

## Using Differential Aerodynamic Forces for CubeSat Orbit Control

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### Abstract

This paper focuses on the use of aerodynamic forces for CubeSat orbit control. This would enable fleets of CubeSats to fly in formation without the need for thrusters. To gain a better understanding of the behavior of satellites in orbit, a mathematical analysis of satellites in circular orbits with different altitudes was conducted first. This analysis shows that very small altitude differences could result in comparatively large changes in satellite separation over a reasonable time interval. A numerical-integrator based software simulation was developed to provide a more accurate orbit model and a control algorithm for changing satellite separation. Simulation results indicate that aerodynamic forces in low earth orbit would be strong enough for orbit control, but weak enough that any orbital maneuver would take days to weeks to complete (simulations at 600km altitude). This would allow satellite operators to determine satellite positions using NORAD data and Doppler shift measurements and control satellite configurations as needed from the ground.

### I. Introduction

As technology has progressed, the presence of CubeSats in space has become more and more prevalent. These satellites are generally between 1 unit and 3 units in size, where a unit is considered a 10x10x10cm cube with mass of no more than 1.33kg (a 3 unit or 3U CubeSat will be a 10x10x30cm rectangular prism with mass under 4kg). Due to their small size and relatively low costs in comparison to traditional large satellites, many CubeSats are built by universities or other institutions that would otherwise lack the capability to build larger, more expensive satellites. These CubeSats generally ride as secondary payloads on other larger missions. In the beginning, CubeSats were seen as primarily educational tools with limited scientific value. Now, as electronics have become smaller and CubeSat technology has become more sophisticated, it has become possible to conduct valuable scientific studies and observations using CubeSats. While a single CubeSat may not be able to outperform a larger satellite, CubeSats have the advantage of being able to operate in fleets due to their relatively low costs and sizes. For the price of one large satellite, many CubeSats could be built. For example, the European QB-50 project plans to utilize this concept and launch a fleet of 50 CubeSats to investigate the

lower thermosphere (90-320km altitudes). While the QB-50 project will have a dedicated launch, most other CubeSats that ride as secondary payloads have to meet strict rules and requirements to ensure that the CubeSat will not damage the primary payload. For example, for CubeSats that are launched by NASA as an ELaNa (Educational Launch of Nanosatellites) mission (many US CubeSats), thrusters are not allowed. The lack of thrusters makes CubeSat formation flight difficult. The purpose of this study is to show that the orbits of CubeSats can indeed be controlled without active thrusters, facilitating formation flight. This can be done by generating differential aerodynamic forces on two CubeSats initially along the same low Earth orbit, thereby altering their orbits and producing and maintaining a desired relative separation between the satellites. A CubeSat's drag force is primarily determined by the free stream (ambient) density, satellite velocity, and surface area perpendicular to the velocity direction. By orienting a CubeSat such that the drag forces is greater, the satellite can be made to drop to a lower orbit and consequently have a shorter orbital period than a similar satellite that began in the same orbit, but had less drag. This procedure could be used to increase, decrease, or maintain the separation between two satellites in the same orbital plane. Aerodynamic

lift could also be utilized to facilitate a change in the orbital plane of a satellite. The use of these passive forces eliminates the risks and space/mass requirements that would be associated with a thruster system. All that would be needed would be 3-axis attitude control. This is achievable for many CubeSats. In this study, an analytical analysis of the angular separation as a function of time between two satellites in the same orbital plane but with slightly different altitudes is conducted. The concept of aerodynamic drag effects is then introduced and equations are derived for the calculation of the aerodynamic drag forces. The equations of motion of the satellites are then calculated based on gravitational and aerodynamic forces, and a numerical integrator is utilized to integrate these equations. A control algorithm to produce a desired change in satellite separation is then generated, and its effects are modeled by the numerical integrator. Data from the simulation are graphed and tabulated, and the feasibility of an aerodynamically based orbit control system is demonstrated. In this study, only gravitational forces between the satellite and Earth (both treated as point masses) and aerodynamic drag forces are considered. In reality, there are other forces in space including solar radiation pressure, non-uniformity in Earth's gravitational field, magnetic hysteresis effects, and gravitational effects from the moon, sun, and other planets that will affect the satellites. Since both satellites will be in approximately the same orbit and will travel within minutes, or possibly even seconds, of each other, forces such as gravitational attraction that are independent of satellite orientation will act almost equally on both satellites, and will not cause one satellite's orbit to be more perturbed than the other. Aerodynamic forces and solar radiation pressure both vary depending on satellite orientation, but aerodynamic force outweighs solar pressure below an altitude of about 600km<sup>6</sup>. Additionally, for this simulation, the satellites are only orientated to take advantage of differential aerodynamic forces, so solar pressure will have little effect as far as differential perturbations of the satellites' orbits. For example, solar pressure may decrease the velocity of one satellite more than the other during the first half of an orbit but increase its velocity by the same amount during the second half. Thus, only aerodynamic forces are taken into account in this study. Future work will investigate the effects of other environmental forces, especially solar radiation pressure.

## II. Existing Literature and Expected Contributions

There is much existing literature regarding the effects of aerodynamic forces on satellites. However the majority of this literature focuses on the effects that aerodynamic forces have on orbital decay.<sup>5,7</sup> Research exists regarding the use of environmental forces such as aerodynamic drag and solar pressure for satellite orbit and attitude control, but most of these studies are geared towards larger satellites where a greater degree of precision is required.<sup>1,4,6</sup> There is also some literature regarding the use of aerodynamic forces for CubeSat orbit control,<sup>2,3</sup> but most of these studies are purely theoretical. The NanoSail-D CubeSat mission was launched in November 2010 to investigate the effects of a deployable drag sail on orbital decay, but this mission consisted of only a single satellite and there was no way to reduce the drag once the sail was deployed. The goal of this work is to provide a software simulation that will allow for the estimation of the effects of aerodynamic forces on CubeSats as well as an orbit control algorithm for the satellites. The results of these simulations will serve as groundwork for a mission to test the use of aerodynamic control algorithms for managing satellite fleets in space. The simulation code will be compiled into a software suite that will be useable by others without programming experience.

