

Rendezvous/Proximity Operations Workbook

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Foreword

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This training material is part of a series of lectures, handouts, and workbooks for instruction on Proximity Operations/ Rendezvous.

This training package should be studied before attending any classroom session on this material or taking any lesson for which this is a prerequisite.

The Proximity Operations and Rendezvous lesson sequence chart shows where this lesson fits in the series, the prerequisites for each lesson, and the optimum presentation order. The lesson sequence charts are located in the Shuttle Training Program Catalog.

This material is for training purposes only; it should not be used as a source of operational data.

The last page of this workbook is a *lesson critique sheet*. This sheet is used to obtain your comments on this workbook. Please complete it and mail to the training coordinator when you have completed this workbook. (It need not be placed in an envelope nor be signed.)

Section 1: Orbital Mechanics Concepts with Rendezvous/Proximity Operations Applications

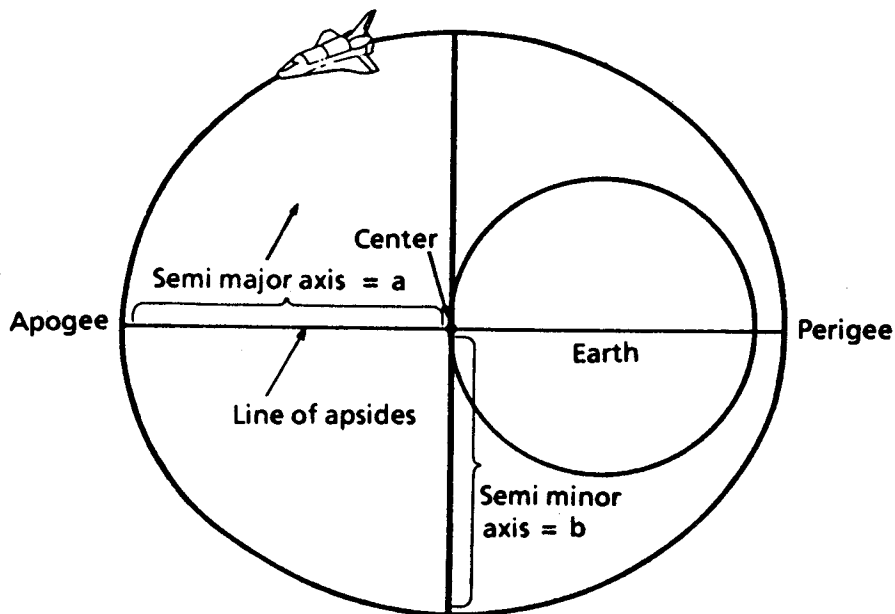
Introduction

The first section of this workbook includes a review of basic concepts in orbital mechanics which are necessary to understand rendezvous and proximity operations. Also, some of the commonly used terms are described.

Orbit Definitions

From the laws of physics, it can be shown that the path of a body (such as a spacecraft) in orbit about the Earth is an ellipse, with the Earth's center as one focus (see figure 1-1.) The point on the ellipse at which the spacecraft is closest to the Earth is called perigee, while the point farthest from the Earth is called apogee.

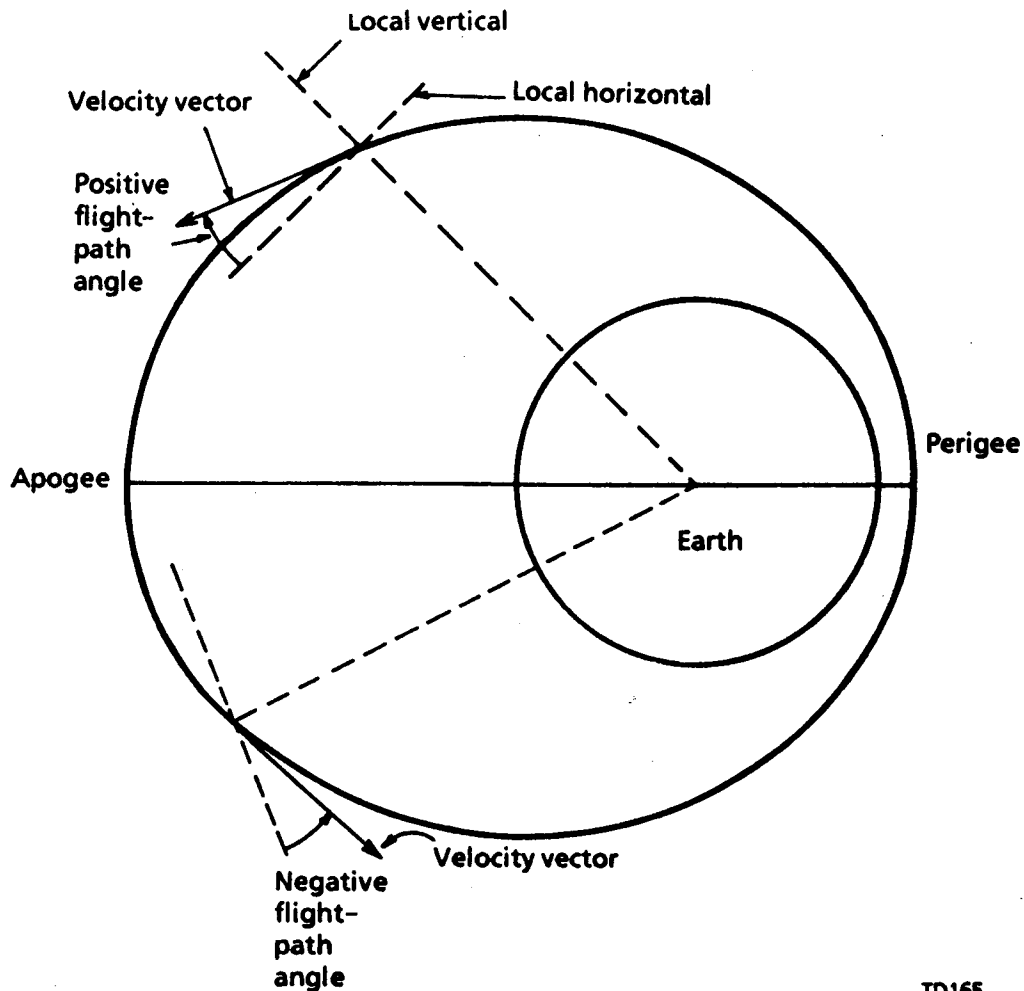
The height of apogee or perigee is typically given in distance above the surface of the Earth. Each perigee and apogee is also called an apside, and the major axis of the ellipse which passes through both apogee and perigee is called the line of apsides. Note that there is no line of apsides in a circular orbit. The line containing apogee and perigee is also the major axis of the ellipse defined by the spacecraft's orbit. The semi-major axis is the segment between the center of the ellipse and apogee or perigee. Often, its length is denoted by "a." Thus, the major axis has length "2a." The minor axis of an ellipse is the line perpendicular to the major axis containing the center of the ellipse. As before, the semi-minor axis is the portion of the minor axis from the center to the ellipse, and its length is often denoted by "b." Thus, the minor axis has length "2b".



