Real-Time Global Solar Observatory

MECHANICAL AEROSPACE ENGINEERING

ASTRONAUTICS
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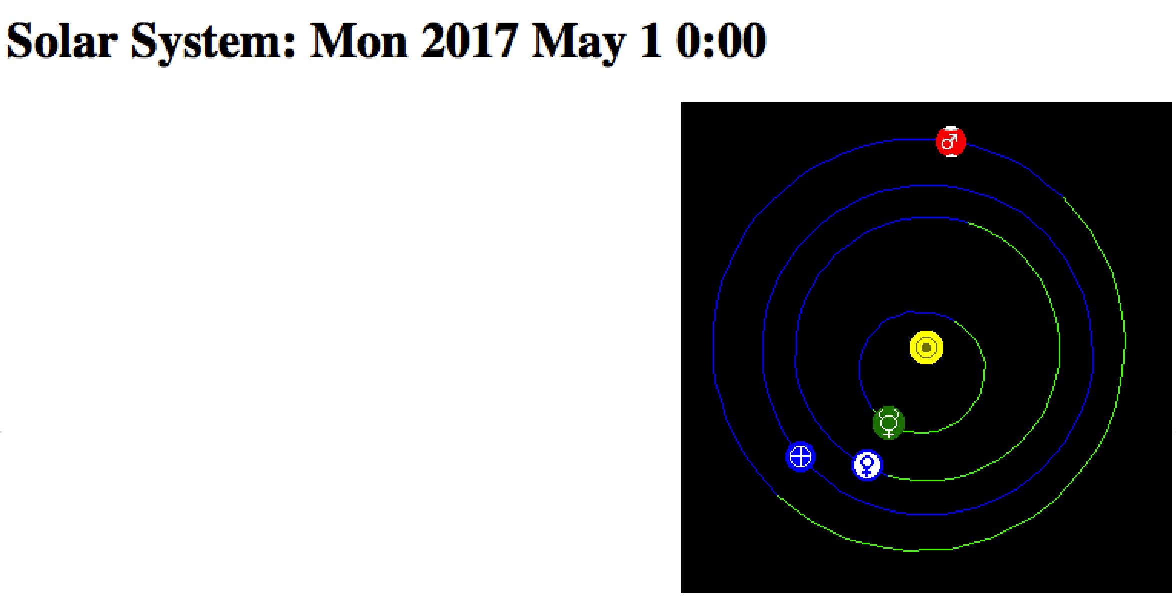
MISSION OVERVIEW

The main goal of this mission is to insert 6 satellites into Sun's orbit within 5 years and it should last for another 5 years. This is the basic requirement that was given by the mission statement. Although it may seem simple as the equations go, such as $F_{centripetal} = m * a = \frac{mv^2}{r}$, the position of the satellites will differ in the sense that it will have inclination in respect to the inertial frame of the Sun. After the satellites have been launched, it will be intact in one spacecraft with total payload. In my report, my method of inserting these satellites into the orbit uses several parameters and methods depending on where the satellites would want to orbit around. One of the mission requirements is that there should be satellites watching over North and South Pole. The inertial frame of the Sun in the X, Y, Z axis, and the satellites will be positioned in respect to other satellites. I will be naming the satellites in SC1, SC2, SC3, SC4, SC5 and SC6. These satellites will be grouped in two spacecraft that will be attached to the main spacecraft where it can be launched from the Earth. By using www.fourmilab.ch, I was able to find out the approximate position of Venus, Mercury and Earth in respect to the Sun. Since the orbital speed increases as distance from the Sun and decreases in planets orbital velocity, Hohmann transfer and bi-elliptical orbits around the planet Venus in order to conserve energy. The gravitational pull of Venus will launch the spacecraft 5 and 6 with the flyby to slingshot the space craft to North and South Pole.

The launch window was given in 5 years, which is approximately 3/01/17 as shown. Although this orbital of the inner solar system does not appear a perfect circle, it should be assumed that all the orbits that were made by the planets should be considered to be a circular orbit and it would simplify the radial and tangential velocity according to the (h), momentum and the radius of the orbit.

Figure 1: Date and position of the planets



INTRODUCTION

Imagine how this paper got to your own desk or whatever the reader might be reading. How were these prints made? The paper that you are currently reading has been processed by numerous technologies before it was printed. All the technology including our very own UC Irvine engineering gateway printers, servers, and the connection to the rest of the world through internet, wireless phone, and data transfers, has been the smallest or biggest part to printing out this paper. This technology involves satellites and internet which can easily be disrupted by the Sun. The Sun is an active nuclear reactor that goes under nuclear fusion in order to create energy. Sun is also like a planet that has North and South Pole that creates magnetic field around the planet itself. However, the magnetic field of the sun changes its poles every 22 years. At the time when there are solar activity, some spots in the Sun has so much difference in magnetic field that it is like a spring that is about to go messy when given energy. At this moment of explosion (not really explosion, but an occurrence on the surface) it will shoot out parts of the Sun, which are called the "Solar Flares." Recently the Sun was active as it approach the 22nd year, and the Solar Flares has been shot out into space in the direction of the Earth's orbital path. Solar flares can not only disrupt satellites and communications, but it also creates Aurora which is solar particles that are caught by Earth's magnetic sphere. There are two types of Auroras depending on the North or South pole of the Earth and these are called Aurora Borealis in the Antarctic regions and Aurora Australis for the Aurora made on the southern hemisphere of the Earth. Solar and Heliospheric Observatory were the product of NASA and ESA for the first international cooperation for the study of the Sun. Also, for solar satellites, one of the main purposes is to determine the location, direction and magnitude of the solar flares that might be occurring now, at this present time, or -500 seconds since the light travels 500 seconds to get to the Earth. For the astronauts in the I.S.S or LEO orbiters such as the space shuttle are warned as the neutrinos and other particles can rip out parts of the human DNA. To avoid this problem or to solve this, NASA and other space agencies sent space probes which constantly orbit the Sun and image the Sun and its solar activity through infrared and visible light waves. Also, having to transmit the radios and datas into other satellites is the key to this mission because one orbiter cannot be imaging and be in direct contact with the Earth, or have the line of sight. Thus, having 6 greatly improves the communication angles when the satellites are in conjunction with the Earth.

Gravitational parameters differ for every orbit because the distance that the satellites are going to be traveling will determine how much the gravitational pull will have from the Sun, Earth, Venus and Mercury. Mars will be too far away to have much effect on the satellites orbit.

