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To the Far Side of the Sun Using Venus Gravity Assist

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The recent NASA initiative to investigate our sun in depth, called "Living with a Star," includes consideration of several possible deep-space missions, one of which is placing a satellite on the far side of the sun. This satellite, together with Earth, can then observe the full evolution of solar phenomena as the sun rotates. Unfortunately, in using a direct approach celestial mechanics does not permit satellite placement to be done efficiently and in a timely manner. Here, the direct approach for satellite placement is shown to be costly compared with the use of Venus gravity assist. This paper investigates specifically the insertion of a far-side sentinel satellite in the year 2009 into the third quadrant (180 to 270 deg ahead of Earth) using single and double Venus gravity assists. Options for other possible missions, including a distributed set of solar observers, are briefly analyzed.

Nomenclature

a_e	=	semimajor axis, km
C_3	=	hyperbolic excess velocity squared, km^2/s^2
P_e	=	final spacecraft heliocentric period, s
r_v	=	Venus distance from the sun, km
$\boldsymbol{U}_v$	=	Venus heliocentric velocity vector at flyby time, km/s
$\boldsymbol{U}^-$, $\boldsymbol{U}^+$	=	spacecraft heliocentric inbound, outbound velocities, km/s
$\boldsymbol{V}^-$, $\boldsymbol{V}^+$	=	spacecraft Venus-relative inbound, outbound velocities, km/s
V_∞	=	hyperbolic excess velocity, km/s
ΔV	=	injection velocity from 200-km Earth parking orbit, km/s
μ_s	=	sun gravitational constant, km^3/s^2

Introduction

A WORLDWIDE chain of ground stations called the global oscillation network group has gathered evidence that release of stress in solar magnetic fields might be driving the 11-year cycle of solar eruptions. This finding has led NASA to propose a more intensive study of the sun using deep-space observation satellites to view solar activity from a vantage point other than just the Earth.

Several projects related to NASA's living-with-a-star theme (LWS) are now being considered, one of which is the far-side sentinel (FSS; NASA Goddard Space Flight Center Web site, lws.gsfc.nasa.gov). One project would be developing and launching a solar far-side observer whose purpose would be to probe the sun's three-dimensional structure, both magnetic fields and mass flow, from deep within the surface to the outflowing corona. It would then be possible, combining these observations with those from Earth, to follow the evolution of active regions, taking full disk magnetic and velocity field observations. Mission requirements include a two-year, on-station observation time, plus a possible three-year extension.

Direct Transfer Modes

Consider the types of solar orbits that would be suitable for observing the sun, such as for the FSS. The simplest orbit would be a 1-astronomical-unit (AU) circular orbit, like the Earth, but positioned a fixed number of degrees ahead or behind the Earth. The satellite's orbital period would be that of the Earth, or 365.25 days. To achieve this orbit, the transfer from Earth of the satellite would be similar to a rendezvous with a hypothetical asteroid located, say, on the back side of the sun. To reach this position, a spacecraft would have to be launched from Earth such that it would either lead the Earth by sending it inward toward Venus or lag the Earth by sending it outward toward Mars. Then, when the spacecraft reached the desired angular position relative to Earth and returned to a distance of 1 AU, it would perform a maneuver to circularize the orbit and remain stationary relative to the Earth–sun line. These direct transfer modes are depicted in Fig. 1a (in an inertial heliocentric system) for flight times of 2.5 years. It is more informative, however, to present the trajectory transfers in a rotating system, with the Earth–sun line fixed, say, along the y axis. Figure 1b presents the same trajectories as shown in Fig. 1a but in this rotating system.

