Page 12
[‡‡] https://arxiv.org/pdf/1509.03270.pdf, Page 12

This was then divided by the time it took to make the maneuver to give a change in relative velocity. The final step was to divide the relative velocity by the time the maneuver took, which gave a relative acceleration rate of 5.35 x 10^{-7} m/s^2.

Table 6. Calculations of Flock Satellite 0905 During the Same Time Period, Making Comparisons between a Non-Maneuver Trajectory and a Maneuver Trajectory.

Flock ID #	Start (°)	End (°)	Total (°)	Start Date	Total Time (Days)	Distance Traveled (m)	Relative Change (°/s)	Relative Velocity (m/s)
0905	100	156.25	56.25	Jul 16th	16	6870270	4.069E-05	4.9698
0905	93.75	141.625	47.875	Jul 16th	16	5847363	3.463E-05	4.2299

Table 7. Change in Distance and the Relative Distance Rate of Satellite 0905 to Satellite 090C.

Change in Distance (km)	Change in Relative Distance Rate (m/s)	Relative Acceleration (m/s^2)
1022	0.740	5.35E-07

Aerospace Corporation's AeroCube-6 – Analysis

As a comparison study, The Aerospace Corporation's AeroCube-6 satellites were analyzed [5]. At 1U, these satellites are slightly smaller than the RANGE satellites, but also use differential drag to change the relative velocities between satellites. Table 8 is the parameters for this mission. It should be noted that the mass was taken from the AeroCube-4 mission [6], and although the satellites are not identical they are close in size. The big difference between the Flock satellites and the RANGE satellites is the orbit that they travel in. These satellites are in a slightly elliptical orbit traveling between an altitude of 620 to 700 kilometers.

Table 8. AeroCube-6 Characteristic Values. [5]

Mass of AeroCube-4 (kg) [6]	Periapsis Altitude (km)	Apoapsis Altitude (km)	Inclination (°)	Initial Separation Velocity (km/day)
≈1.3	620	700	98	12

One of the other big differences is the ballistic ratio that is being used for the differential drag maneuvers. The satellites have an initial kick-off velocity of 12 meters per second due to a spring-loaded separation mechanism, as listed in Table 8. AeroCube-6 creates differential drag by doing very small changes in its orientation. From The Aerospace Corporations presentation "Flight Results from AeroCube-6: A Radiation Dosimeter Mission in the 0.5U Form Factor" Slide 14[§§] has a plot of the ballistic coefficient ratio as a function of time [5]. They clearly delineate the regions where they were performing differential drag maneuvers. During this time period peaks of up to 1.3 were seen, but the average appears to be at or just below 1.2 for the first maneuver. This was the period that was evaluated for comparison.

Slide 15[***] from this same presentation is another plot provided by the Aerospace Corporation [5] of the separation distance as a function of time and was used in conjunction with Slide 14 for separation acceleration calculations. It appears that the differential drag maneuver was begun

[§§] http://mstl.atl.calpoly.edu/~bklofas/Presentations/DevelopersWorkshop2015/Gangestad_AeroCube-6.pdf, Slide 14, "AC6: Ratio of BSTAR (AC6B/AC6A)"

[***] http://mstl.atl.calpoly.edu/~bklofas/Presentations/DevelopersWorkshop2015/Gangestad_AeroCube-6.pdf, Slide 15, "AeroCube-6: In-Track Formation since July 2014"

around the beginning of August 2014 and continues through early to middle October of 2014. The other key date was when the separation distance reaches its peak point during this first maneuver, which is near the middle of September 2014.

The calculations to find the acceleration rate due to the differential drag is found in Table 9. The process involved the assumption that the velocity at the time the maneuver began was still at a constant 12 kilometers per day. This translates to about 0.13889 meters per second. To calculate the acceleration, the peak separation distance was used and assumes that the separation rate was zero. By taking the time in days it took to achieve this value, the relative acceleration of the second satellite could be found. This value was, approximately -3.57 x 10^{-8} m/s^2.

Table 9. AeroCube-6 Relative Acceleration Calculations.