Modeling Assumption and Target Configuration Analysis

In modeling part of the trajectories, the orbits of planets were assumed to be circular and thus eccentricity is 0, as all the calculations were done in Sun Inertial Frame. The key assumptions here are the eccentricity value. All the distances are equal in every direction, and that this model is with the real time. However, this mission report relies on the time function that relates to the spacecraft from the launch t = 0. Gravitational parameters are used in almost every case, but only the hyperbolic planetary departure and parking orbit will have the planetary gravitational parameter other than the Sun's. Only the spacecraft’s 5-6 are to do flyby and it will be doing the incline and plane change as well. Only the spacecraft’s 1-4 will be rotating the Sun in positive and negative X and Y axis at the same time at the same rate, assuming circular orbit of the Sun. By using Hohmann and Bi-elliptical, I was able to play around with the time and delay the Bi-elliptical transfer time by factor of T/4 of the rotational period on 90,000,000 km away from the Sun. As for the launch and the parking orbit around the Earth, the hyperbolic departure using angle of 65.2 degrees and the time that the spacecraft’s will be on LEO will be a week, 168 hours.

In the modeling of this mission, the orbit diagrams of the orbit, orbit transfers, plane/inclination change, and the parking orbit has been shown with the following data.

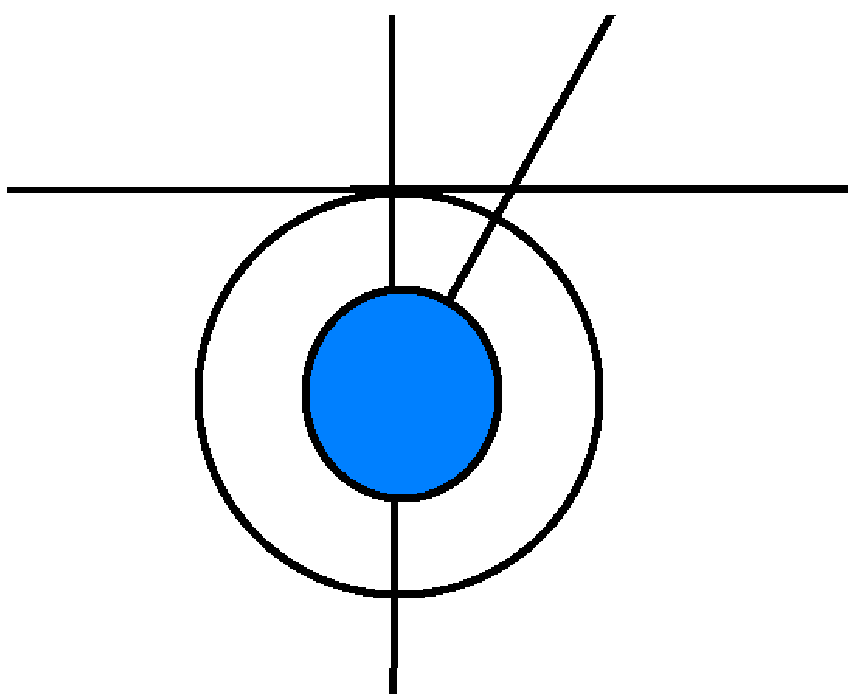


Figure 2: Parking Orbit Hyperbolic departure with 65.2 degrees

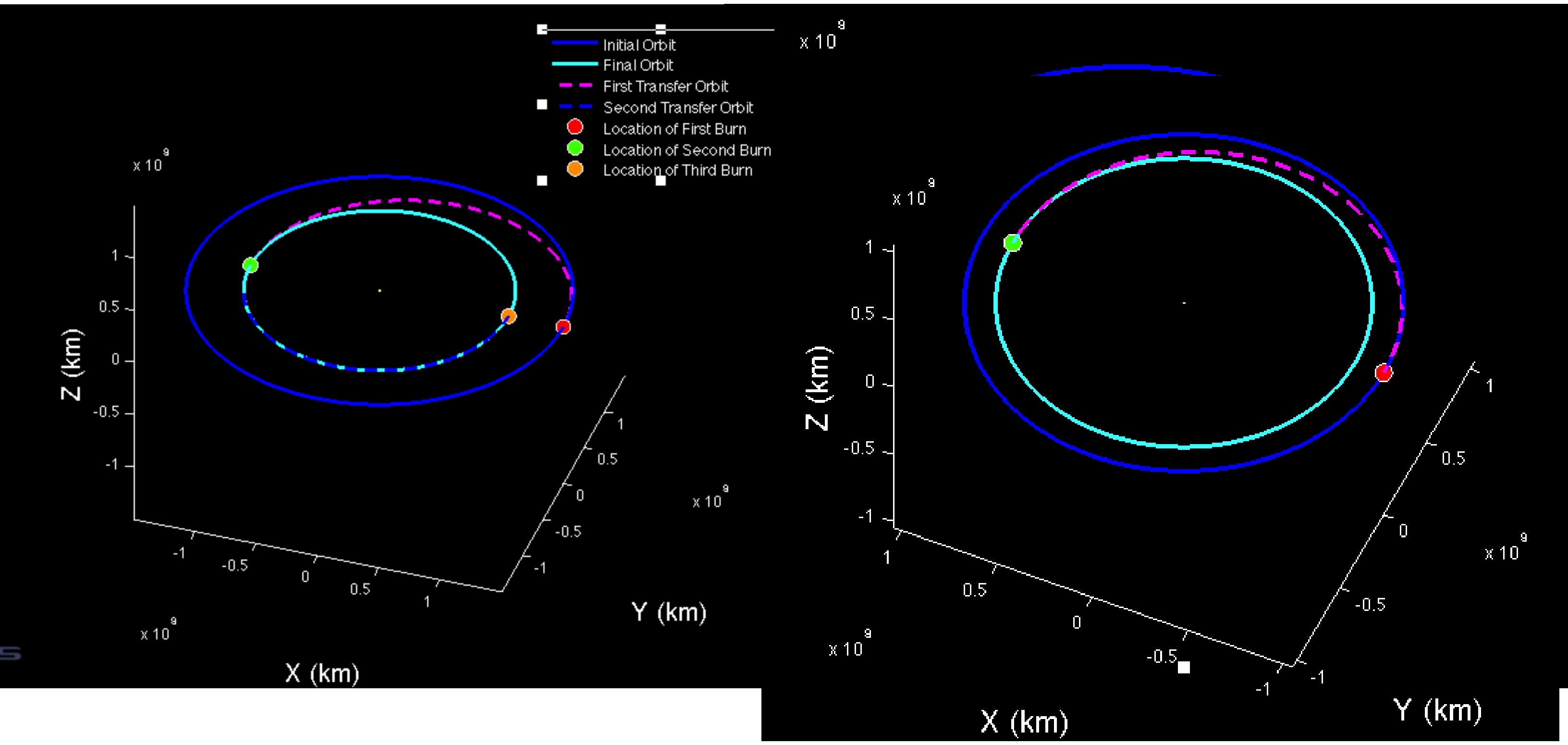
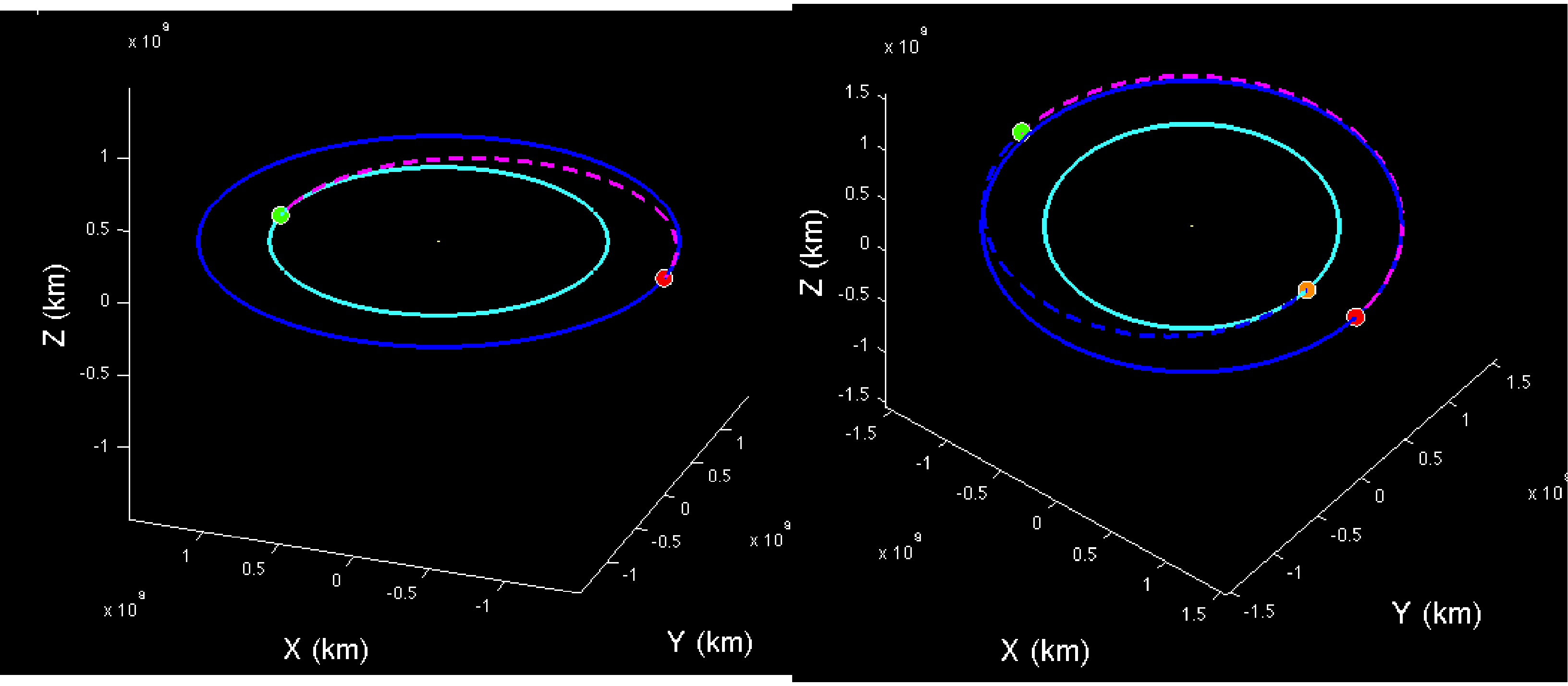


Figure 3: Spacecraft 1. Orbit 1(left) Orbit 2 (right

Figure 4: Spacecraft 2.

Orbit 1(left) Orbit 2 of spacecraft 2-4 is same as Spacecraft 1 Orbit 2



These are the orbit models for the spacecraft 1-4 on the X and Y plane with no inclination change.