## III. Orbital Analysis

Earth's gravitational force ( $F_g = \frac{GMm}{r^2}$ ) is by far the most powerful force that acts on a satellite in Earth orbit. From this, equations governing satellite orbits can be derived. A basic analysis of circular orbits was conducted first to gain a better understanding of fundamental orbital mechanics concepts and to provide preliminary numerical data regarding the behavior of satellites with different orbital conditions (caused by aerodynamic forces). In a circular orbit of radius  $r$ ,

$$\text{translational velocity} = v = \sqrt{\frac{GM_E}{r}} = \sqrt{\frac{\mu_E}{r}} \quad (1)$$

The period of the circular orbit is given by

$$T = \frac{2\pi r}{v} = 2\pi r \sqrt{\frac{r}{\mu}} = 2\pi \sqrt{\frac{r^3}{\mu}} \quad (2)$$

In the general case of an elliptical orbit, the orbital period is given by

$$T=2\pi\sqrt{\frac{a^3}{\mu}} \quad (3)$$

where “a” is the semi-major axis of the orbit. The derivations for these equations can be found in any basic Orbital Mechanics book. These equations show that as the altitude of an orbit is decreased, the translational velocity increases and the orbital period decreases. Thus, if two satellites are on the same orbit but are separated by a given distance and a satellite distance closure is desired, the chasing satellite must be brought to a lower altitude in order to catch up to the leading satellite. The leading satellite must then be brought down to the same altitude as the chasing satellite once the maneuver is finished to maintain the desired separation. To calculate the change in angular separation with respect to time between two satellites in circular orbits with different radii (which would result when the satellite with more drag drops more quickly than the other), we must compute the difference between the mean motions of the satellites (average change in true anomaly with respect to time). Mean motion (n) is calculated by

$$n=\frac{2\pi}{T}=2\pi\left(\frac{1}{2\pi}\sqrt{\frac{\mu}{r^3}}\right)=\sqrt{\frac{\mu}{r^3}} \quad (4)$$

The rate of change of satellite angular separation ( $\varphi$ ) with respect to time is given by

$$\frac{\Delta\varphi}{\Delta t}=n_2-n_1=\sqrt{\frac{\mu}{r_2^3}}-\sqrt{\frac{\mu}{r_1^3}}=\sqrt{\mu}\left(\frac{1}{r_2^{3/2}}-\frac{1}{r_1^{3/2}}\right) \quad (5)$$

If we set  $r_2 = r_1 + \delta$ , where  $\delta$  is the difference in the orbital radii of the satellites, the term

$$\left(\frac{1}{r_2^{3/2}}-\frac{1}{r_1^{3/2}}\right) \text{ becomes } \left(\frac{1}{(r_1+\delta)^{3/2}}-\frac{1}{r_1^{3/2}}\right) \quad (6)$$

The term  $\frac{1}{(r_1+\delta)^{3/2}}$  can be written as  $r_1^{-3/2}\left(1+\frac{\delta}{r_1}\right)^{-3/2}$ .

Since  $\frac{\delta}{r_1} < .001 \ll 1$ , The binomial expansion theorem can be used to carry out the simplification

$$r_1^{-3/2}\left(1+\frac{\delta}{r_1}\right)^{-3/2}=r_1^{-3/2}\left(1-\frac{3\delta}{2r_1}\right)=\left(\frac{1}{r_1^{3/2}}-\frac{3\delta}{2r_1^{5/2}}\right) \quad (7)$$

where only the first term in the expansion is considered significant. Inserting this into the previous equation for  $\frac{\Delta\varphi}{\Delta t}$  gives us

$$\frac{\Delta\varphi}{\Delta t}=\sqrt{\mu}\left(\frac{1}{r_1^{3/2}}-\frac{3\delta}{2r_1^{5/2}}-\frac{1}{r_1^{3/2}}\right)=\sqrt{\mu}\left(\frac{-3\delta}{2r_1^{5/2}}\right) \quad (8)$$

where  $r_1$  is the orbital radius of the satellite with the lower altitude. We will assume this satellite is behind the satellite with orbital radius  $r_2$  and is trying to catch up to it. Since  $r_1 < r_2 < 1.001r_1$  we can consider the above equation to be

$$\Delta\varphi=\sqrt{\mu}\left(\frac{-3\delta}{2r_1^{5/2}}\right)\Delta t \quad (9)$$

Below is a MATLAB plot of this function for different values of altitude difference ( $\delta$ )

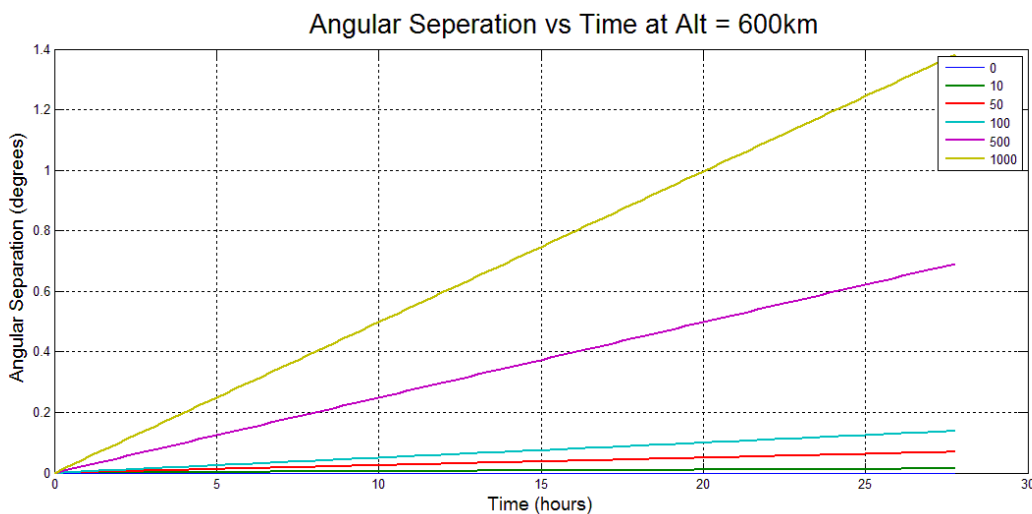


Figure 1

Note that one degree corresponds to about 120km at an altitude of 600 km (Orbital radius = 6978km). In the

course of a day, two satellites in circular orbits with an altitude difference of 1 km will change their relative

separation by 143km. This rate is a good deal more than will be necessary for the types of maneuvers discussed in this study. This preliminary analysis provides us with a general sense of some of the physics behind satellite orbital mechanics and provides groundwork for the writing of a more accurate software simulation. One conclusion we can draw from the above data is that it is necessary to reduce the orbital radius of the chasing satellite in order for it to catch up to the leading satellite, and that very small altitude changes are enough to make substantial changes in satellite separation over reasonable time intervals.