Initial Separation Rate (km/day)	Initial Separation Rate (m/s)	Time to Achieve Zero Rate of Separation (Days)	Relative Acceleration (m/s^2)
12	0.13889	45	-3.5722E-08

Comparison to RANGE Simulations

To compare the calculations performed for the Flock and AeroCube-6 satellites to RANGE a rough ballistic coefficient approximation needed to be obtained. Since ballistic coefficients during the Flock satellite differential drag maneuvers were not provided, only very rough approximations could be made. Ballistic coefficients for each satellite of this series were provided at a specific moment in time and the maximum and minimum ballistic coefficients were gathered from this list. Since each satellite is of identical configuration then it could be assumed that each satellite could achieve these ballistic coefficients. From this, an estimated ballistic coefficient ratio was computed in Table 10. It should also be noted that when looking at other Flock satellites from different series' that this ratio increases, but since these satellites cannot be verified to have the exact same configuration they were discounted.

Table 10. Maximum and Minimum Ballistic Coefficients for the 09XX Series of Flock Satellites. [3]

Maximum Ballistic Coefficient (kg/m^2)	Minimum Ballistic Coefficient (kg/m^2)	Ballistic Ratio (~)
65.48	16.81	3.896

To calculate the ballistic coefficient for the RANGE satellite Equation 1 was used. The cross-sectional area was calculated based on the satellite's orientation for each specific drag mode and mass was held at a constant 2.25 kilograms.

$$BC = \frac{M}{C_D A} \tag{1}$$

By finding the ballistic coefficient for the two extreme drag modes, a ballistic ratio was computed. Table 11 is the ballistic ratio between the maximum and minimum drag orientations.

Table 11. Ballistic Ratio Calculation for the RANGE Satellites.

Maximum Ballistic Coefficient (kg/m^2)	Minimum Ballistic Coefficient (kg/m^2)	Ballistic Ratio (~)
66.6390	22.1653	3.0065

When comparing this ratio with a theoretical ratio for the Flock satellites, it is found that this max ratio is lower than a possible ratio for Flock. For RANGE, this value is approximately 3 while it is closer to 4 for Flock. Because the ballistic ratios between these satellites are similar a direct comparison for study was deemed acceptable. It should be noted that RANGE is expected to operate at a lower altitude than Flock was designed to operate.

Even though Flock has a likely greater ballistic coefficient ratio, RANGE flies at a lower altitude which allows a slightly faster separation rate due to the increase in atmospheric density. Flock has an acceleration rate of 5.35 x 10^{-7} which is lower than RANGE's max acceleration rate of 2.60 x 10^{-6}, found in Table 3, but is not so different given the drop in altitude.

For the comparison with The Aerospace Corporation's AeroCube-6, it was found that AeroCube-6 had a relative acceleration magnitude of 3.5722 x 10^{-8}, which was also smaller than RANGE's maximum relative acceleration rate. Some important notes, though, is that RANGE operates at a much lower altitude and the calculations for RANGE's acceleration rate had a ballistic coefficient ratio of around 3, as per Table 11. AeroCube-6 is over 120 kilometers higher in altitude at its lowest point with only slight changes in its orientation, which limits its ballistic coefficient ratio to around 1.2. AeroCube-6 is limited to a change in orientation angle of about 30° and when comparing RANGE at a pitch angle of 30°, with an acceleration rate of 4.43 x 10-7 m/s^2 the values become more in line. Given RANGE's lower altitude, it would be expected that RANGE would have a faster acceleration rate, additionally, the 30° limit for AeroCube-6 was its maximum and not a constant, so the lower acceleration rate appears to fit.

From these evaluations, the data generated using the plugin compares well with the limited mission data available for the Planet and Aerospace Corp satellites. This is with full understanding that when actual RANGE satellite data is gathered some small adjustments may need to be made to achieve even more accurate results. The acceleration rates that were computed for RANGE appear to fall close to those for both the Flock satellites and for the Aerocube-6 satellites with slightly faster rates for the RANGE satellites expected due to the lower altitude. Additionally, the values between Flock and AeroCube-6 are close even though they have very different ballistic coefficient ratios and fly at slightly different altitudes.

CASE STUDY

The purpose of the case study was to evaluate a possible maneuver schedule for the RANGE satellites after they finish their detumble and initiate a small kick-off force to separate the two satellites. The starting orbit, pre-separation but post de-tumbling, was set as a circular, near polar, orbit with an altitude of 500 km. A kick-off force would impart an instantaneous velocity change of 1 cm on each satellite in opposite directions along the in-track velocity direction. The orientation of the two satellites would be in the low drag, 90° pitch angle, mode when the kick-off maneuver would be initiated.