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Figure 1-1.— Elliptical orbit definitions.

Recall that local vertical is defined as the line from an orbiting spacecraft to the center of the Earth, and that local horizontal is the line through the spacecraft which is perpendicular to local vertical and lies in the spacecraft's orbit plane. Therefore, the flightpath angle is the angle between local horizontal and the spacecraft's velocity vector (the line tangent to the spacecraft's orbit) (see figure 1-2). Note also that if a spacecraft's flightpath angle is zero, the spacecraft is at apogee, perigee, or in a circular orbit. If a spacecraft is in a circular orbit, its flightpath angle is constantly equal to zero. Furthermore, the flightpath angle is positive as the spacecraft is traveling from perigee to apogee and is negative as it travels from apogee to perigee.



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Figure 1-2.- Flightpath angle.

Another pair of useful terms concerning a spacecraft's orbit are *ascending node* and *descending node*. As illustrated in figure 1-3, the nodes of an orbit (with respect to the Earth's equatorial plane) are formed by the intersection of the spacecraft's orbit path and the Earth's equatorial plane. The node formed as the spacecraft travels from south of the equator to north of the equator is called the ascending node, while the node formed as the spacecraft travels from north to south is called the descending node. The line containing the ascending and descending nodes of an orbit is called the line of nodes.

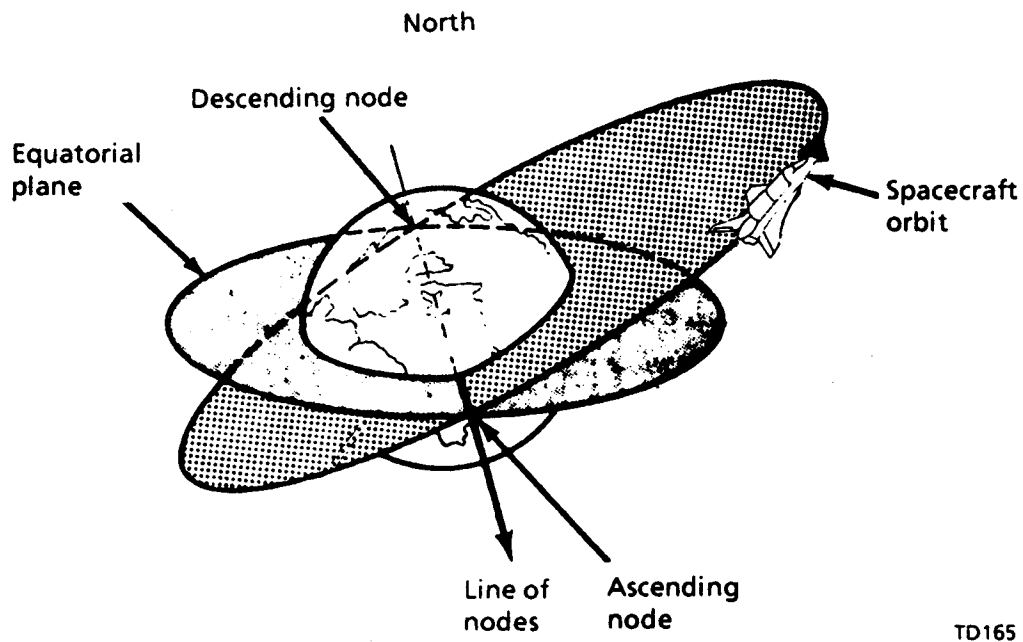


Figure 1-3.— Nodes of an orbit.

The word *node*, however, is also used in the more general sense to refer to the intersection point of a spacecraft's orbit with another orbital plane (as opposed to the Earth's equatorial plane). Another useful term along these lines is that of a wedge angle, which is defined as the angle between two intersecting orbit planes (figure 1-4).