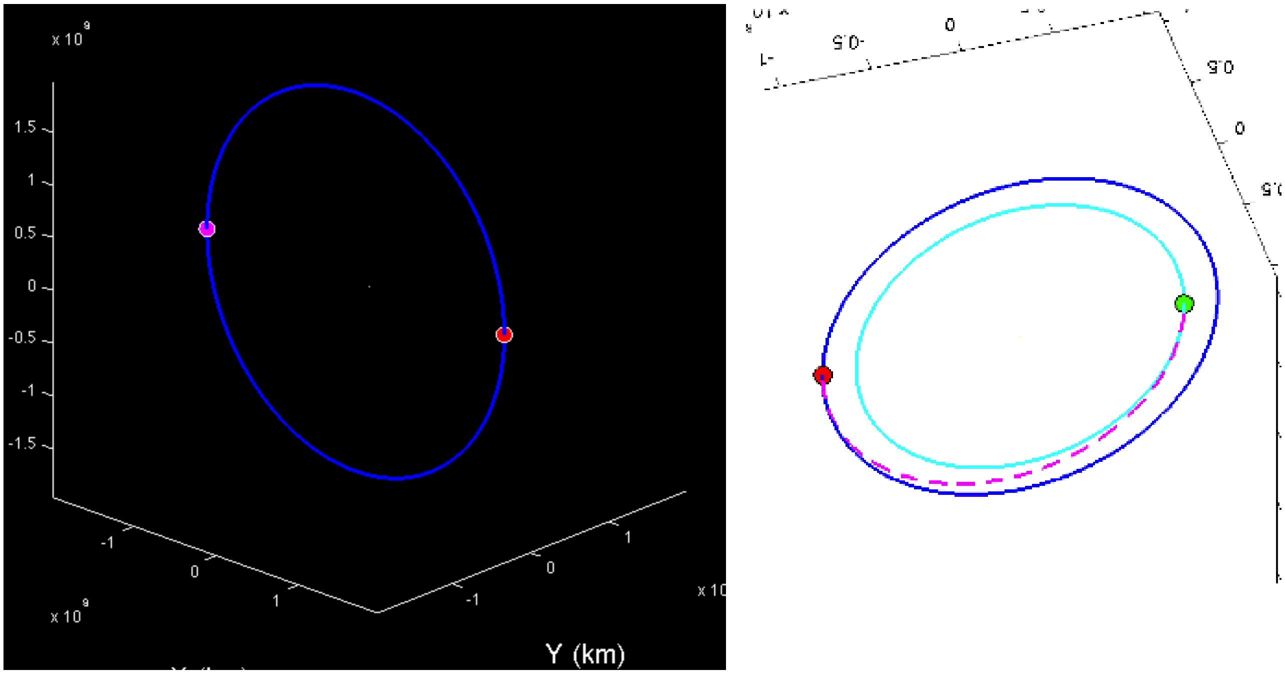


Figure 5: Spacecraft 4

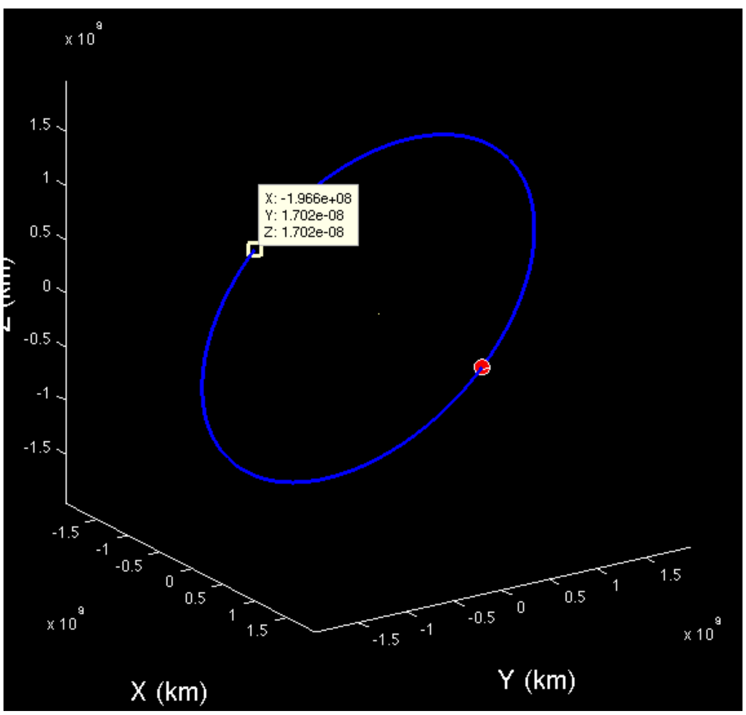


Figure 6: Spacecraft 5 and 6

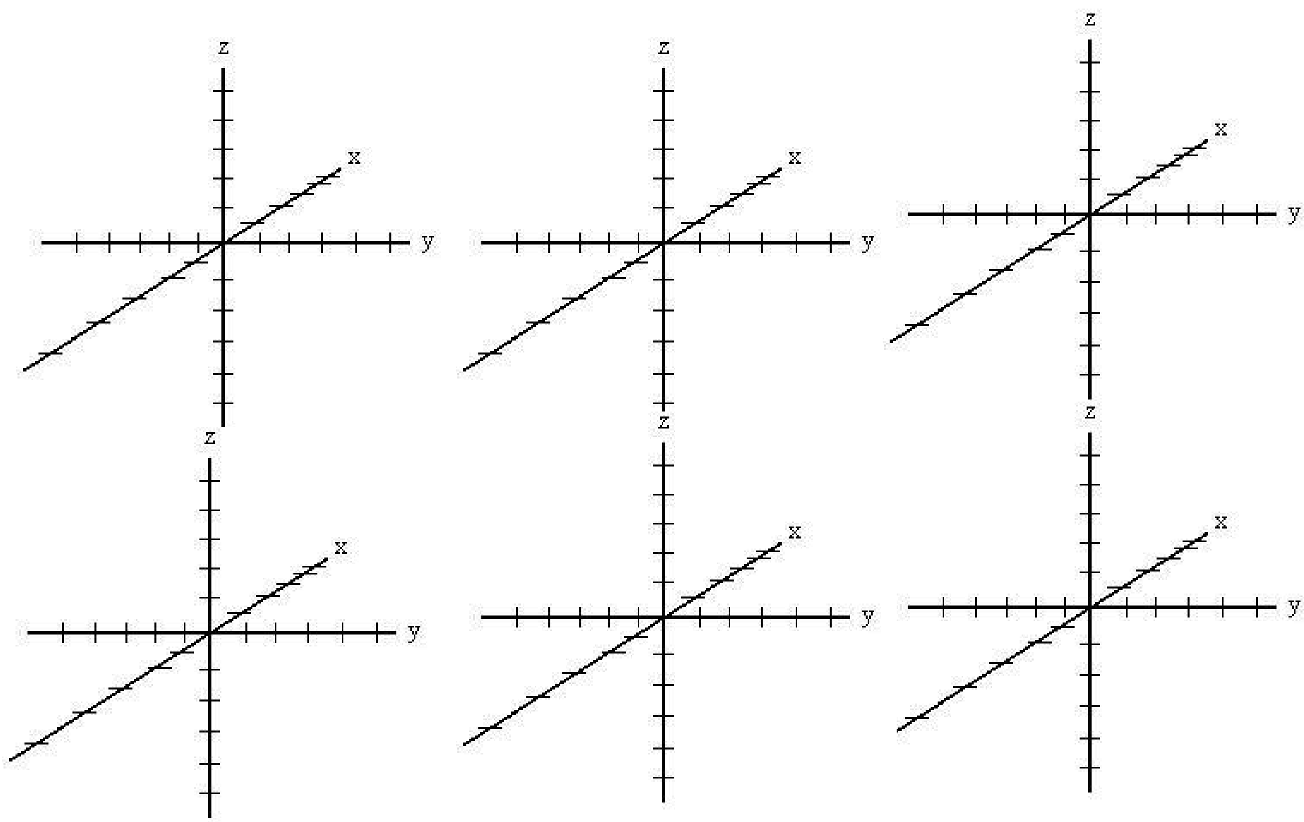
This is the inclined orbit of 90 degrees with true anomaly 180 degrees and eccentricity = .09221 radius of apoapsis = 180,000,000 km. At time = 0, $R_x = -2.52\ x\ 10^8\ km$ v_y= -17.76 km/s T = 11569.9 hrs

On the left is the inclination change from angle 0-45 with the bi-elliptical orbit transfer and on the right is the Hohmann transfer into the position from the Sun on the final orbit AFTER flyby of the Earth. Third picture is when inclination is 45 degrees and the space craft is at the far point when the inclination change is occurring.