While the kick-off force is very low it alters both orbits by just enough to create a difference in the orbital periods of 0.0447 seconds. With a periapsis velocity of 7612.6 m/s, this results in a separation between the satellites through orbital drift of about 340 meters per orbit. At that rate, it would only take about 5 hours for the two satellites to drift beyond 1000 m. To counteract this separation an initial maneuver would be performed. After about 5 minutes to allow the satellites to clear enough distance, the satellite with the longer period changes its orientation to the high drag mode. The high drag mode causes this satellite to drop in altitude faster than the other satellite, causing its orbit to speed up.

Figure 6 is the separation distance between the satellites after the first maneuver was performed. The final planned orientation of the satellites is in the high drag mode where the antennae will be pointing at Earth. Because of this plan, the next step would be to match the orbital periods of the two satellites to stop the separation. Figure 6 also has the difference in the orbital period between the satellites and from this data the periods are relatively close approximately 6 hours after the kick-off separation. At this point the second satellite, the satellite in the low drag mode, slews until its orientation matches the high drag satellite.

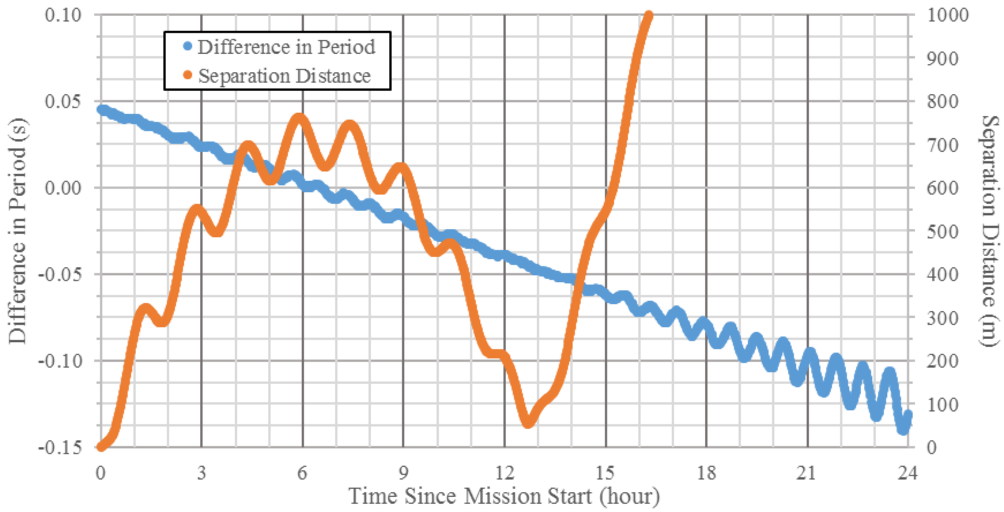

Figure 6. The Difference in Orbital Period and the Separation Distance between the Two Satellites after the First Maneuver.

After this second maneuver the satellites achieve a stable separation distance, as observed in Figure 7. Due to the kick-off maneuver and the fact that the drag forces do not impart instantaneous changes in velocity the final difference in separation and period have some oscillatory behaviors. It should also be noted that with a very small kick-off force, the absolute minimum mean distance between the satellites to stabilize is approximately 730 meters.

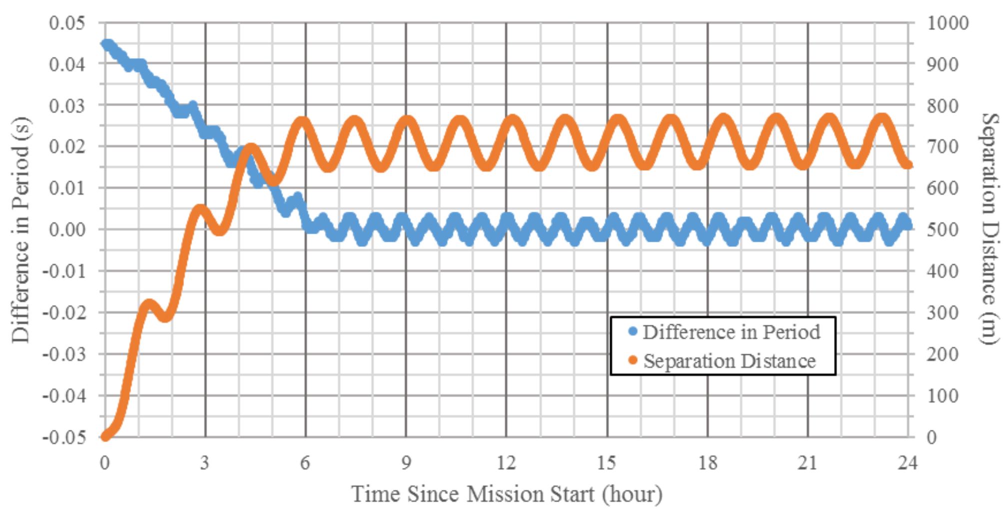

Figure 7. The Difference in Orbital Period and the Separation Distance between the Two Satellites after the Second Maneuver.