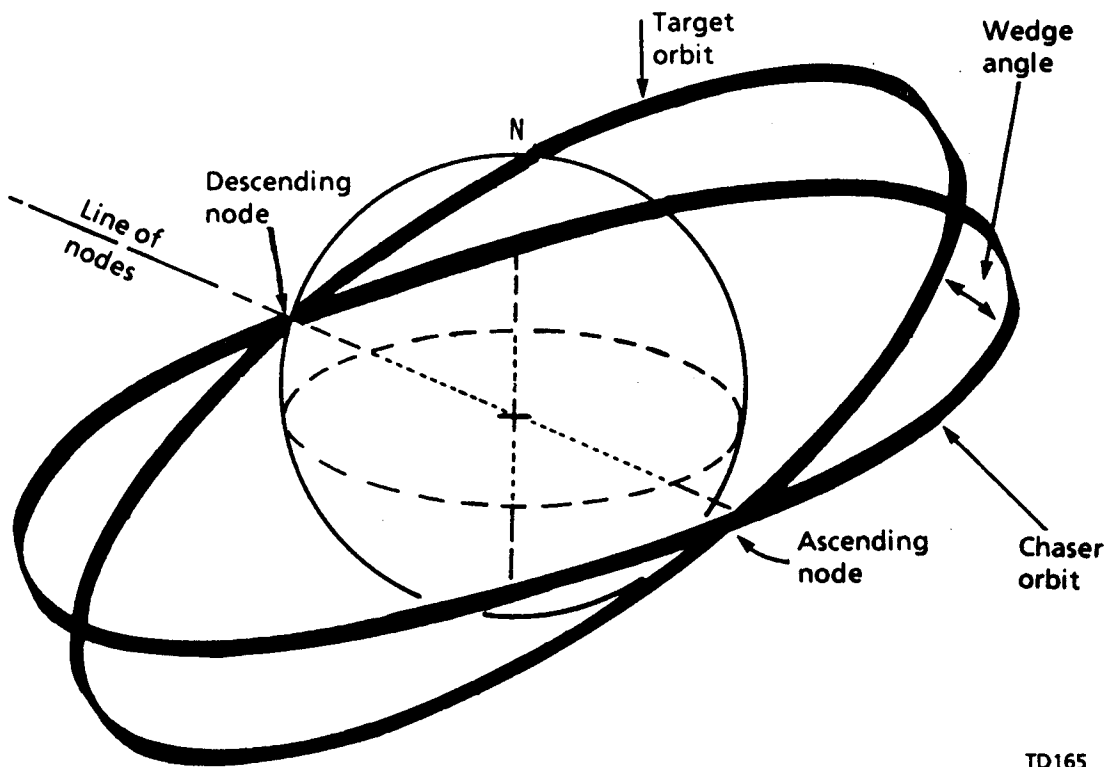


Figure 1-4.- Wedge angle.

Orbital Period/Velocity

As an extension of Kepler's third law, the square of the period of a spacecraft orbiting the Earth is proportional to the cube of its mean distance from the center of the Earth. To be more exact, if a spacecraft is in orbit and

$$\mu = G \times M = \text{Gravitational constant} \times \text{mass of the Earth} = 1.408 \times 10^{16} \text{ ft}^3/\text{s}^2$$

$$R_a = \text{Height of apogee from the center of the Earth (Note: the equatorial radius of the Earth is approximately 3444 n. mi.)}$$

$$R_p = \text{Height of perigee from the center of the Earth}$$

$$a = \frac{R_a + R_p}{2} = \text{length of the semimajor axis}$$

Then the period, t , of the spacecraft can be written as

$$t^2 = \frac{4\pi^2 a^3}{\mu} \text{ or } t = 2\pi \mu^{-1/2} a^{3/2}$$

As an example, suppose a spacecraft is in a 140 x 150 n. mi. orbit. (Perigee height is 140 n. mi. and apogee height is 150 n. mi. from the Earth's surface.) Then,

$$\mu = 1.408 \times 10^{16} \text{ ft}^3/\text{s}^2$$

$$R_a \sim 150 + 3444 \text{ n. mi.} = 2.184 \times 10^7 \text{ ft}; \\ 1 \text{ n. mi.} = 6077 \text{ ft}$$

$$R_p \sim 140 + 3444 \text{ n. mi.} = 2.1777 \times 10^7 \text{ ft}$$

$$a = 2.181 \times 10^7 \text{ ft}$$

$$t = 5.39 \times 10^3 \text{ sec} = 89.89 \text{ min}$$

It can be observed from the equation that the period of an orbit depends on its semi-major axis, and thus on the sum of the apogee and perigee heights and not on the orbit shape (its semi-minor axis). Also, it shows that

the period of an orbit increases with its average altitude (the average of the apogee and perigee heights), and so its average angular rate decreases with its average altitude. Thus, it is sometimes said that a lower orbit is faster (higher average angular velocity) than a higher orbit.

As a result of orbital mechanics, it also is possible to determine the velocity (V) of a spacecraft at any point in its orbit in terms of its distance, r , from the center of the Earth at that point. Assuming the definitions of μ , R_a , R_p , and a from above, then

$$V^2 = \mu \left(\frac{2}{r} - \frac{1}{a} \right) \text{ or}$$

$$V = \left\{ \mu \left(\frac{2}{r} - \frac{1}{a} \right) \right\}^{1/2}$$

From the previous example, this means that the velocity at perigee (V_p) is given by

$$V_p = 25,446.7 \text{ ft/s}$$

and the velocity at apogee (V_a) is given by

$$V_a = 25,373.2 \text{ ft/s}$$

From the equation for velocity above, it is apparent that maximum velocity of the spacecraft is at perigee, and its minimum velocity is at apogee. Furthermore, its velocity decreases from perigee to apogee and increases as it goes from apogee to perigee. This fact is related to an extension of Kepler's second law: A spacecraft in an elliptical orbit about the Earth sweeps out equal areas in equal amounts of time. This means that a spacecraft's angular velocity at perigee is greater than its angular velocity at apogee.