Δv calculation for inclination change.

Modeling steps for spacecraft 5-6 will be into different steps

1. Do two elliptical orbits and do inclination/plane ("Orbit Cranking") change at the apoapsis with 45 degrees each.
2. Do a flyby around the earth on the 2nd orbit and change direction to Solar orbit
3. After the fly by, the insertion into solar orbit will be in Hohmann for the spacecraft 5 and Bi-Elliptical for spacecraft 6.



Bi-Elliptical orbits have more efficient Δv for plane changes rather than Hohmann transfer

Limitations:

The limitations in this mission are that the cube will be only formed when either spacecraft 5 or 6 goes over the north and the south pole of the Sun. At some of these times, the North and South Pole would not be covered for some period. Another limitation is that by launching 6 different satellites at the same time, no mass can occupy same space at same time, which means by +-1km or +- t = 1 hour should be used at least to minimize the collision probability. The relative range of the spacecraft 1-4 will be constant, which is 127300000 km to the axis to the left or the right.

The Δv required by the planetary departure is 3.50 km/s with the orbital velocity 7.78 km/s. The departure speed in relative to the Earth is 2.48 km/s which can be added up to 32.55 km/s relative to the Sun. It departs the Earth at 65.2 degrees that is away from the Sun.

Description and method of computation:

There will be two different spacecraft that will have different amount of payload, but if there are velocity changes to be made, it would have slightly different payload depending on the velocity boost that the spacecraft will be given. The vehicle will be intact during the parking orbit and will start to go when the amount of fuel to be burned according to the mass total is calculated using Hohmann transfer. The spacecraft will have change in inclination by using the law of conservation of momentum. At true anomaly is zero, these two spacecraft that has been divided for the 2nd time will have equal amount of inclination in both of the directions. The orbits that the Satellites S1, S2, S3, S4 are the ones that will be going around in the X and Y direction, that are equal to both sides in respect to X axis as shown above. That inclination and velocity will be included in the plane change equations for S5 and S6. As for the North Pole and the South Pole, the spacecraft will be traveling towards the Venus with the main craft and do a fly by while having its plane change from the Earth by having elliptical orbit near Earth, but at the Apopasis, the velocity of the spacecraft relative to the Sun is at the slowest. At this time, the plane change Δv must occur in order to have inclination. After spacecraft 5 and 6, S5 and S6 respectively, achieve the flyby around the earth, it will be the distance of the earth is at the furthest from the Sun, the conservation of momentum would be used to calculate v_∞ and would be transferred through bi-elliptical orbit maneuver within Venus's gravitational field. S1-S4 will be going on similar trajectories, but S5 and S6 will be going on different trajectory with inclination and plane change occurring at the apoapsis in order to have small Δv. At the end, the total time that is needed for this mission will take more than 2-3 years and will be there all together at the end. It will be at separate times when the spacecraft will be positioned in to its own designated spot.

The fact that the cubic formation only happens when the either SC 5 or SC 6 will be crossing over either the north or the south pole, which has period of 4090.6 hours, and dividing this by 2 will give out 2045.3 hours and the spacecraft will form cubic formations at every 2045.3 hours because the SC 1-4 will have the square formation at all times after the planetary rendezvous. SC 5 and SC 6 will be doing planetary flyby around the Earth as the position of the spacecraft approaches earth on the second return trip from the two elliptical plane change maneuver at the apoapsis. All of these spacecraft SC 1-6 will be on a parking orbit around the Earth about 200 km above the surface of the Earth. The mass chosen for this mission has been decided to be 3600 kg because the Solar and Heliospheric Observatory weighs about 610 kg. I decided to have 600 kg per spacecraft. In visa versa, I will be able to use the Δv calculated and the impulsive maneuver equation to find the mass and the fuel mass

$$\Delta m / m_o = 1 - e^{-\Delta v / (I_{sp} * g_o)}$$

So about each spacecraft, 6.48 kg of fuel is burned

Gravitational Constants:

μ Earth = 398600 km^3 / s^2

μ Sun = $1.327 \, x \, 10^{11} km^3 / s^2$

μ Venus = 324900 km^3 / s^2

	Maneuver Method	Necessary Computation	Positions
S1, S2	Δv 1 Hohmann and Bi-Elliptical for S2 and S1	Trajectories and Hohmann and Bi-elliptical in respect to the sun.	X-Y plane. In circular orbit.
S3, S4	Δv 1 Hohmann orbit and 1 Bi elliptical for S3 Δv 1 Hohmann and Bi-Elliptical for S4	Trajectories and Hohmann and Bi-elliptical in respect to the sun.	X-Y plane. In circular orbit.
S5, S6	It will have inclination change with Δv and slingshot to Sun using Venus Flyby	Venus Flyby. Trajectory with (bi-elliptical S5) Hohmann and plane/inclination change	It will be on Z axis + and – side of the Z.

All the spacecraft will have different traveling time due to the fact that it will have to achieve cubic formations. It will be launched and positioned into the periapsis of the Sun at every Hohmann and Bi-Elliptical orbit transfers. The orbital period of the spacecraft in $90 \, x \, 10^6$ km is 4090 hours. By dividing this hour into 4, I was able to achieve the launch window for the spacecraft. The steps into configuring the orbital transfers, I used 2 Hohmann orbital transfers on the spacecraft 1 to compute the relative position of the other spacecraft's and launch spacecraft 2-4 at every 1022 hours to get the 90 degrees in true anomaly. The eccentricity of the orbitals are mostly 0. The total Δv maneuver in space for the spacecraft 1-4 was calculated to be 3410.67 m/s.