The second part of the planned maneuver was to bring the satellites to a closer distance to better facilitate the mission's parameters. To do this, the leading satellite enters a low drag mode causing the trailing satellite to accelerate towards the leading satellite. The challenge with this maneuver is that the trailing satellite will not only close the distance, it will drop in altitude, so much so that even when the satellite hits the desired distance it will have a shorter orbital period and be moving at a faster pace. The trailing satellite would therefore just overtake the leading satellite, even if it was in the same orientation.

The solution to this problem was to have the trailing satellite perform an identical maneuver after a delay period. By mirroring the maneuver, the drop in altitude should be nearly identical for both satellites. There is, ultimately, a small difference in the drop in altitude because the leading satellite will return to high drag mode at a higher altitude than the other satellite, but as long as the maneuver is for brief periods, a few hours at max, then the difference is nearly negligible.

The maneuver would be a two step process for each satellite. The first step would be to enter the low drag orientation and to hold that mode for a set period. At the end of the desired period, the satellite would return to the high drag orientation. After a set delay period, the second satellite would follow the exact same process. This problem leads to two unknown values; the amount of time a satellite should be in the low drag mode and the length of time for the delay maneuver.

A key component to finding the unknown values was to calculate what the separation acceleration between the satellites. To find this, the leading satellite was placed into a low drag mode after the stabilized orbit was achieved. The maneuver was initiated when the difference in periods of the two satellites was close to zero during the small oscillations. Figure 8 is the difference in period and linear distance between the two satellites, just as in Figure 7, but the decrease in distance after the maneuver is evident.

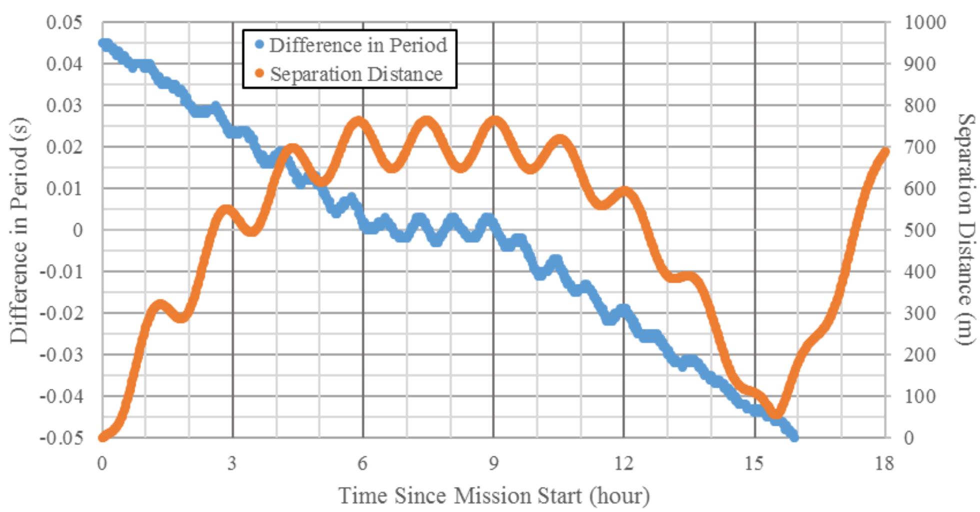

Figure 8. The Difference in Orbital Period and the Separation Distance between the Two Satellites after the Third Maneuver.

To calculate the acceleration, a second plot was created for the period during this phase of the mission before the trailing satellite overtakes the leading satellite. Figure 9 is the linear distance between the satellites and a second order trendline curve fit. When the actual acceleration value was calculated, this was factored in. From the plot, the acceleration between the satellites was calculated to be -2.34 x 10^{-6} m/s^2.