Note also that if a spacecraft is in a circular orbit, a in the above equation is equal to r and is the radius of the circular orbit. The equation then becomes simply

$$V = (\mu/r)^{1/2}$$

where the velocity of the spacecraft remains constant. This velocity is called the circular velocity at the altitude, r .

This means that the velocity of a spacecraft in a lower circular orbit is greater than the velocity of a spacecraft in a higher circular orbit. The relationship between orbit size and velocity for noncircular orbits is not as clear cut.

Orbital Inclination/Launch Azimuth Angle

The inclination of an orbit is defined as the angle between the orbital plane and the equatorial plane (figure 1-5(a)). A spacecraft launched at the equator may be forced to any orbital inclination by altering the direction of its velocity at launch. At any other latitude, however, there is a minimum orbital inclination that can be achieved, which is equal to the latitude of the launch site, unless out-of-plane burns are made after launch. As changing the orbital inclination is very expensive fuel-wise, only a limited capability (to be determined (TBD) degrees) exists to change the orbital inclination after launch. As a reference point, Kennedy Space Center (KSC) is at 28.5° north latitude, and so the minimum attainable orbital inclination achieved with a launch from there, with no other out-of-plane burns, would be 28.5°. Figure 1-5(b) shows the groundtrack of a spacecraft launched due east from Kennedy Space Center. Points A and C correspond to the northern- and southern-most points of the orbit in figure 1-5(a). Point B identifies the descending node.

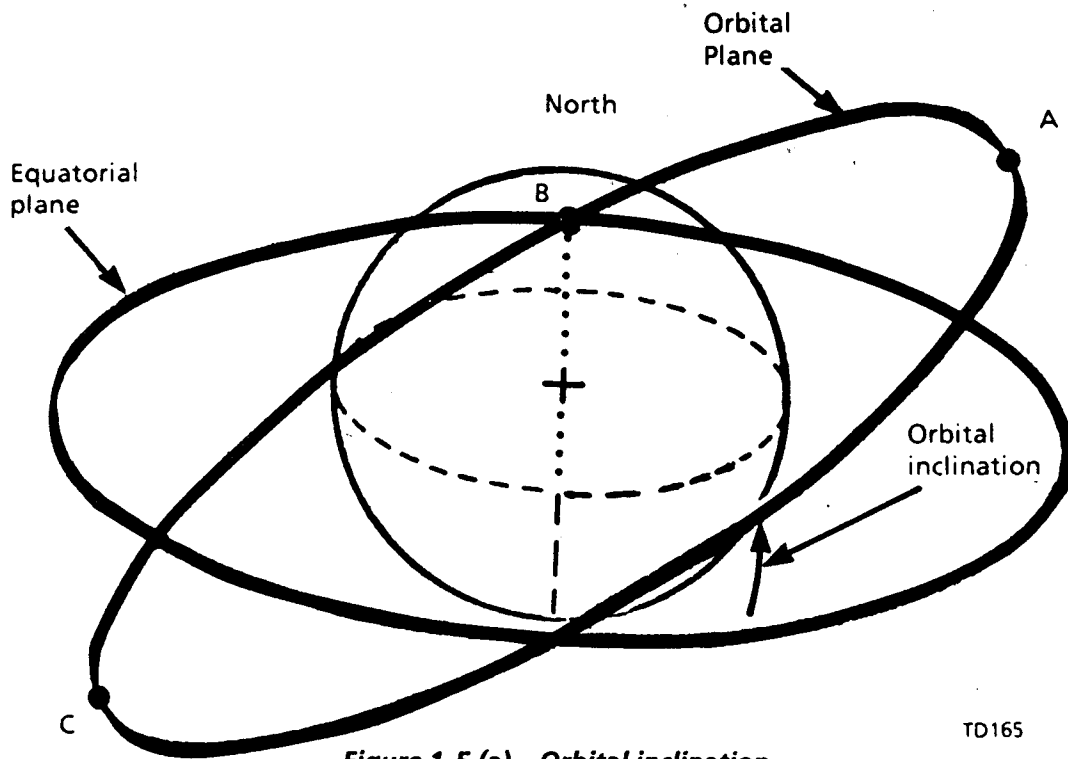


Figure 1-5.(a).- Orbital inclination.