<u>Steps into Insertion of Solar Orbit of Spacecraft 1-4.</u>

1. SC 1-4 will be launched to LEO around 2017 near Feb-March.
2. It will have 2 different launches but all different Δv maneuver which will carry S1-S4 and S5,S6 due to huge inclination/plane change and it will require too much Δv that it would be easier to launch it at different rockets.
3. The mass of the Spacecraft will be about 600 kg each. Since the SOHO Solar and Heliospheric Observatory weighs about 1850 kg.
4. The rocket that I will be using is Atlas V 401. It has maximum load that can send into SSO about 6,670 kg.
5. After the rockets have been launched, it will stay in orbit for about a week, making about 16 revolutions per day.

6. At the point of departure, it will be directly in same X axis as the sun for easier computation and e=0 was used.
7. Spacecraft 2-4 will use both Hohmann transfer to get into 90x10^6 km around the Sun, taking about 7297 hours total. First Bi-Elliptical transfer time will be 2306.21+1022.4*(n-1). (N = Spacecraft number.)
8. By using the similarity of the Hohmann transfer, I calculated the Bi-elliptical transfer of the other spacecraft by (time SC 1 Total – SC 2 Hohmann = time for Bi-elliptical transfer for other spacecraft and it will be + or – 1022 hours to position the spacecraft.
9. SC 1-4 will be using Bi-elliptical orbit from 149600200 into 105,000,000 km and do Hohmann transfer to 90,000,000 km. The time that was subtracted from SC1 second Hohmann from the total time + 1022 multiply by factor of n (n= 1,2,3,4...) hours will give the position of the spacecraft at the periapsis, while the SC 1 will be at position on Y axis R distance from the Sun. n= 2 and n= 3 for SC 3 and SC 4 respectively. In this way, the spacecraft will be positioned in a square formation. The time will start at t= 0 and it will translate into Julian calendar to be exact with the dates on earth.
10. Since the spacecraft are near the sun, the spacecraft can have solar sails to make the necessary Δ V calculation.
11. Spacecraft 5-6 will be launched separately in different rocket to have different plane of travel in X-Z directions rotating in Y axis.
12. SC 5-6 will have plane change in the near Earth's orbit by using elliptical orbit. The speed of the spacecraft at the Apopasis is the slowest and will give small Δv maneuver to conserve fuel.
13. Also, as SC 1-4 was done with time, SC 5-6 will be given their initial time equal to SC 1-4 into the inner orbit. Because SC 1-4 was given 90x10^6 km in $\pm$ (X,Y) coordinates, SC 5-6 will have 89,999,990 km to have no collision in space.

Parking Orbit

The parking orbit has been given as 200 km above the Earth's surface. It is easy to compute the velocity that the spacecraft is traveling at that point. Also, when the spacecraft is on 200km above the surface, it will have velocity of 7.78 km/s. From that value and the speed of the planet in relative to the sun, I was able to find out the relative hyperbolic departure velocity of the spacecraft relative to the sun. That velocity can be maneuvered from the hyperbolic departure velocity and its trajectory. The spacecraft will have its own fuel and structural mass depending on the Δv required by the Hohmann, Bi-elliptical, or plane/incline change. Spacecraft 1-4 will have the hyperbolic departure towards the inner solar system by using Hohmann/Bi-elliptical transfers. However, for spacecraft 5-6, which will be using the incline/plane change around the Earth's orbit with eccentricity = .09221, will require more Δv.

Planetary Departure

After the spacecraft has been in parking orbit for about a week, it will have the initial Δv which will cause the spacecraft to travel faster than the orbital velocity of the planet itself. In that way, the spacecraft can be positioned into the right place at right time by using Hohmann and Bi-elliptical. For bi-elliptical, maximum transfer time is one of the given functions of the transfer input. Thus, by using 105,000,000 km, the spacecraft will not be too near but not too far away from Venus but to the radius of 90,000,000 km around the Sun. The time that it takes from 105,000,000 km to 90,000,000 km with Hohmann transfer is about 2306.21 hours. This transfer time for my project was basically the constant value for S1-4 because all the satellites will be launched at the same time; however, the time that they arrive at the periapsis will be in sequence of quarter of a whole period (2pi). The orbital period of 90,000,000 km is about 4090.6 hours, which gives 1025.15 hours for the arrival of the spacecraft.
By the way, that is the final orbit configuration with the time variable. However, the middle time that is missing are the transfer time of another form of orbit transfer, which can include either bi-elliptical or Hohmann transfer. Bi-elliptical transfers are more fuel efficient on the Δv with the incline/plane change; however, Hohmann transfer can also be efficient when the orbit is coplanar.
Here is the data of the spacecraft SC 1-4 in the first orbit transfer.

Orbit Insertion #1	S1 Hohmann	S2 Bi-Elliptic	S3 Bi-Elliptic	S4 Bi-Elliptic
Radius input km	149600200	149600200	149600200	149600200
Eccentricity	0	0	0	0
Radius output km	105000000	105000000	105000000	105000000
Period (t) hours	4991.04 hours	6013.44 hours	7035.54 Hours	8057.74 Hours
Δv)1 km/s	2.73	-2.74	-1.03	-0.26
Δv)2 km/s	2.99	-2.99	2.85	2.23
Δv)3 km/s	0	-0.01	1.88	3.27
TotalΔv) km/s	5.73	5.74	5.75	6.24
SC 1-4 Δv	23.46 km/s			

At Orbit Insertion #2, the period defined per spacecraft is on the R of the positive axis addition from the Orbit Insertion #1 period (t).

Orbit Insertion #2	S1 Hohmann	S2 Hohmann	S3 Hohmann	S4 Hohmann
Radius input	105000000	105000000	105000000	105000000
Eccentricity	0	0	0	0
Radius output	90000000	90000000	90000000	90000000
Period (t)	2306.21	2306.21	2306.21	2306.21
Δv)1	1.39	1.39	1.39	1.39
Δv)2	1.45	1.45	1.45	1.45
Δv)3	0	0	0	0
Total Δv)	2.84	2.84	2.84	2.84
SC 1-4 Δv Orbit 1 + Orbit 2	8.57	8.58	8.59	9.08

Δv total 58.28 km/s for spacecraft 1-4

The equation $\Delta m/m_o = 1 - e^{-\Delta v/(I_{sp} * g_o)}$

Amount of Fuel Used and Mass of the Spacecraft	Mass in kg	Amount of Fuel Needed for Δv in kg
Spacecraft 1	600	1.9281 kg
Spacecraft 2	600	1.499 kg
Spacecraft 3	600	1.500 kg
Spacecraft 4	600	1.586 kg

Total amount of fuel in mass = 6.513 kg for spacecraft 1-4

Thus, the total amount of Δv required by spacecraft 1-4 is equal to 37.29 km/s and the total amount of fuel and mass is shown at the table above.