Tables 1 and 2 present direct inward and outward launches for a range of flight times, assuming circular coplanar orbits for the planets. They show flight times of 1.5, 2.5, and 3.5 years, which imply that the Earth at these times will be located directly opposite from its initial position at launch. Thus if the launch is in January, Earth's position will be on the other side of the sun in July. These flight times, on the other hand, represent the periods of the satellites, if they are to enter 1 AU circular orbits, because they must make full revolutions about the sun to return to their initial starting points of 1 AU. The launch C_3 and the rendezvous ΔV required for different flight times are given in Tables 1 and 2. In this direct mode, then, the flight time controls Earth's final position relative to its launch position, and the satellite will make an integral number of revolutions to return to 1 AU.

Tables 1 and 2 show that the results of both transfer modes are very similar and that the cost in transfer time and ΔV are high. The most reasonable would be the 2.5-year transfer time and a ΔV of about 2 km/s. The C_3 value is low, only about 4 km^2/s^2. Fortunately, inserting Venus flybys can reduce the flight times to one year and eliminate the orbit insertion ΔV. The final satellite orbit, however, would be elliptical and not circular.

Single Venus Gravity Assist

Compared with direct to the far side of the sun, Venus gravity assist can avoid both long flight times and large ΔV for the station insertion into a one-year orbit. However, the condition, which is compromised, is the 1-AU circular orbit achieved with the direct mode. Instead, the one-year final orbit will have a perihelion roughly at Venus' orbit and an aphelion at 1.28 AU. Accepting an elliptic orbit may be a small price to pay to avoid a 2.5-year flight time and a 2.0-km/s insertion velocity requirement.

A launch to Venus requires about five months for a simple near-Hohmann transfer. Venus perihelion is 0.72 AU, which is about halfway between the 1.5- and 2.5-year flight time transfers shown in Table 1. The five-month transfer time is fortuitous because now,

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Table 1 Direct inward transfer to far side of the sun

Revolutions				
Earth, years	Spacecraft, number	Energy, C_3	Perihelion, AU	ΔV,[a] km/s
1.5	2.0	10.71	0.668	3.70
2.5	3.0	3.83	0.788	3.40
3.5	4.0	1.95	0.846	3.31

[a] From 200-km altitude parking orbit.

Table 2 Direct outward transfer to far side of the sun

Revolutions				
Earth, years	Spacecraft, number	Energy, C_3	Aphelion, AU	ΔV,[a] km/s
1.5	1.0	10.83	1.637	3.71
2.5	2.0	3.85	1.337	3.40
3.5	3.0	1.95	1.233	3.31

[a] From 200-km altitude parking orbit.

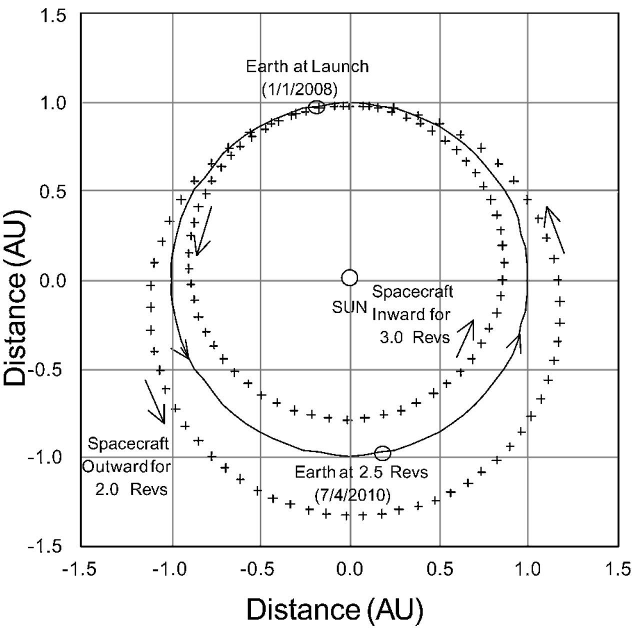

Fig. 1a Direct transfer to the sun's far side in an inertial heliocentric frame.