#### IV. Aerodynamic Effects

This research hinges on the use of aerodynamic forces to induce satellite altitude changes. Orbital maneuvers that change an orbit's semi-major axis operate on the principle of adding or removing energy from the orbit. The specific energy of an orbit is given by

$$E = \frac{v^2}{2} - \frac{\mu}{r} = \frac{-\mu}{2a} \quad (10)$$

Where "a" is the semi-major axis of the orbit. Solving for "a" gives

$$a = \frac{-\mu}{2E} \quad (11)$$

Note that the energy will be negative for elliptical orbits. Aerodynamic drag force will act parallel but opposite in direction to the velocity vector and thus will reduce the energy of the orbit (make it more negative) by an amount equal to the magnitude of the work done by the drag force. This, in turn, decreases the value of "a" which decreases the orbital period ( $T=2\pi\sqrt{\frac{a^3}{\mu}}$ ). An analogous orbital maneuver is the traditional Hohmann transfer. A Hohmann transfer allows a satellite to transfer to a higher or lower orbit by adding or removing energy from the orbit. To transfer to a higher orbit (larger "a"), energy is added (forward thrust), while transferring to a lower orbit (smaller "a") requires the removal of energy (retro-thrust) as shown in figures 2 and 3.

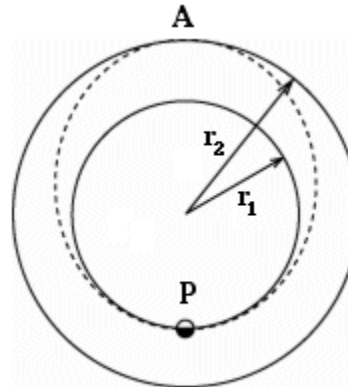


Figure 2

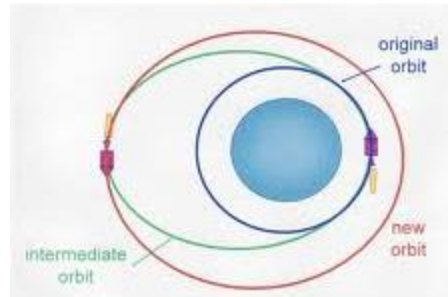


Figure 3

Aerodynamic drag can be viewed as essentially an ongoing retro-thrusting maneuver that constantly removes energy and transfers the satellite to an increasingly lower orbit. This continuous force will cause the satellite to travel in a spiraling trajectory towards the center of Earth. This effect is exaggerated in figure 4.

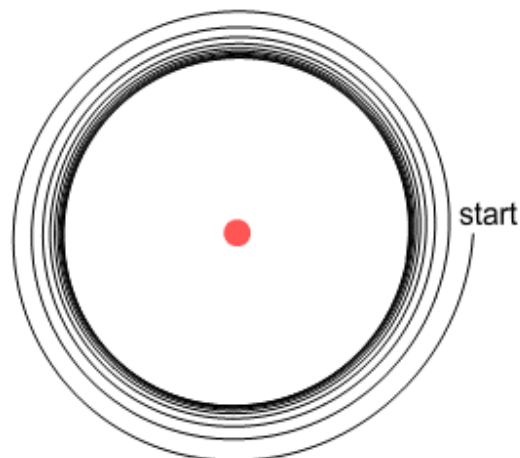


Figure 4

In reality, the change in orbital radius after each orbit is very small (generally less than ten meters per orbit) because aerodynamic forces are relatively weak. Thus, this spiral can be approximated by a series of circular orbits with steadily decreasing radii. Furthermore,

elliptical low Earth orbits will tend to naturally circularize due to increased drag at the orbit's perigee (caused by increased atmospheric density), adding validity to our initial analysis of circular orbits.

### Calculating Aerodynamic Drag Force

The next step is to calculate the aerodynamic drag force. This force is given by the equation

$$F_d = \frac{1}{2} C_d A_{\perp} \rho_{\infty} v_{\infty}^2 \quad (12)$$

Where  $C_d$  is the object's drag coefficient,  $A_{\perp}$  is the area of the object perpendicular to the velocity vector,  $\rho_{\infty}$  is the free stream (ambient) density, and  $v_{\infty}^2$  is the free stream velocity (velocity of object relative to airstream).  $C_d$  is possibly the hardest value to determine in this equation since it is dependent upon the geometry and orientation of each individual structure as well as on flow characteristics, but a value of  $C_d = 2.22$  is often used to model drag on satellites with reasonable accuracy<sup>8,9</sup>. This value has been calculated based on aerodynamic modeling and empirical observation, and its calculation will not be treated here. A more accurate  $C_d$  value would need to be calculated depending on the specific mission at hand. Since we are concerned primarily with the acceleration induced by the aerodynamic force rather than the value of the force itself we can use the knowledge that  $F=ma$  (Newton's second law) to calculate acceleration due to drag as

$$a_d = \frac{F_d}{m_s} = \frac{C_d A_{\perp}}{2m_s} \rho_{\infty} v_{\infty}^2 = C_B \rho_{\infty} v_{\infty}^2 \quad (13)$$

It is convenient to define  $C_B$  (the ballistic coefficient) in order to have a single coefficient that handles satellite geometry, mass, and orientation. Drag force and acceleration act parallel to the velocity vector, so in vector form,

$$\vec{a}_d = a_d \hat{v}_{\infty} = C_B \rho_{\infty} v_{\infty}^2 \hat{v}_{\infty} \quad (14)$$

Note that  $C_B$  and  $a_d$  for two satellites in an identical orbit (same velocity and density) are determined by the satellite geometry and perpendicular area to mass ratio. If two satellites have the same general geometry, then the difference in  $a_d$  will be due solely to the value of  $\frac{A_{\perp}}{m}$ .