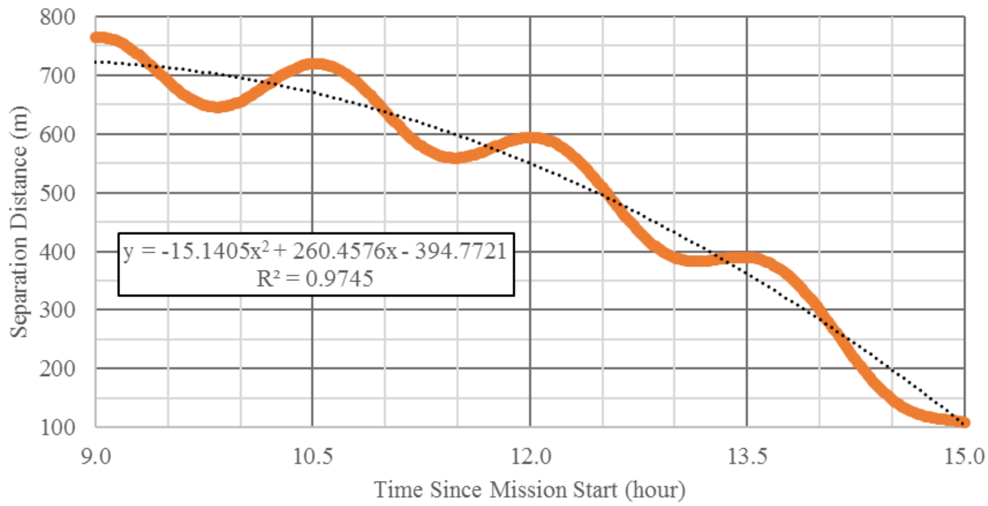

Figure 9. Curve Calculation of the Linear Position as a Function of Time after the Third Maneuver to Calculate the Acceleration Rate of the Separation.

To calculate the time the maneuver should run for, t_M, all that was needed was the desired change in distance, d, and the acceleration rate, a. Equation 2 is the formula that was used to find this value. This calculation was found by integrating the acceleration twice and rearranging it to solve for time.

$$t_M = \sqrt{\frac{2*d}{a}} \tag{2}$$

The other variable that needed to be calculated was the time to delay, t_D. Through testing and verification, Equation 3 was found. This equation is a very simple formula, but gives a first approximation of how long the delay should be. Once the first approximation is tested in STK with the plugin, the delay can be modified, usually only by a few minutes, to find a stable mean distance orbit where the satellites do not drift closer or further away from each other.

$$t_D = \sqrt{\frac{d}{2*a}} \tag{3}$$

In this scenario, using the same value for desired distance and acceleration as in Equation 2, the maneuver delay period was calculated to be 10647 seconds, or about 177 minutes. Combining the delay period with the maneuver time period for the leading satellite, a total maneuver time period of 532 minutes, or little under 9 hours. Figure 10 is the linear distance between the two satellites after this final maneuver process was undertaken. While the outcome did have the two satellites closing the separation and entering a stable distance apart, the actual final mean distance was much closer than anticipated, at about 70 meters apart. This could cause a slight issue in that at the lowest point of the oscillation the two satellites end up about 10 meters apart from each other, which would be closer than desired for safety reasons. The reason for this discrepancy in distance currently requires further testing, but is likely due to other factors, such as the different altitudes that the satellites begin their maneuvers at or possibly from having the maneuvers take place at different times in the orbital period. Secondarily, the oscillations are likely to grow in magnitude every time one of these types of maneuvers is performed. In this instance the peak to peak distance has grown to about 120 meters, so keeping these maneuvers to a minimum would be very prudent.