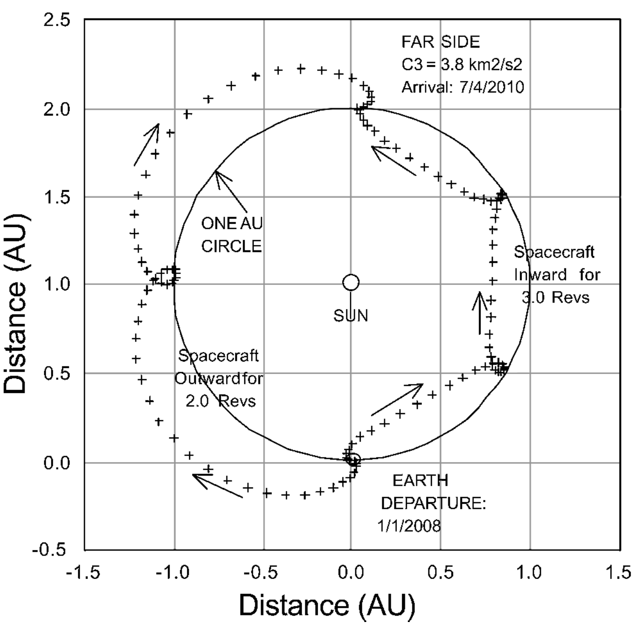

Fig. 1b Direct transfer to the sun's far side in Earth–sun rotating frame.

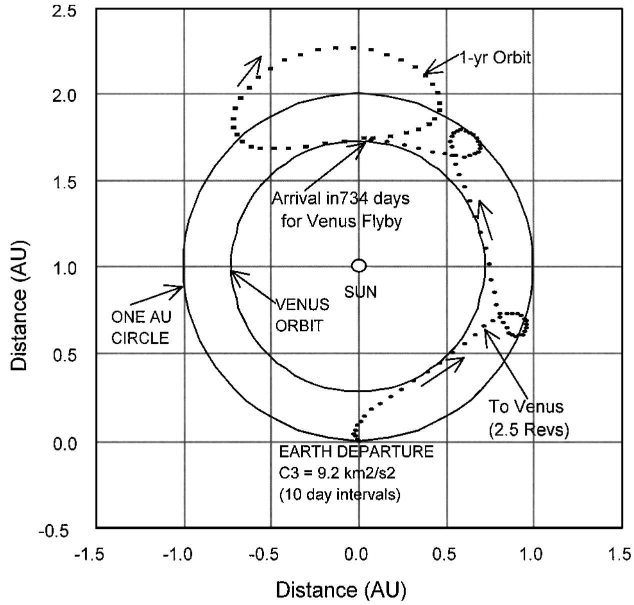

Fig. 2 2.5-revolution transfer to a Venus flyby and into a far-side, one-year orbit (rotating frame).

in a two-year period, the near-Hohmann transfer orbit will make 2.5 revolutions (in 5 × 5 months) so that the satellite will end up at Venus, and the Earth will return nearly to its launch location. The launch date, of course, must take the extra two revolutions of the transfer orbit into account in computing the Earth–Venus transfer. A typical transfer of this sort, in the rotating system, is shown in Fig. 2.

In Fig. 2 the Earth is fixed at the origin and the sun at 1.0 on the y axis. The orbit of the transfer to Venus has an aphelion of 1 AU and a perihelion of 0.72 AU. The spacecraft motion is ahead of the Earth and completes one revolution when it returns to 1 AU at about 75 deg from Earth. A second revolution takes it to about 150 deg. Another half-revolution takes it to an encounter with Venus where an unpowered flyby raises aphelion to 1.28 AU or to an orbit with a period of one year. In the rotating system this orbit is stationary with respect to the Earth–sun line, with its motion in the sky appearing to trace out a large loop from 23 deg east to 14 deg west relative to the sun. Figure 2 shows this one-year loop.

Gravity Assist Computation

As will be seen later, specifying terminal orbits for other than a one-year period will be useful for the design of the FSS. For this reason the capability to specify the postflyby heliocentric period was added to one of the author's trajectory programs[1,2] to avoid a difficult iterative search for the flyby conditions, which would give a desired period.