Thus, it is possible to have two satellites of different masses that will experience the same drag effects in a given orbit. Atmospheric density is another component of this equation that is often difficult to model. For any given altitude, density can vary by up to two orders of magnitude based on solar and geomagnetic activity as shown in figure 5.

**JB2008 mean air density with altitude for low, moderate, and high long- and short-term solar and geomagnetic activity**

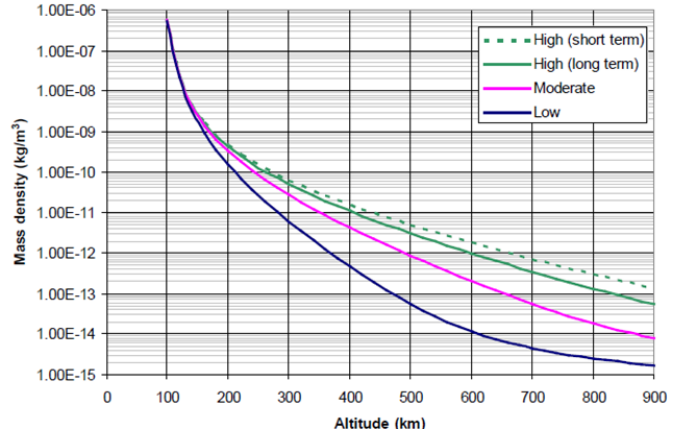


Figure 5

CubeSats generally have the requirement to de-orbit within 25 years and have minimum orbital lifetime requirements dependent on specific mission at hand. As shown by figure 5, air density decreases exponentially with altitude. Thus, since drag force increases linearly with density, orbital lifetime will increase exponentially with altitude as shown by figure 5 below.

**Orbit Lifetime vs Initial Circ Orbit Altitude (CdA/m=200 cm²/kg; Equatorial Orbit)**

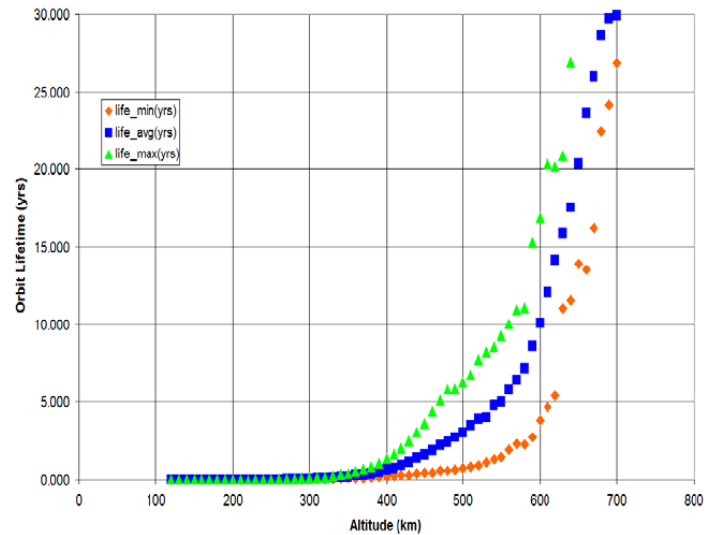


Figure 6

The following calculations will assume an initial circular orbital altitude of 600km. This altitude is ideal because it allows for a de-orbit within 25 years and provides the satellite with an orbital lifetime long enough to conduct almost any mission. Also, based on atmospheric density, aerodynamic forces at 600km will be strong enough to generate a meaningful change in

the relative position between two satellites, but will be weak enough that this change will occur slowly (days or weeks). This would enable the satellite operators to track satellite positions and update the control algorithms as needed from the ground through the course of any maneuver.

### Equations of Motion

To provide an accurate model of how a fleet of satellites will behave in the presence of a real atmosphere, it is necessary to use a software algorithm to numerically integrate the satellites' equations of motion. A total acceleration equation is first derived by summing the gravitational and aerodynamically induced acceleration vectors. Gravitational acceleration acts along the radius vector (from the satellite to Earth's center) and is given by the equation

$$\vec{a}_g = \frac{F_g}{m_s} \hat{r} = \frac{GM}{r^2} \hat{r} = \frac{GM}{r^3} \vec{r} \quad (15)$$

$$\vec{a}_T = \vec{a}_g + \vec{a}_d = \frac{GM}{r^3} \vec{r} + C_B \rho_\infty V_\infty \vec{v}_\infty \quad (16)$$

This total acceleration equation must be integrated twice with respect to time to obtain equations for  $\vec{r}$  and  $\vec{v}_\infty$  as functions of time. No closed form solution for this integral exists, so the acceleration equation must be integrated over a series of time-steps using a numerical integrator to generate a position vector for a given time based on initial  $\vec{r}$  and  $\vec{v}_\infty$  conditions. This is done using the Runge-Kutta fourth order integration algorithm.

### V. Java Simulation

A software suite was written in Java using the NetBeans integrated development environment to model and display the satellites' behavior in space. This program required the initial position and velocity vectors of two satellites as well as atmospheric density and  $C_B$  values as inputs. These arguments were sent to a Runge-Kutta fourth order integration algorithm to calculate the resultant position and velocity vectors after a one second time step. Since a CubeSat fleet would most likely consist of two satellites launched from the same launch vehicle, these simulations modeled cases of two satellites that were in the same orbit with a 2000km initial separation between them. Since the satellites would be at roughly the same altitude and since the changes in altitude would be very small for any maneuver ( $\delta < 1\text{km}$ ), the assumption was made that the atmospheric density experienced by each

satellite would be roughly the same and would be constant throughout the maneuver. This method yields reasonable accuracy, but density changes with altitude would need to be taken into account and could easily be supplied to the integrator if a more accurate model is required. For this particular simulation, a model of two 3kg, 3U CubeSats, each with two 10x30cm single deployed long-edge type solar panel arrays was used.