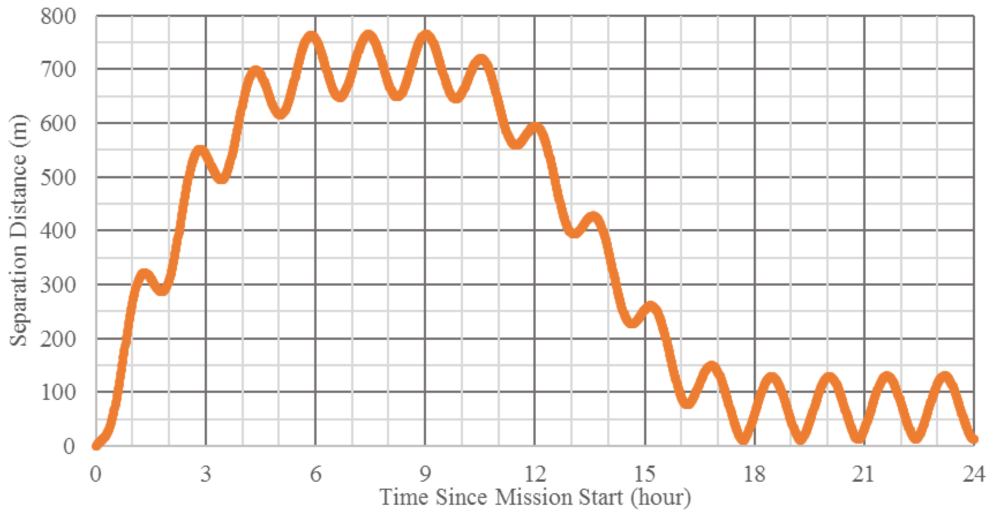

Figure 10. Difference in the Separation Distance between the Two Satellites after the Last Maneuver.

AUTONOMOUS FORMATION FLYING

An additional goal of the RANGE satellites is to incorporate autonomous formation flying without any user input, utilizing only an onboard closed-looped system to maintain stable distances between satellites. Ideally, the spacecraft will be able to evaluate the distance between the satellites and whether they are drifting closer together or apart. The satellites will then be able to make small adjustments to their orientations to correct this drift. The purpose of the autonomous maneuvers is to have the satellites make corrections without the operator having to make these adjustments themselves. This is important because the operators may not have access to the satellites for prolonged periods and if the satellites are drifting during the entirety of this period they may begin to separate to a range that would call for much more drastic maneuvers, which could cause greater oscillations in the separation distance and decay the altitudes of the satellites at a faster rate. Additionally, it increases the chance of error due to over or under correction, which could exacerbate any problem with the satellite's distance. Figure 11 is a flow diagram that the satellite's onboard computer would conceivably follow to utilize autonomous formation flight control. For this diagram, t is the time since the acquisition of GPS positioning data, e is the allowable error, tc is the time constant the operator chooses for the maneuver period, and K is a multiplication constant to adjust the overshoot for the time constant. K is also needed because this process would be updating with each iteration and without K this creates an underdamped system. AT and BT are the orbital periods for satellites A and B, respectively. The 'a' matrix would be a large table of acceleration rates for various degrees of orientation for both satellites at multiple altitudes.

The two satellites should be in near constant contact with each other through the onboard UHF radios, which will transmit the GPS-derived position and inter-satellite range from the laser ranging system. This knowledge should enable the satellites to compute the linear distance between each satellite at any given time. Due to the nature of the orbits, after the separation maneuver there is likely to be an oscillation between the satellites distance. As seen in the previous section, even the initial maneuvers to stabilize the orbits can create oscillations of over 100 meters. Therefore, it is important for the satellites to have a sample period that would include these oscillations in distance and to calculate a mean distance. The oscillation period matches the current orbital period of the satellite, so the sample period should continuously adjust to match. To reduce error, a sample period that could include multiple orbits would be preferred. The smaller the iteration time the higher the fidelity can be achieved, but this will ultimately be limited by the satellite's hardware limitations. The mean distance can be calculated on a rolling basis in order calculate whether the mean distance is changing. A rate of change of the mean distance will be calculated onboard the satellites over a period set by the operators.

If the onboard computers calculate that the rate of change does not exceed the allowable error, then the satellites will not perform any corrective maneuvers. The allowable error should be fairly small so that any drift that is observed does not impact the mission objectives. If the mean distance rate is not within the acceptable range, then the satellites would then calculate an acceleration rate that would work with the chosen time constant by using Equations 2 and 3 in a slightly modified fashion. It would also utilize the previously mentioned 'a' matrix. This matrix would potentially be a small database of acceleration rates for the two satellites at different angles of attack and different altitudes. This would be generated using the plugin and STK to calculate the rates as was done before, though this could potentially have many more increments of angles of attack between -90° and 90°. Secondly, it would not assume one satellite would be in the 0° high drag orientation. Since the satellites will be performing multiple maneuvers and tests, it is conceivable that the satellites may need to perform corrective maneuvers even when at different orientations. This matrix would incorporate all these various pitch angles into the matrix. Additionally, since the acceleration rates are dependent on atmospheric density the matrix would have to have acceleration rates for different altitudes as well. The altitude and pitch angle increments would be decided by the operators based on the time it takes to process them in STK and the onboard storage capabilities of the satellites.