Using the point-conic model[3] for the Venus flyby calculations and a quite accurate three-dimensional analytic model for the planetary ephemerides,[4] a planetary encounter is approximated by translating the approach spacecraft heliocentric velocity vector $\boldsymbol{U}^-$ into a Venus-centered velocity vector $\boldsymbol{V}^-$ by subtracting the sun-relative Venus velocity vector $\boldsymbol{U}_v$ at encounter, or

$$\boldsymbol{V}^- = \boldsymbol{U}^- - \boldsymbol{U}_v \tag{1}$$

Similarly, the outbound hyperbolic velocity $\boldsymbol{V}^+$, which has the same magnitude as $\boldsymbol{V}^-$, would then be used in reverse to get the outbound heliocentric velocity $\boldsymbol{U}^+$, or

$$\boldsymbol{U}^+ = \boldsymbol{V}^+ + \boldsymbol{U}_v \tag{2}$$

The problem then is, given the inbound $\boldsymbol{V}^-$ and the desired heliocentric outbound period P_e, find the Venus flyby conditions.

The outbound semimajor axis of the orbit can be computed from

$$a_e = \left[\mu_s (P_e/2\pi)^2\right]^{\frac{1}{3}} \tag{3}$$

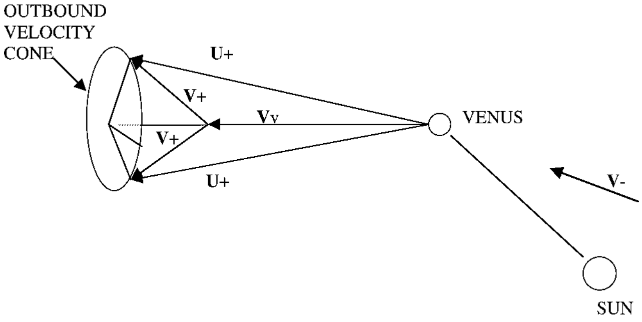

Fig. 3 Venus departure velocity cone for a fixed heliocentric orbit period.

Then, given the Venus distance from the sun r_v, the outbound heliocentric velocity of the spacecraft must be

$$U^+ = [\mu_s(2/r_v - 1/a_e)]^{\frac{1}{2}} \tag{4}$$

which is the magnitude of $\boldsymbol{U}^+$.

It is necessary to refer to Fig. 3 because there are multiple solutions depending on the inclination of the Venus flyby. In this figure the Venus velocity vector is known, but only the magnitudes of the outbound $\boldsymbol{U}^+$ and $\boldsymbol{V}^+$ are known. Because three sides of the vector diagram are known, the set of solutions consist of this vector triangle rotated about the Venus velocity vector. The vector $\boldsymbol{V}^+$ forms a cone, and specifying a cone angle determines the vector directions of $\boldsymbol{V}^+$ and $\boldsymbol{U}^+$.

The next step is to choose a specific cone angle and use the resulting inbound and outbound vectors ($\boldsymbol{V}^-$ and $\boldsymbol{V}^+$) to compute the specific Venus flyby inclination and altitude. The details will not be given here (see www.lws.gsfc.nasa.gov), but the two vectors give the inclination of the hyperbolic flyby, and the angle between them gives the Venus flyby altitude. The altitude can be examined to see if the flyby passes beneath the Venus surface. These solutions are obviously rejected.

Furthermore, a given cone angle will specify a given outbound heliocentric velocity vector, which together with the Venus position vector will determine the heliocentric orbit elements. Again, the characteristics of the orbit can be examined to see if some constraint is violated, such as passing too close to the sun for communication with Earth.