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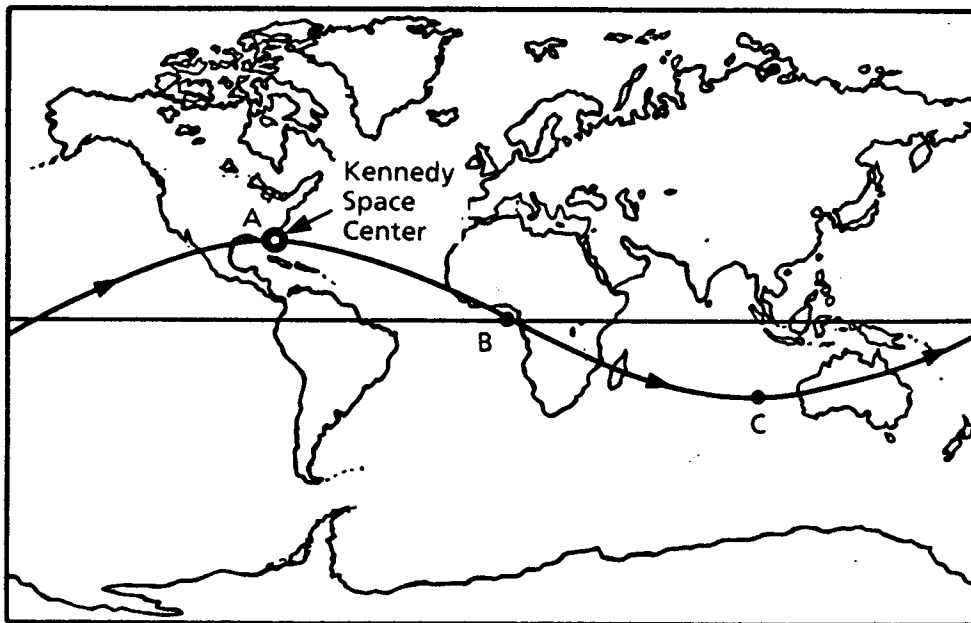


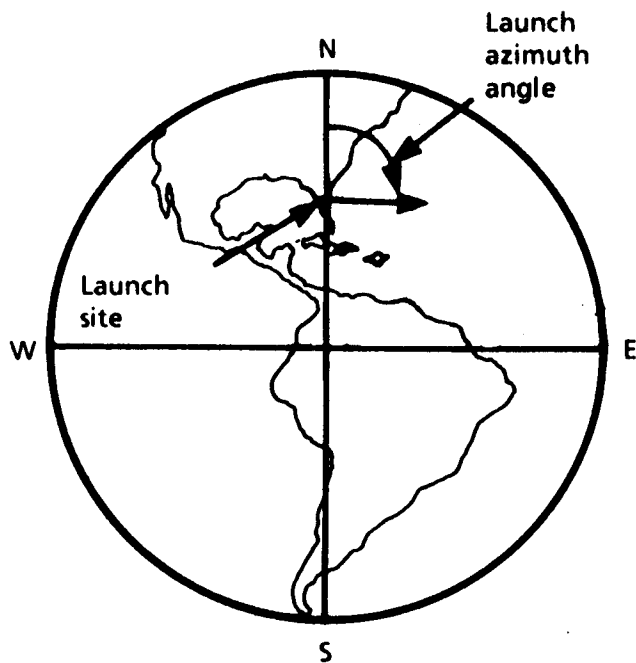
Figure 1-5.(b).- Groundtrack of orbital inclination.

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Figure 1-5.- Orbital inclination and launch azimuth angle.

The launch azimuth angle is the angle between due north and the direction in which the spacecraft is launched. For a launch at the equator, the orbital inclination is equal to the difference between 90° and the launch azimuth angle.

As another example, if the spacecraft were launched directly east from KSC, its launch azimuth angle would be 90° and its orbital inclination would be 28.5° (figure 1-5(c)).



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Figure 1-5(c).— Launch azimuth angle.

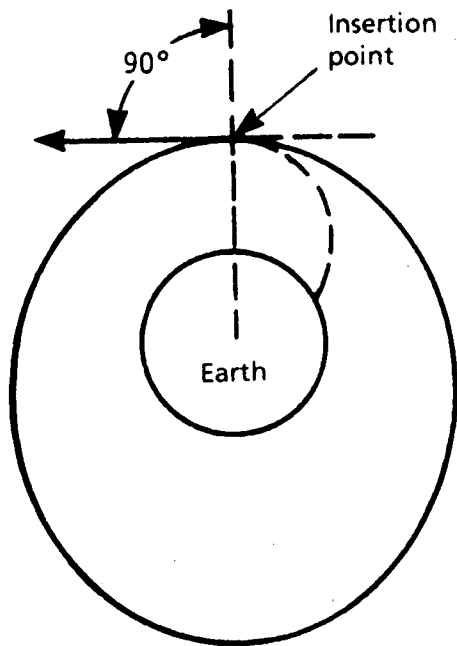
Figure 1-5.— Concluded.

Orbit Insertion

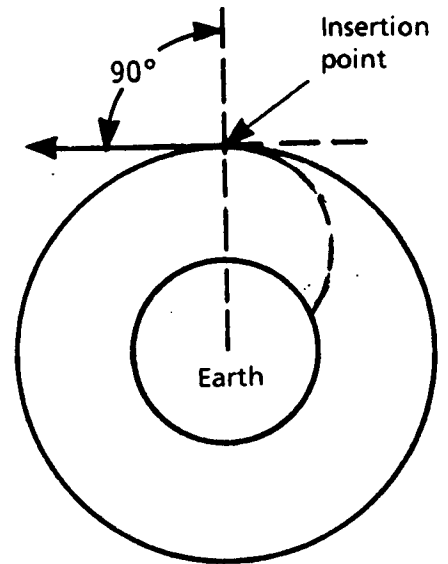
While the inclination of a spacecraft's orbit is primarily determined at launch, the size and shape is determined by its velocity (magnitude and direction) at the point at which powered flight ends. This point when powered flight ends (i.e., main engines cut off) is called the insertion point.