With the acceleration rate needed now computed, it would be multiplied by some constant value. This allows the system to become a critically damped closed-loop system that could be more easily adjusted instead of completely redoing the 'a' matrix should the values need to be adjusted. Since this would be a closed-loop system this value would allow the satellites to make quicker corrections because the system will constantly be updating and without the constant it becomes an under-damped system that could take extremely long periods to correct any drift rates. The next step involves a comparison of the orbital periods between the satellites. Whichever satellite has the lower orbital period will reorient itself to a lower drag orientation based upon the acceleration rate chosen in the 'a' matrix. For example, should the drift rate be non-zero and Satellite A have a larger orbital period with both satellites at a pitch angle of 45°, then Satellite B would adjust its orientation an angle between 46° and 90°, depending on the preselected angle required for the time constant. The reason that the satellite in the lower orbital period goes into the lower drag mode is to extend the life of the mission. The other satellite could enter a higher drag mode with the same affect, but this method would achieve stability at a lower altitude than the previous method. If the satellites are constantly making adjustments it could reduce the satellites' lifespans. Since this is an closed-loop system, these corrections would continuously update, making a more adaptable system. Fundamentally, the autonomous formation flying should allow the satellites to require less station keeping and reduce the amount of communication between the operator and the satellites, thus increasing the amount of communication bandwidth that can be dedicated for the science portion of the mission.

The value of the autonomous formation flight is that the satellites themselves can handle the drift that could appear between the satellites. Drift could be caused by extremely small amounts of unexpected drag due to variations in atmospheric density while performing maneuvers that lead to tiny changes in the orbits of the satellites. Additionally, if there are minor errors in the expectations of the reaction wheels or torque rods the two satellites may not be matching orientations exactly as planned. This slight difference between the expected and real pitch angles can have drastic drift results. Table 12 is the number of days until the satellites exceed 1000 meters in linear separation distance with one satellite in the fixed high drag orientation and the other satellite altered by the listed change in degrees. If the satellite is off its pitch angle by only 15° then the satellite could exceed ideal mission parameters within a little over a day and a half. This also presumes that the two satellites are right next to each other. If the operators are unable to contact the satellites during this period, then the satellites may drift beyond a point of no return.

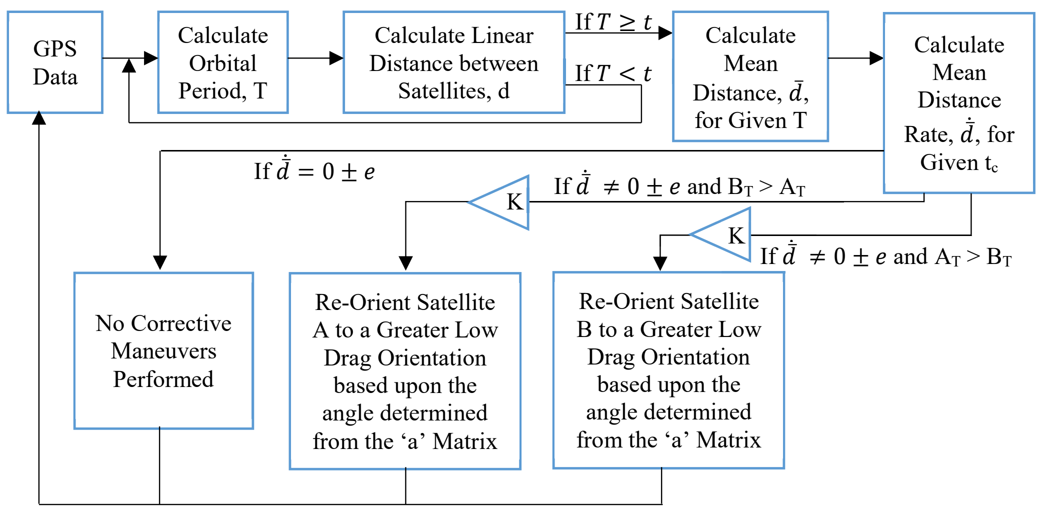

Figure 11. Block Diagram of the Theoretical Procedure Onboard for Autonomous Formation Flight

Table 12. Days for the Linear Distance between RANGE Satellites to Exceed 1000 meters as a Function of the Change in Degrees from the High Drag Orientation for the Second Satellite.