In the vector triangle of Fig. 3, no solution is available if the magnitude of $\boldsymbol{U}^+$ is greater than the sum of U_v and V^+. Because the Venus orbital velocity is fixed, this implies that $\boldsymbol{V}^-$, which has the same magnitude as $\boldsymbol{V}^+$, must have a minimum value. That is, the approach V^- must be larger than a certain value, which might restrict minimum energy transfers to Venus. Fortunately, suitable transfers can usually be found close to the minimum for a one-year orbit, with a large enough V^-.

Double Venus Gravity Assist

An alternative to performing two-and-one-half revolutions before arriving at Venus for the gravity assist into a one-year orbit, it is possible to insert an intermediate Venus gravity assist to decrease the heliocentric period and advance faster ahead of the Earth. Specifically, a direct transfer to Venus could be flown for a gravity assist into a Venus-type circular orbit. This initial Venus gravity assist would require a flyby over the north or south pole so that the outbound $\boldsymbol{V}^+$ would be perpendicular to $\boldsymbol{U}_v$. The result would be an inclined Venus-type orbit having a period, like Venus of 225 days, or about seven months. In this time the spacecraft would advance about 140 deg ahead of Earth. This advance, together with the 35-deg advance during the Earth–Venus transfer, would place the spacecraft almost directly on the sun's far side, where a second Venus encounter would result in a one-year orbit about the sun.

This transfer, the two flybys, and the one-year orbit are shown in Fig. 4a for the type 1 trajectory. The longer type 2 is shown in Fig. 4b. The launch date chosen is suitable for the FSS. In addition, the Earth-sky view, spacecraft radial distance from Earth, and spacecraft-sun radial velocity are given in Figs. 5 and 6. For both type trajectories the Venus arrival date for the first flyby is chosen so that the approach

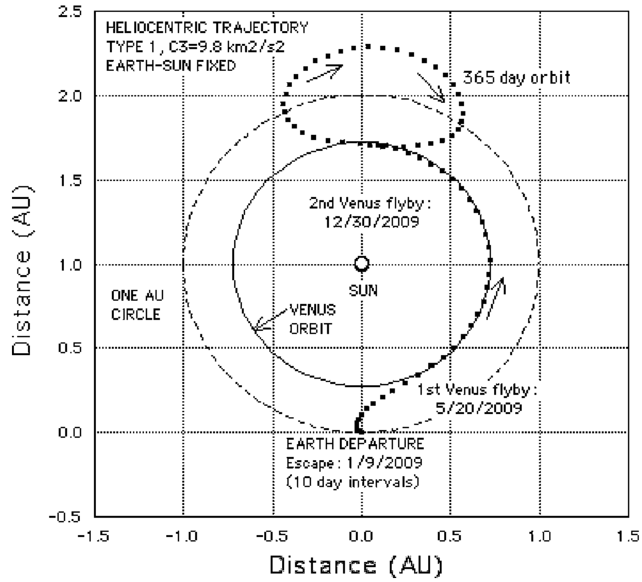

Fig. 4a Type 1 double Venus flyby into a far-side, one-year orbit in an Earth–sun rotating frame.

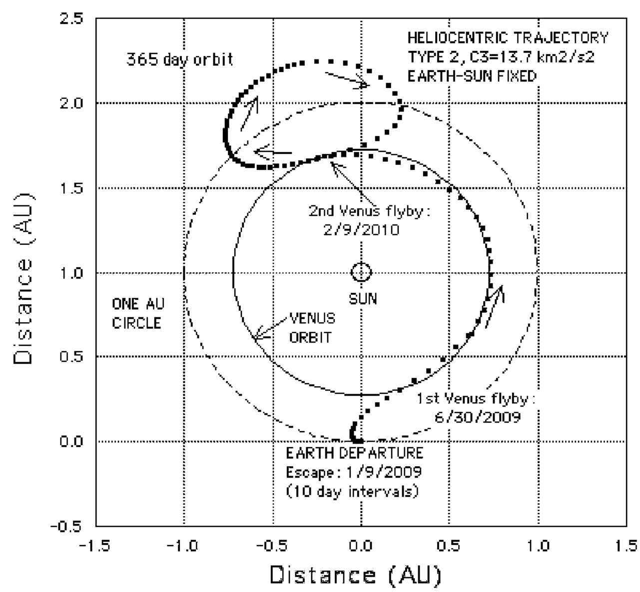

Fig. 4b Type 2 double Venus flyby into a far-side, one-year orbit in an Earth–sun rotating frame.