Spacecraft and Orbit Number	Spacecraft 5 Orbit 1	Spacecraft 5 Orbit 2 Hohmann	Spacecraft 6 Orbit 1 Hohmann
Apoapsis	180,000,000	108,200,500	180,000,000
Eccentricity	0.0922	0	0
Radius output	108,200,500	90000000	90,000,000
Period (t)	8597.5	2363.21	3757.44
Δv)1	-1.4847	1.6467	3.7
Δv)2	**3.08825 (flyby)**	1.724	5.94
Δv)3	3.1794	0	0
Total Δv)	7.7524	3.371	9.641

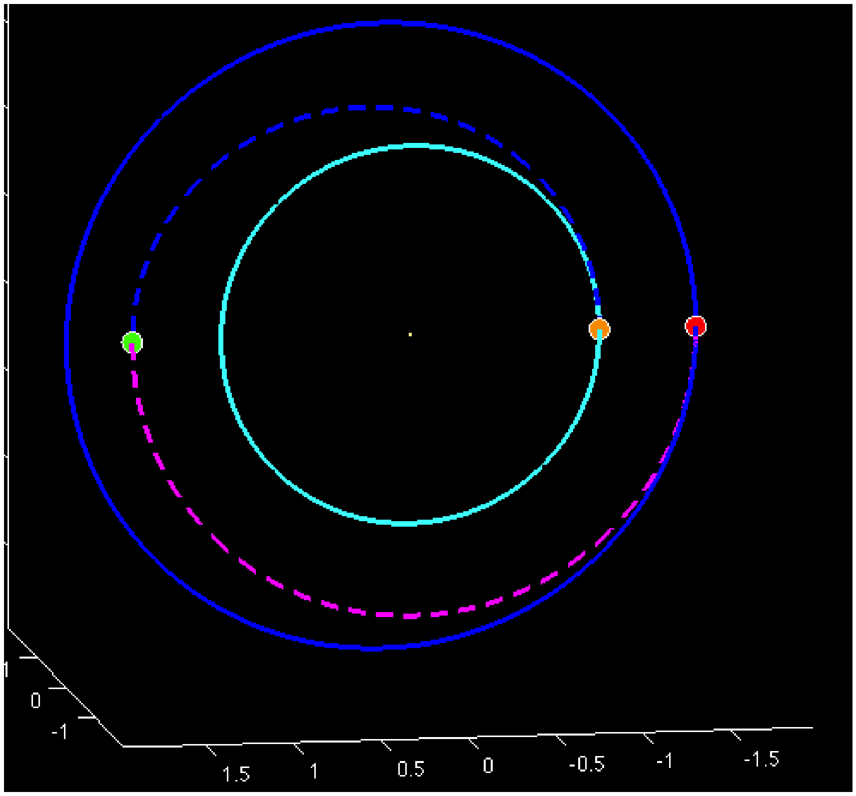


Figure 8: Inclination change of spacecraft 5 and 6

The inclination and plane change data is on the txt file

Δv of incline change is determined by the equation

$\Delta \mathbf{v} = 2 \times v_{circular} \sin\frac{\theta}{2}$

Thus, the result from the plane/incline changes with Δv is10.46 km/s each.

Orbit Transfer of 6 Satellites into 90,000,000 km from the Sun.

Spacecraft 1-6 just from the transfer Hohmann + Bi-Elliptical

58.28 km/s + 20.76 km/s = **79.04 km/s**

Inclination/plane change maneuver δ = 45 degrees each.

20.92 km/s

Planetary Departure Δv

3.50 km/s

The total Δv = 79.04 + 20.92 + 3.50 =

120.96 km/s

Rendezvous

Δ = 1.27 x 10^6 km

δ = 70.74

Θ∞ = 125.37

With no inclination or plane change, the orbit insertions of SC 1-4 only includes the Δv burn of the engine to either decelerate or accelerate the spacecraft to insert into certain orbit. By the end of the SC-4 arriving on the periapsis with the radius of 90x10^6 km, all four spacecraft will have a square formation going around in a circle with same radius and angular velocity.