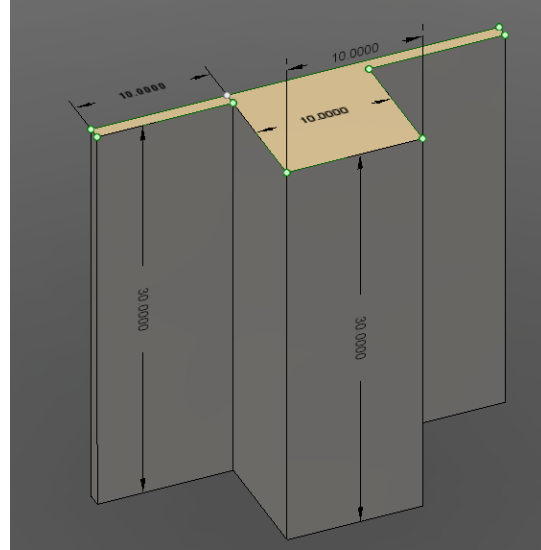


Figure 7

For the purposes of this model, the satellites had two possible orientations; minimum drag, and maximum drag. The maximum drag option would have the large 9U face (30x30cm) perpendicular to the velocity vector. In this configuration, the exposed area would be  $.09\text{m}^2$  and  $C_B$  would be calculated by

$$C_{B_{\max}} = \frac{C_d A_\perp}{2m_s} = \frac{2.2(.09\text{m}^2)}{2(3\text{kg})} = .033 \frac{\text{m}^2}{\text{kg}} \quad (17)$$

For the minimum drag configuration, the 1U top face (10x10cm) would be facing forward (The panels will be very thin ( $<3\text{mm}$ ), so they would not contribute much to the exposed surface area). Thus,  $A_\perp \approx (.1 \times .1)\text{m} = .01\text{m}^2$ , and

$$C_{B_{\min}} = \frac{C_d A_\perp}{2m_s} = \frac{2.2(.01\text{m}^2)}{2(3\text{kg})} = .0036 \frac{\text{m}^2}{\text{kg}} \quad (18)$$

As shown by the  $C_B$  values, for a given density and velocity, a satellite in maximum drag configuration would have approximately nine times the drag of a satellite in minimum drag configuration. The software simulation allows drag coefficients to be changed during the simulation and includes an autonomous control algorithm for the satellites. The control algorithm works as follows:

1. The user enters a desired satellite distance closure value in km.
2. The  $C_B$  of the satellite in front is changed to its minimum value, and the  $C_B$  of the chasing satellite is changed to its maximum value. This makes the chasing satellite fall faster than the leading satellite, allowing it to catch up to the leading satellite.
3. When half of the desired distance has been closed, the satellite orientations are switched; the leading satellite is set to its maximum drag coefficient value, while the chasing satellite is set to its minimum drag coefficient value. This allows the leading satellite to fall faster, bringing it closer to the altitude of the chasing satellite.
4. When the change in satellite angular separation over time  $(\frac{d\phi}{dt}) = 0$  (satellites at the same altitude), both satellites are set to the

same minimum value of  $C_B$ . If the density values are assumed constant, the ascent and descent paths of the satellites will be symmetrical, and the satellites should arrive at equal altitudes at the same time as the desired distance closure is achieved.

Satellite and orbital characteristics for each satellite are displayed by the software while the simulation is running. These include velocity, orbital radius, true anomaly (angle from the horizontal), satellite angular separation, time, satellite separation in km, and satellite drag coefficient. Several tests were conducted using this simulator assuming two 3U 3kg CubeSats in initial 600km altitude circular orbits (orbital radius = 6978 km), and the results were plotted using the software's built in graphing utility.

## VI. Results

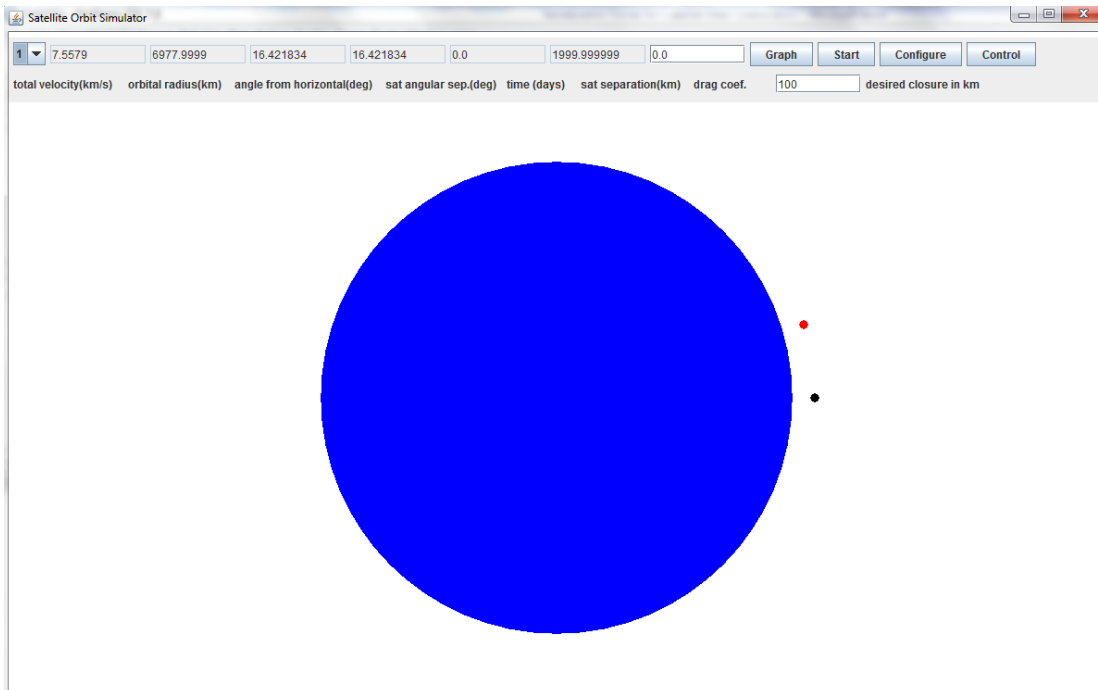


Figure 8

Above is a screen-shot of the orbit simulator. The blue circle represents earth, while the red and black dots represent the leading and chasing satellites respectively (satellites 1 and 2). The satellites are traveling in a counter-clockwise orbit in this simulation. Velocity, orbital radius, true anomaly (angle from horizontal),