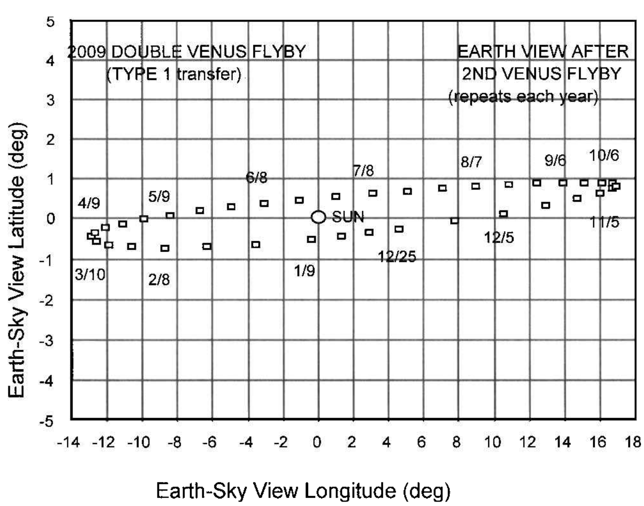

Fig. 5 Earth view of the far-side, one-year orbit period spacecraft.

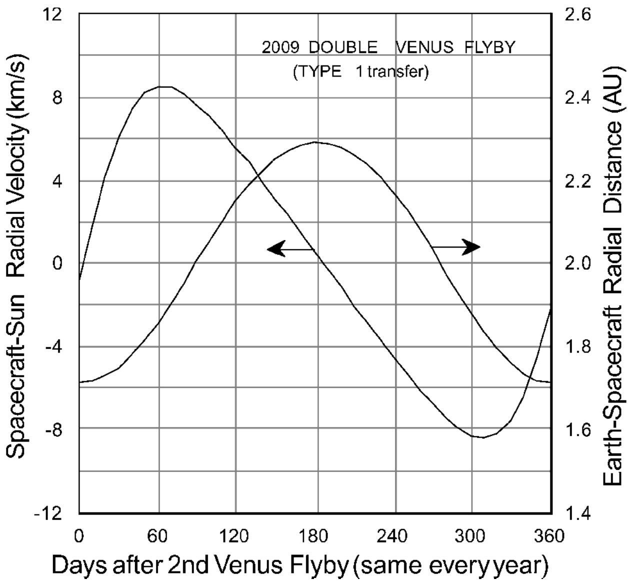

Fig. 6 Earth distance and solar radial velocity of the one-year orbit period spacecraft.

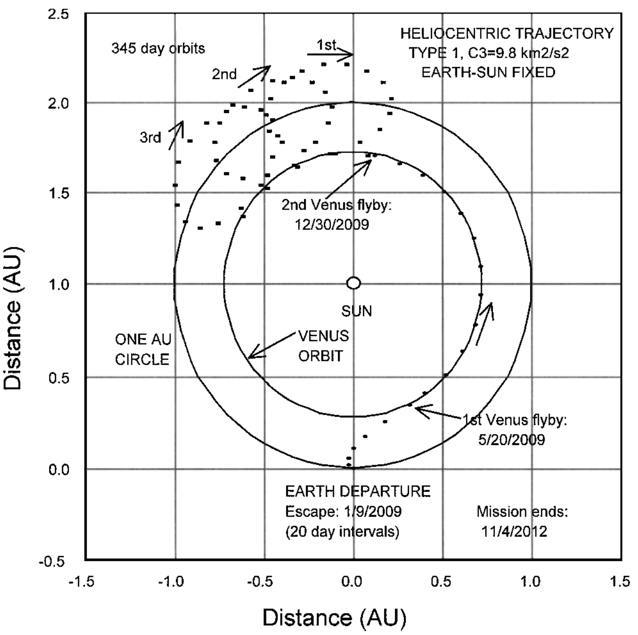

Fig. 7 Double Venus flyby into a sentinel 345-day third-quadrant mission.

velocity magnitude V^- (which is the same at the second flyby) is large enough to enter the one-year orbit.