Change in Degrees (°)	Days	Change in Degrees (°)	Days
3	9.144	-3	7.104
6	4.674	-6	7.304
9	3.162	-9	No Value[†††]
12	2.302	-12	3.590
15	1.713	-15	1.896
30	0.763	-30	0.786
45	0.512	-45	0.524
60	0.408	-60	0.412
75	0.349	-75	0.350
90	0.313	-90	0.312

There are some potential problems with this process that would eventually have to be addressed. First, the biggest issue would be from any loss in data acquisition. Since the linear distance is almost guaranteed to be oscillatory, should data at the peaks be dropped, especially for an extended period, then the mean distance between satellites could potentially be unreliable. This could cause the satellites to perform corrective maneuvers for perceived, but non-real, problems. Depending on the

[†††] Exceeded time parameters of recorded data

amount of correction that the satellites perform, it may be necessary for an operator to step in before the satellites spiral out of range. This can be mitigated by extending the period for the evaluation of the drift rate and by limiting the angles of the corrective maneuvers to small changes. A second potential problem arises if the satellites try to perform corrective maneuvers while they are close to each other, raising the potential for an inflight collision. To combat this problem, a minimum safe distance may need to be implemented such that no maneuvers would be performed while the satellites are too close together, or to alter which satellite performs the maneuver. Another potential problem is related to the limits of control of the satellite. Currently the orbital maneuvers only incorporate a pitch angle change and limits the range to ±90°. If the two satellites are near the end ranges they may not be able to go into a greater low drag orientation. To overcome this, a system would have to be in place to enable the other satellite to engage in a maneuver that would push it further towards a high drag orientation. Another possible solution would be for both satellites to perform maneuvers, but this could complicate the results and increase the potential for error. The last potential problem relates to the fact that the predicted acceleration rates are heavily determined by theoretical atmospheric densities that are in STK. Should the satellites enter regions where these densities are vastly different than the expected the acceleration rates could be incorrect, leading to unexpected results. Ideally, this would be overcome through the constant iterations of the process and by potentially tweaking the K value.

CONCLUSION

The findings of this report are that CubeSat's should be able to be flown in reliable formation flying using only differential drag maneuvers. Using STK's HPOP propagator along with the designed plugin that takes into account rarefied flow characteristics the RANGE satellites should theoretically be able to maintain orbits with fairly stable, albeit oscillatory, distances and through changes in satellite orientation be able to alter the distance between them. This was achieved through testing the two satellites in different orientations in STK and plotting the relative acceleration rates that were observed between them. This data was useful in order to plan future maneuver schedules, allowing planning that will meet the mission requirements. It was also useful for the evaluation of the behavior of the satellites after the initial separation was performed using a kickoff force.

The differential drag acceleration rates also appear to be close to both Flock and Aerocube-6's acceleration rates, thus lending credence to the verification process. Additionally, the model comparison analysis showed that while the acceleration rates and distances are close to the plugin, the difference between the constant ballistic coefficient against the plugin's data could have very different results when it is performed in a real world scenario. By utilizing the rarefied aerodynamics database, the goal is to achieve more accurate predictive models for the satellites, thus increasing the lifespan of the mission by keeping them within the limiting distance of the mission parameters.

Ultimately, this data should be able to be used in creating an autonomous formation flight system onboard the satellites. This system would allow the satellites to keep stable distances while not under observation, thus increasing the time that can be dedicated towards the scientific goals and reducing the bandwidth dedicated to orbital station keeping. The value of this system goes beyond the RANGE mission, in that keeping CubeSats in stable formations could have many positive uses.

All of this data is still in the realm of numerical simulation for this paper and the next step would be to gather real world data once the RANGE satellites are launched. When this data is processed it will be compared to the current plugin and the drag forces that were generated using the rarefied flow characteristics model. This model has a built in constant modifier value and can be altered to better fit the real world data. Ideally, this process will provide a more predictable simulation model that can be used with STK to better plan future CubeSat missions.

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NOTATION

a	acceleration rate
A	cross sectional area
A_T, B_T	period of satellite A and satellite B
BC	ballistic coefficient
C_D	coefficient of drag
C_DA_{ref}	drag area
d	desired distance
$\bar{d}$	mean linear distance
$\dot{\bar{d}}$	rate of change of mean linear distance
e	allowable error
ECI	Earth Centered Inertial frame
$HPOP$	High Precision Orbit Propagator
$LVLH$	Local Vertical, Local Horizontal
M	mass
q	dynamic pressure
R	radius to the center of the Earth
ρ	atmospheric density
STK	System Tool Kit
t	time since GPS acquisition
T	orbital period
t_c	time constant
t_D	time to delay
t_M	time interval of the maneuver

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