and ballistic coefficient are shown for the selected satellite. The time and angular and distance separations between the satellites are also shown. The satellites in this simulation begin 2000km apart. The user can choose to either enter custom ballistic coefficients for each satellite and press the start button to initiate the

simulation, or can input the desired closure in km and press the control button. This activates the autonomous control algorithm discussed earlier using the minimum and maximum  $C_B$  values calculated above. All display fields as well as the graphic simulation update in real time while the simulation is running. The simulation can be paused and at any time by pressing the start/pause button. Ballistic coefficients can then be manually changed, a new control algorithm can be initiated, or the user can press the graph button to view

graphs of orbital radius, altitude separation, and angular and distance separation of the satellites over time. In this simulation, a control algorithm to close the distance between the satellites by 1000km was initiated first. As soon as this was completed, another control algorithm was implemented to close the distance by another 500km. Distance closures of 250km, 100km, 50km, 25km, and 10km were conducted next. The figures below show the results of these control algorithms (except the 25km and 10km closures).

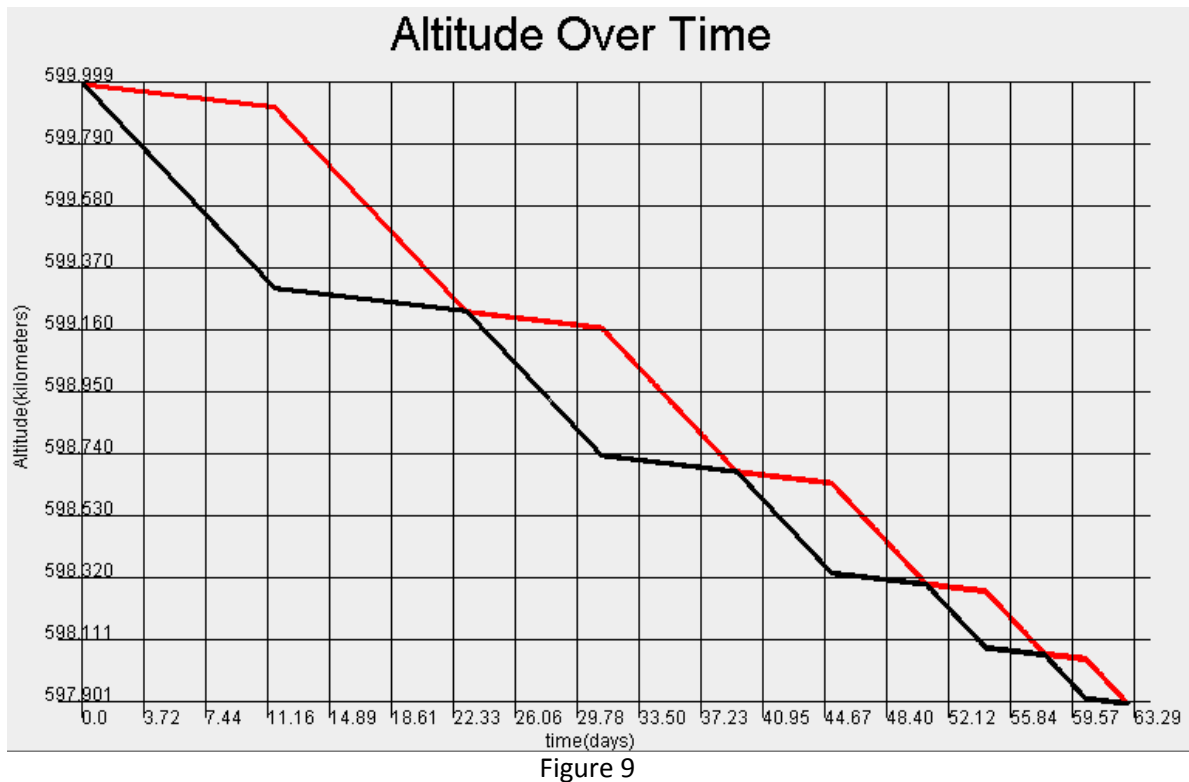


Figure 9 shows the altitude of each satellite over the course of the control algorithms. The red line represents the red satellite (leading satellite) and the black line represents the black satellite (chasing satellite). Each intersection point between the two lines represents the completion of a control algorithm (the satellites end up at the same final altitude so that  $\frac{d\theta}{dt} \approx 0$ ). We see that during each control algorithm, the black line is always below the

red line. This makes sense because the chasing satellite must be at a lower altitude if it is to catch up to the leading satellite. If it were desired that the satellites increase their separation by a given amount, the red line would be where the black line is and the black line where the red line is (assuming same initial orbital conditions). The slopes of the lines and the times required would be identical to those for the distance closure algorithms shown above.