The angle between local horizontal and the spacecraft's velocity vector is called the

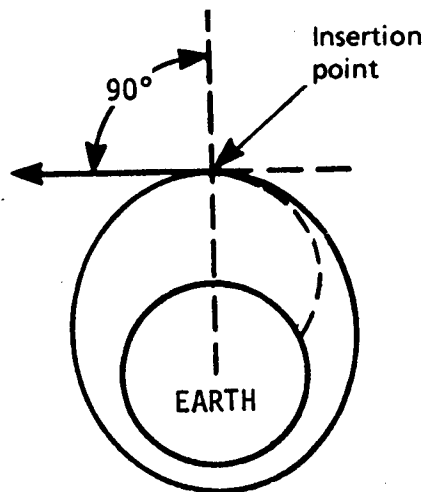
flight path angle. If the flightpath angle is zero at the insertion point, then that point will be an apsis of the resulting orbit. If the velocity equals circular velocity ($V_c = (\mu/r)^{1/2}$) as defined in the Orbital Period/Velocity section, the resultant orbit will be circular. If the velocity is less than circular velocity, the insertion point will coincide with apogee; and if the velocity is greater than circular velocity, the insertion point will correspond with perigee. The rule of thumb is that with an initial flightpath angle of 0° , the height of the spacecraft on the opposite side of the orbit is directly related to the insertion velocity (figure 1-6).



Insertion velocity > Circular velocity



Insertion velocity = Circular velocity



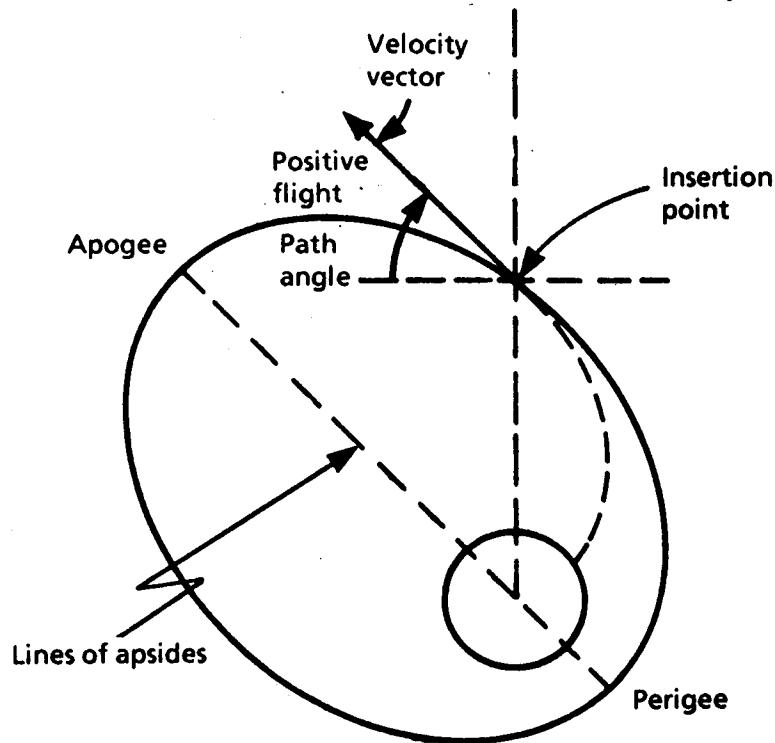
Insertion velocity < Circular velocity

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Figure 1-6.- Effect of insertion velocity with zero insertion flightpath angle.

If the flightpath angle at the insertion point is positive, the spacecraft will be moving toward apogee and will reach apogee somewhere within the next 180° of orbit travel. The exact point (within the next 180°) apogee will occur depends on the magnitude of the insertion velocity and flightpath angle. Conversely, if the insertion flightpath angle is negative, the spacecraft is moving toward perigee and will reach it within the next 180° of

orbit travel. Once again, the exact location of perigee will depend on the flightpath angle and the magnitude of the insertion velocity. However, if the magnitude of the insertion velocity is approximately equal to circular velocity ($V_c = (\mu/r)^{1/2}$) at that altitude, the line of apsides of the new orbit is approximately parallel to the insertion velocity vector (figure 1-7).



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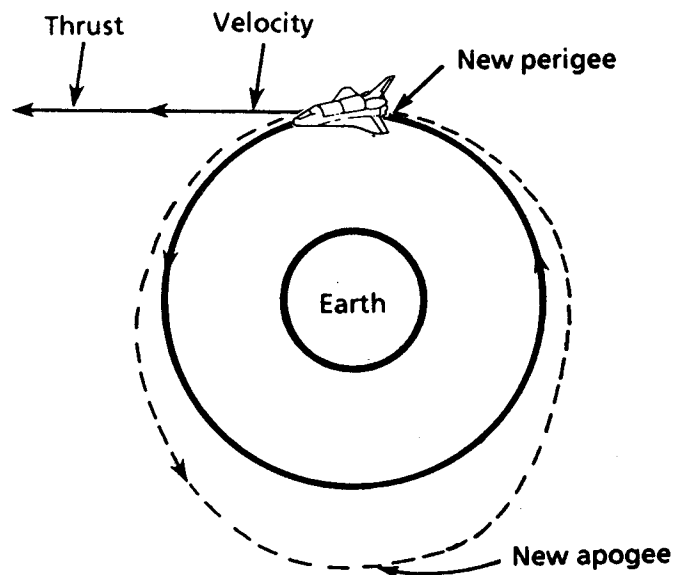
Figure 1-7.—Effect of positive insertion flightpath angle (insertion velocity = circular velocity).