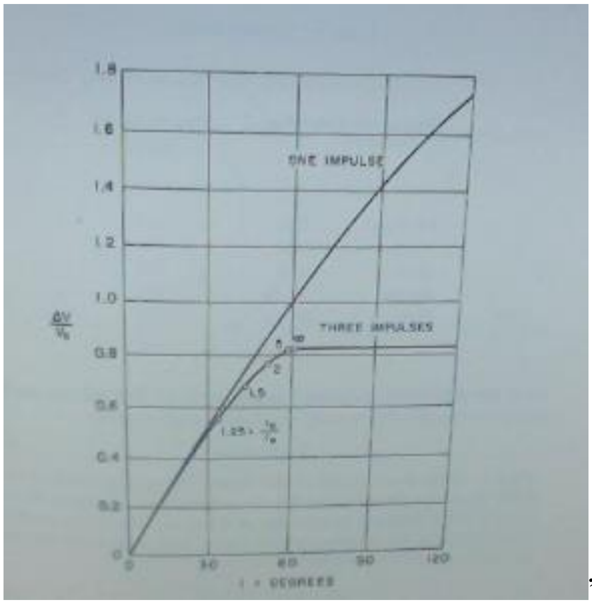
<u>Spacecraft 5-6</u>

These spacecraft that are launched will have same parking orbit and orbital with SC 1-4. However, SC 5-6 will have to go through plane changes which requires incline thrust and change the plane of orbit to 90 degrees or -90 due to the fact that it will be launched to the north and the south pole of the Sun. From the data from previous experiments and missions, ellipses that have high eccentricity with true anomaly being 180 degrees will experience the slowest velocity possible in that orbit due to the fact that it is at the furthest point of the orbit. With the equation of Δv $= 2 \times v_{circular} \sin\frac{\theta}{2}$, I was able to find out the Δ-v required for the plane change maneuver in the parking orbit. At a circular orbit, the velocity will be constant, however with the eccentricity being .0922, and R apoapsis being the lowest velocity and the change in inclination at this point will have the least Δ-v. At this point, I chose to have 2 elliptical orbits that will have inclination change at the apoapsis each with theta = 45 degrees, at which on the 2nd orbital, by the time it will reach the periapsis the spacecraft will have the vertical 90 degrees, with the formula Δv $= 2 \times v_{circular} \sin\frac{\theta}{2}$. At this time, the spacecraft will do a Δv maneuver by flyby from the earth's gravitational pull. Because of the fact that the Earth does not have the radial velocity towards the Sun at the time, the flyby will just have effect on the angle change in the velocity. After the flyby direction change, it will have Bi-elliptical pattern to the 89,999,990 km to the inner solar system radius. It will have two maneuvers which will allow the spacecraft to have orbital transfers. Spacecraft 5 will have two bi-elliptical; however the spacecraft 6 will have Bi-elliptical as first orbit and Hohmann transfer as the second orbit. Due to having difference in the distance is 10 km, it would be enough for the spacecraft to not hit each other. The fact that these two elliptical orbits will be having to travel twice around the earth to make the plane changes, will give the Earth at least 1-2 years because the period of the spacecraft will have to match the period of the Earth in order to do a flyby on the second orbit around the earth. Approximate launch window is 2018-01-16 with Venus being near Earth, but it will be at conjunction in 179.11 days to do flyby.

Transfer Options Analysis

As stated before, the spacecraft 1-4 will have Hohmann transfer from 105,000,000 km to 90,000,000 km orbit, which is the 2nd orbital transfer to the Sun. Also, spacecraft 2-4 will have bi-elliptical transfer orbit which will be the initial orbit of the launched spacecraft. Velocity change from circular to hyperbolic is ${v_p}^2 \quad = 2 \times {v_{circular}}^2 + {v_\infty}^2$ and the velocity difference is tangential to the Earth's orbit. For $0 < {}^{v_\infty}/_{v_{circular}} < 0.5$, the $\Delta v > v_\infty$ whereas ${}^{v_\infty}/_{v_{circular}} > 0.5$ then $\Delta v < v_\infty$. Δv in this case is $v_p - v_{circular}$.

The key relationship about the orbital transfer is mostly the angular momentum, eccentricity, true anomaly, apoapsis, periapses and transfer velocity in the previous orbit to transfer. One of the key relationships about the gravitational parameter is that most of the computation was done under the gravitational parameter of the Sun; however the departure velocity and the parking orbit used Earth's gravitational parameter. In my computation, the true anomaly factor has been 0 degrees due to my initial point on the X-axis, which means that for the orbital transfer, using the time that passed during the transfer is related to the position that the satellites will be. For the flyby, the position of the satellite will be captured by the Venus's gravitational field and the satellites will have vertical Z velocity only at the point of the intersection of the ellipse and the X-axis. The flyby will conserve the energy of the motion, not to add, but to be same as the initial velocity: not the vector form in the velocity but the magnitude will be equal. By using planetocentric hyperbola and using center as the Sun, and $e = 1 + {}^{r_p * {v_\infty}^2}/_{\mu_{Sun}}$ and to find the turn angle and the true anomaly of the asymptote are given in this formula. The capture radius is determined by the rendezvous. However, since the capture orbit has eccentricity 0, the Δv or the aiming radius to the Sun will be 127,000,000 km. The other options that were available were to do a flyby maneuver to Venus in the same plane as Earth and Venus, which is X and Y axis. However, due to time constraints, flyby was not available for spacecraft 1-4. Thus, having to do a flyby was another idea from not using Δv for the spacecraft 5-6 to conserve fuel due to incline and plane change that it has to do in the beginning of the flight.



"Transfers between Inclined Circular Orbits of Equal Radius" courtesy of American Institute of Aeronautics and Astronautics, 1966"

Launch Vehicle Selection

	Booster (light)	Booster (heavy)	1st Stage	2nd Stage XX1	2nd Stage XXII
Engines	Solid	RD-180	1RD-180	1RL10A	2RL10A
Thrust	1270 kN	4152 kN	4152 kN	99.2 kN	147 kN
Isp	275s	311s	311s	451s	449s
Burn time	94s	253s	253s	842s	421s
Fuel	Solid	RP-1/LOX	RP-1/LOX	LH2/LOX	LH2/LOX

The rocket that can take the satellites to the orbit will be Atlas V, which is the rocket that was used to take "Spirit," the Mars rover that explored the surface of Mars. Atlas has several specifications.

The Atlas V has the payload up to 7000 kg and it has the structural mass of 334,045 kg and has average thrust of 4000 kN. The Atlas V 401 rocket will be launched at the Cape Canaveral where the air density above 13,000 ft is 39% less dense than the normal air and Max Q will be a less factor. Kennedy Space

Center is located 28 28' north 80 33' East. The reason for the chosen Atlas V 401 rocket is because the fact that NASA has sent Mars rover to land on Mars. It can shoot up to 7000 kg which is more than what the chorographic satellites (spacecraft 1-6) will be. The sum of the payload mass is 3600 kg and the Atlas V rocket has 334,045 kg of structural mass and 305,000 kg is the mass of the fuel. It is the smallest Atlas V family. The performance of this vehicle is beneficial for payloads that are considered to be massive in mass. Also, during Mars orbit insertion, the Atlas separated into 2-stage engine which created more power with less mass with directional capabilities to get to Mars.

Summary

In this mission, it is possible to go to the Sun and observe the Sun in cubic formation at all times. There are other scenarios that are created to maximize the visual contact of the Sun. When the satellite is behind the Sun, the satellite will have to travel 6,270,000 km because from the syzygy, or conjunction, the satellite will not have line of sight. Thus, pointing + or – 45 degrees pointing towards the sun, the satellite will be able to transmit the data. By launching a space rocket from the Earth, the rocket will go on the LEO at which the spacecraft will separate. As the spacecraft separate, there will be 4 and 2 satellites grouped together in a coplanar trajectory with one another. At a time t, the satellites will be at certain position; all the satellites in 1-4 will be 90 degrees away from the satellite that is in front and the back on the orbital direction. Also, the satellites that pass through the North and South Pole of the Sun will have to have the orbital period divided by 2, in order to pass it at the same time. All the Δv has been calculated and only huge factor from the Δv is the plane change. While sending 6 different spacecraft into the orbit of the Sun, the mass of the spacecraft can differ, which means that spacecraft 5 and 6 might have more fuel than 1-4 due to heavy maneuvers that they have to do.

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