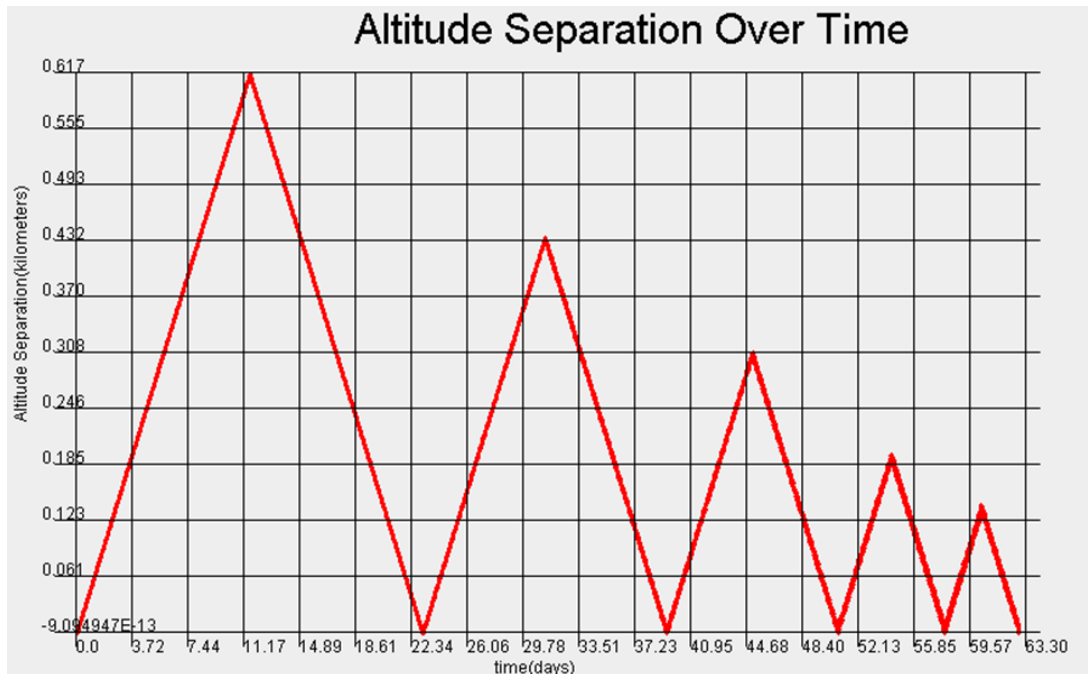


Figure 10

Figure 10 is essentially the difference between the red and black curves in figure 9. It provides a better visual representation of the altitude separation between the satellites over time. Each triangle represents a complete control algorithm (1000, 500, 250, 100, and 50 kilometers of closure). The first half of the triangle (positive slope) is where the trailing satellite has

maximum drag, and the leading satellite has minimum drag. The apex of the triangle is where the drag configurations are switched (leading satellite has max drag, while chasing satellite has min drag) in order to bring the satellites to the same altitude by the end of the control algorithm.

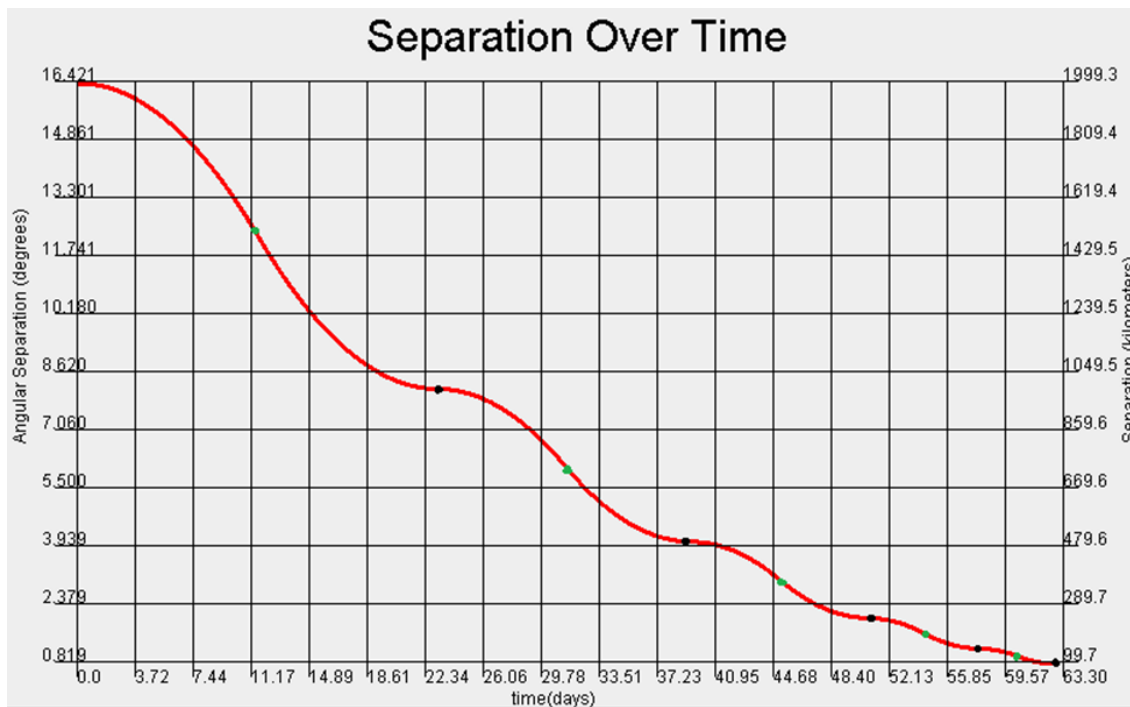


Figure 11

Figure 11 shows the separation between the satellites in both angular and distance terms. The black dots on the graph mark the ends of each control algorithm. They correspond to the intersections between the red and black curves on figure 9. Note that at the end of each control algorithm, the separation between the satellites remains roughly constant. In this simulation, a new control algorithm was initiated as soon as the original

one was finished, but if the simulation were left to run after all control algorithms were complete, the slope of the line (changes in separation over time) would be zero. The green dots on the graph (inflection points) represent the points at which the drag configurations of the satellites are swapped. These points correspond to the tips of the triangles in figure 11.

Desired Distance Closed(km)	Time required(days)	Total Altitude loss(meters)	Max Altitude Difference (meters)	Altitude ( $\delta$ )	Estimated Closure (km) Based on $\delta$ and $\Delta\phi$ Eq.
1000	23.073	771.00	617.36		999.747
500	16.319	545.10	436.63		500.097
250	11.534	385.47	308.70		249.899
100	7.263	242.08	194.74		99.270
50	5.168	172.78	137.81		49.986
25	3.628	121.01	97.06		24.715
10	2.286	76.12	61.18		9.816

Figure 12

Figure 12 displays the information graphed in figures 10, 11, and 12 in tabular form. This shows the specific numerical values associated with each control algorithm. The average altitude difference ( $\delta_{avg}$ ) and the equation

$$\Delta\phi = \sqrt{\mu} \left( \frac{-3\delta}{2r^{5/2}} \right) \Delta t \quad (19)$$

were used to calculate the change in angular separation resulting from satellites in circular orbits with altitude differences  $\delta_{avg}$ . The average value theorem states that the average value,  $A$ , of a function  $f(x)$  over the interval “ $a$ ” to “ $b$ ” is given by the equation

$$A = \frac{1}{b-a} \int_a^b f(x) dx \quad (20)$$

Thus  $\delta_{avg}$  can be computed by

$$\delta_{avg} = \frac{1}{\Delta t} \left( \frac{1}{2} \Delta t \delta_{max} \right) = \frac{\delta_{max}}{2} \quad (21)$$

The change in distance between the satellites can be found by multiplying average orbital radius by the change in angular separation found by solving the  $\Delta\phi$  equation using  $\delta_{avg}$ . This data was calculated and tabulated for comparison with the data obtained by the numerical integrator. As the table shows, the values from this equation differed from the values provided by the integrator by generally less than 2%, showing that

the assumptions made in the derivation of the  $\Delta\phi$  equation were reasonable, and that the integrator is providing precise and expected values.