Orbital Maneuvers – Single Impulse

Following orbital insertion at the end of powered flight, a spacecraft's orbit is fixed, barring secondary effects such as drag, etc., until a thrust is made by the vehicle (e.g.; with the OMS engines, the RCS thrusters, or venting). The direction and magnitude of the thrust, as well as the point at which it is applied, will contribute to the resultant orbit shape. An important fact here is that the thrust point will be common to the prethrust and postthrust orbits. Basically, any such thrust can be thought of as a composition of four types of thrusts: posigrade, retrograde, radial (outward or inward), and out-of-plane.

Posigrade Burn

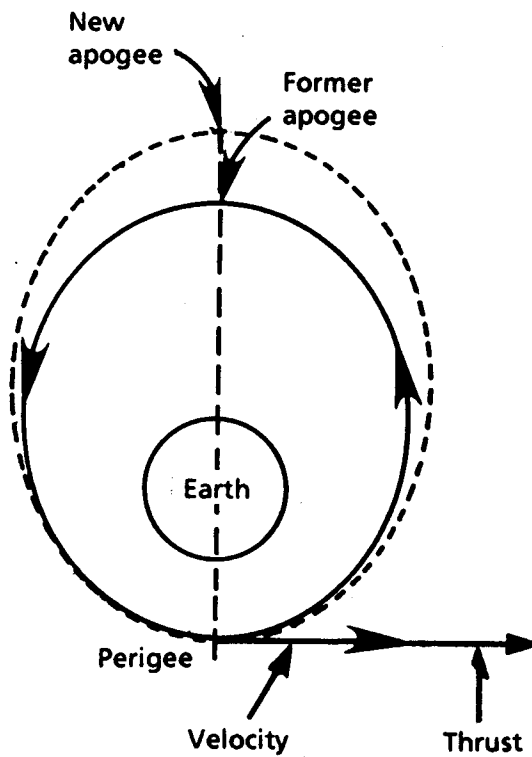
A posigrade burn is one which increases the speed but does not change the direction of the spacecraft at the point it is applied. A posigrade burn will raise every point of the orbit except the thrust point. If the orbit was originally circular, a posigrade maneuver will establish a line of apsides, with the thrust point becoming perigee and apogee occurring 180° of orbit travel away (figure 1-8).



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Figure 1-8.— Posigrade burn in a circular orbit.

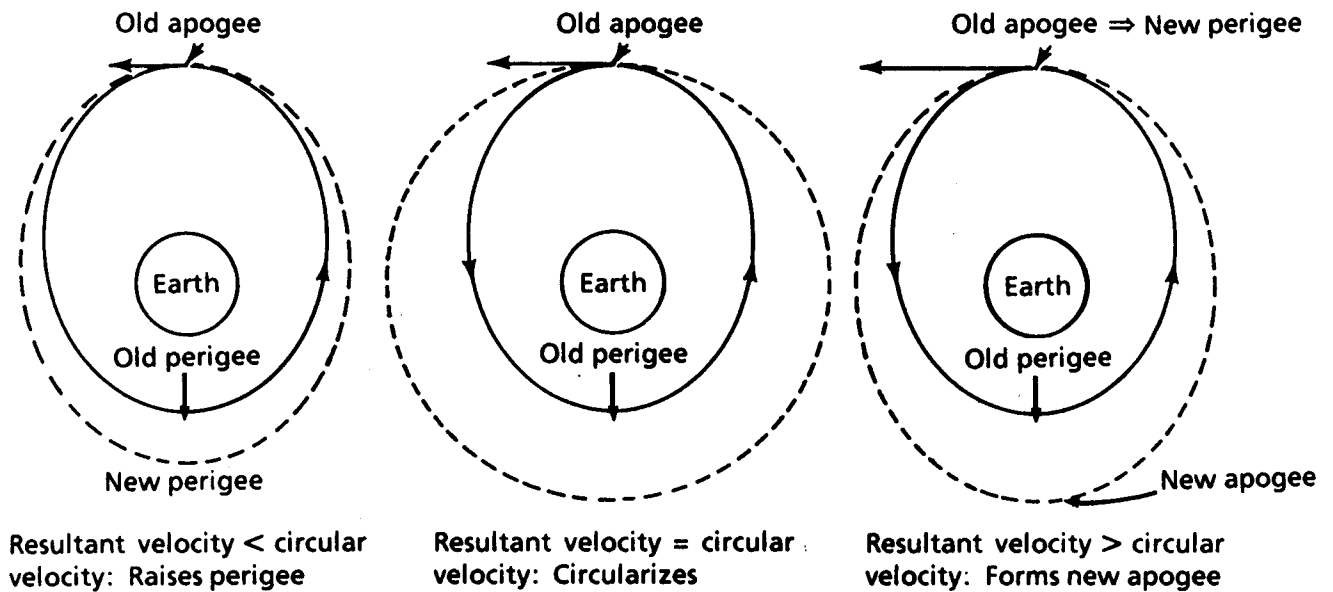
If the original orbit was not circular, the results could vary depending on where in the orbit the thrust is applied. As shown in figure 1-9, a posigrade burn at perigee will always raise the height of apogee.



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Figure 1-9.—Posigrade burn at perigee.