Far-Side Sentinel

The fact that there is a choice of cone angle for a given desired heliocentric period, as shown in Fig. 3, allows other constraints to be satisfied. For example, a requirement might be to enter a one-year orbit and have a solar occultation, as seen from Earth, once or twice a year, for precise relativity measurements of the bending of light. Therefore, a particular cone angle might result in this desired orbit. For the FSS an alternate cone angle would be chosen to avoid such close passages of the sun, which would result in communication interference with Earth.

Another desirable requirement for the FSS spacecraft is for it to drift in the third quadrant over a period of about three years. This slow drift can be accomplished by decreasing the period of the heliocentric orbit entered after the second Venus flyby, so that during each revolution it will move forward faster than the Earth. It is found, through simulation trials, that a period of 345 days will give the desired results. Figure 7 presents an example trajectory plot for these orbits for a type 1 transfer to Venus. The type 2 transfer (not shown) is found to be most favorable if a later Venus arrival date is used, which causes the first 345-day orbit to be further into the third quadrant, avoiding communication interference with the sun.

Another LWS Application

A double Venus gravity assist may also be useful for inserting a set of Inner Heliospheric Sentinels, which is a mission currently being examined for the LWS initiative. Here, several satellites would be placed in orbits interior to Earth's orbit to view the sun on all sides simultaneously. In this case the Earth location would not be a major consideration, so that orbits less than one year could be chosen. The Venus period in this case (which is 225 days) would play a dominant role.

As a simple example, consider that four satellites are launched to Venus on the type 1 trajectory as shown in Fig. 7. At the first Venus encounter one spacecraft can be aimed for a close flyby to place it in a 180-day heliocentric orbit. This short period would place perihelion distance at about 0.5 AU. The Venus flyby altitude would be about 4025 km. The other three spacecraft have flybys, which place them into a Venus-type orbit that will encounter Venus again 225 days later. The spacecraft placed into the 180-day orbit will perform one complete revolution and then travel another 90 deg in the 225 days it takes for the other spacecraft to encounter Venus, and so it will lead Venus by 90 deg. After one Venus revolution a second spacecraft will perform a flyby to enter the 180-day heliocentric orbit, and this spacecraft will be positioned 90 deg behind the first. This process will be repeated for the two spacecraft remaining, resulting in the placement of four satellites in exactly the same orbit (because Venus returns to the same inertial location for each flyby), but displaced by 45 days in their location on the orbit. The time from the first satellite insertion to the fourth will be three Venus revolutions or less than two years. Other scenarios for placement of these four, or more, satellites could probably be devised for which Venus gravity assist would be beneficial.

Conclusions

It has been shown that Venus gravity assist can play a major role in the placement of satellites in the inner solar system. Here, for the FSS, a double Venus gravity assist can position a satellite on the far side of the sun in a year, and for no deterministic ΔV if an elliptic orbit is acceptable. It is also shown that if only the period of the post-Venus flyby is specified, there will be multiple solutions, allowing another condition to be imposed on the trajectory. A computational process is presented by which the set of these possible solutions can be found. Finally, a general method of placing a network of small satellites evenly spaced around the sun using Venus gravity assist is discussed briefly, where these satellites can be flown to Venus in a single launch period and perhaps on a single launch vehicle. In all of these cases, a point-conic model of trajectories at Earth and Venus is assumed, which is sufficient to determine initial configurations; however, these results should be verified with more precisely integrated trajectories.

Acknowledgments

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