## VII. Conclusions

### *Fleet Control*

Based on the above results, it is easy to see that the use of aerodynamic forces is an effective way to control satellite orbits. This opens many new doors for satellites such as CubeSats that lack thrusters. Perhaps the greatest aspect of this method is the opportunity for such satellites to fly in controlled formations. It is now possible for many small, low cost satellites to be built and launched and for those satellites to be distributed throughout an orbital plane to provide resolution through time (when one satellite is unable to view an area, the next satellite advances to a position where it can view that area). If an operator wants to reduce the separation between two satellites that are in the same orbit but a given distance apart, the chasing satellite could be oriented such that it experiences more aerodynamic drag than the leading satellite. This would cause the chasing satellite to drop to a lower altitude than the leading satellite. The lower altitude would cause it to travel with a faster velocity and have a shorter orbital period, allowing it to catch up to the leading satellite. The satellites would then be brought to

the same orbital radius by increasing drag on the higher altitude satellite (allowing it to drop). The time required for this maneuver is dependent on the exposed surface area to mass ratio ( $C_B$ ) and the ambient atmospheric density value. Higher density and  $C_B$  values result in a higher drag-induced acceleration. Since density decreases with altitude, this maneuver will take longer at higher altitudes. For example, at an altitude of 600km, it would take about 7.26 days to change the separation between two satellites by 100km, assuming that both satellites experience roughly the same atmospheric density throughout the maneuver (a reasonable assumption since altitude differences will generally be less than 1km). These times are long enough that at higher altitudes (such as 600km), it would be possible to utilize NORAD radar data augmented with Doppler shift observations to determine the positions of satellites in a fleet and decide from the ground when orbital maneuvering would be required. The satellites could then be supplied commands from a ground station specifying desired orientations and times between orientation changes. Since any maneuver will likely take days, it would be feasible to rely on the methods mentioned above to track the satellites throughout the maneuver and update their control schemes as necessary. This method would enable a fleet-based mission without requiring the satellites to communicate with each other or to have advanced position determination and control systems onboard. This would greatly simplify such missions, allowing them to be conducted by universities or other groups that are unable to implement more advanced fleet control techniques. It is important to note that the time it takes to change a certain distance between satellites is not a linear function. If it takes time  $t_1$  to change satellite separation by a distance  $x_1$ , then the time  $t_2$  it will take to change separation by a distance  $x_2$  can be calculated by

$$t_2 = \sqrt{\frac{x_2}{x_1}} t_1 \quad (22)$$

According to this equation, it would take  $\sqrt{10}$  times the amount of time to close a distance of 1000km than it would be to close a distance of 100km. This non-linearity is because a maneuver designed to change satellite separation by 1000km would allow the satellites to achieve a greater maximum separation before the satellites must begin to be brought back together. This greater separation value would allow a

greater change in separation over time. Thus it is more efficient, both in terms of time and total altitude loss, to perform one 1000km maneuver than to perform ten 100km maneuvers. For this reason, it would be a better idea to wait for the required separation change to be as large as possible before performing any maneuvers.

### ***Orbital Decay***

Controlling satellite orbits using differential aerodynamic forces may slightly increase the rate of orbital decay, but these changes will be small and relatively insignificant as they will only apply for the duration of the maneuver. During the maneuvers described in this paper, each satellite will spend half of the maneuver in minimum drag configuration (1U facing forward) and the other half of the time in maximum drag configuration (9U panel array facing forward). This will have roughly the same effect on overall orbital decay as having a 5U area facing forward throughout the maneuver (1U area represents a 10x10cm face). If a satellite such as the one shown in figure 7 is assumed to have each of its faces facing forward for an equal amount of time during normal mission operation, the average frontal surface area can be calculated to be  $A_{avg}=(9+3+3+3+1+1)/6=3.333U$ . Thus, the rate of orbital decay during a maneuver will be  $(5/3.333)=1.5$  times the rate of decay under average mission conditions. Since maneuvering will probably only be done during 10% or less of a mission's lifetime, these maneuvers will not have an appreciable effect in terms of increasing orbital decay and reducing mission lifetime. However, if specific orientations were not required for the mission at hand, both satellites could be oriented into a minimum, maximum, or intermediate drag configuration for all times during the mission when maneuvering was not required. A mission with satellites in minimum drag configuration would have a lifetime about 9 times as long as that of a mission with satellites constantly in maximum drag configuration. These concepts could be utilized to make a satellite de-orbit faster or slower depending on the specific mission requirements and desires.

### **VIII. Future Work**

As of now, this simulator assumes a constant density value throughout maneuvers. This is a reasonable assumption because altitude changes are small, and maneuvers take long enough for an average density value to be a good approximation. In reality, density

will change with altitude and can fluctuate by up to two order of magnitude for a given altitude based on solar and geomagnetic activity. It will be important to take this into account to improve simulator accuracy. Solar pressure, interactions with the Earth's magnetic field, and external gravitational forces will also affect satellites' orbits. Future simulations must take these into account as well. Additionally, a large drag plate on a satellite could be oriented at an angle to the velocity

vector to generate aerodynamic lift which would act perpendicular to the velocity vector and allow for possible out of plane maneuvers by the satellite. However, this lift force is relatively weak compared to drag forces, and small out of plane transfers would likely be less useful than changing satellite relative positions within the same orbital plane. Nevertheless, aerodynamic lift forces will be important to consider in future simulations.

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