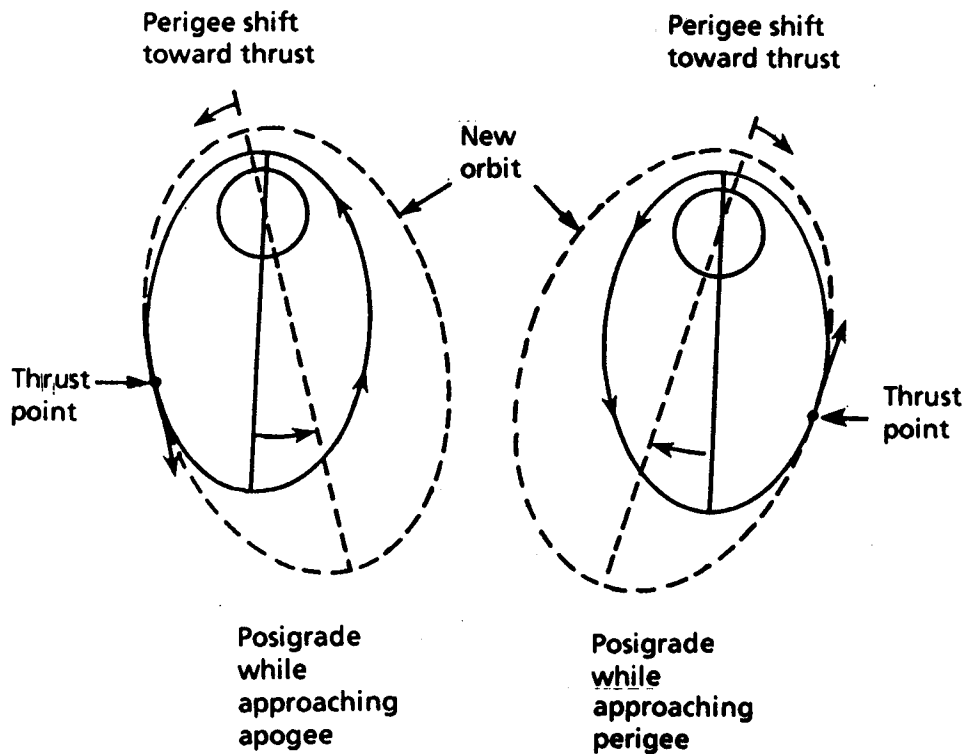
On the other hand, a posigrade thrust at apogee will either raise the perigee height, circularize the orbit at the former apogee height, or cause the thrust point to become perigee in the new orbit. Which one will happen depends on whether the resultant velocity is less than, equal to, or greater than circular velocity at the former apogee altitude (figure 1-10).



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Figure 1-10.— Posigrade burn at apogee.

Observe that a posigrade burn at apogee or perigee will not shift the line of apsides. However, a posigrade burn at a point in an elliptical orbit other than apogee or perigee will rotate the line of apsides, as well as increase the semi-major and semi-minor axes. The rule of thumb is that a posigrade thrust moves perigee closer to the thrust point. This results in a clockwise or counterclockwise rotation, depending on whether the thrust occurred while approaching perigee or apogee, respectively. (See figure 1-11.)

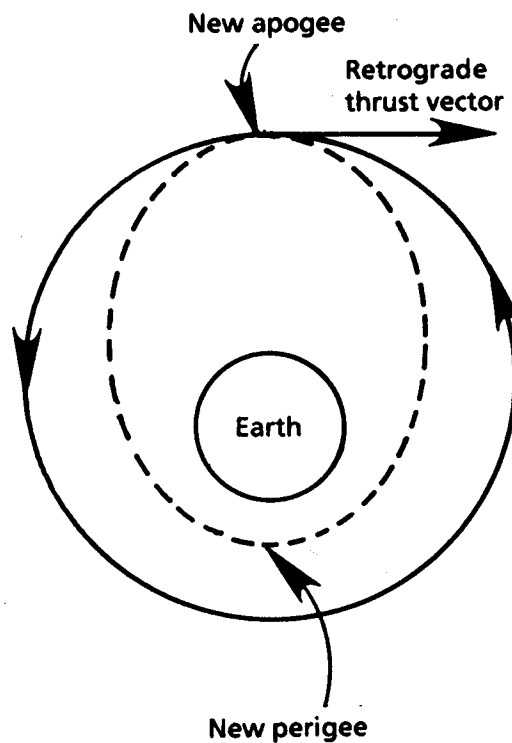


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Figure 1-11.—Posigrade burn at a point other than apogee or perigee.

Retrograde Burn

A retrograde burn is one which decreases the speed but does not change the direction of the spacecraft at the thrust point. Thus, it will lower every point of the orbit except the thrust point. If the orbit was originally circular, a retrograde maneuver will establish a line of apsides with the thrust point becoming the apogee and the perigee occurring 180° of orbit travel away (see figure 1-12).

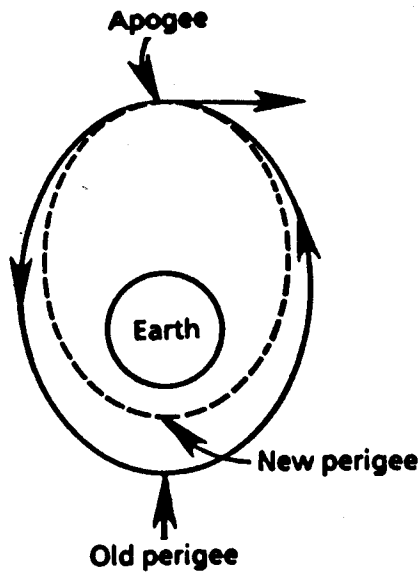


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Figure 1-12.— Retrograde burn in circular orbit.

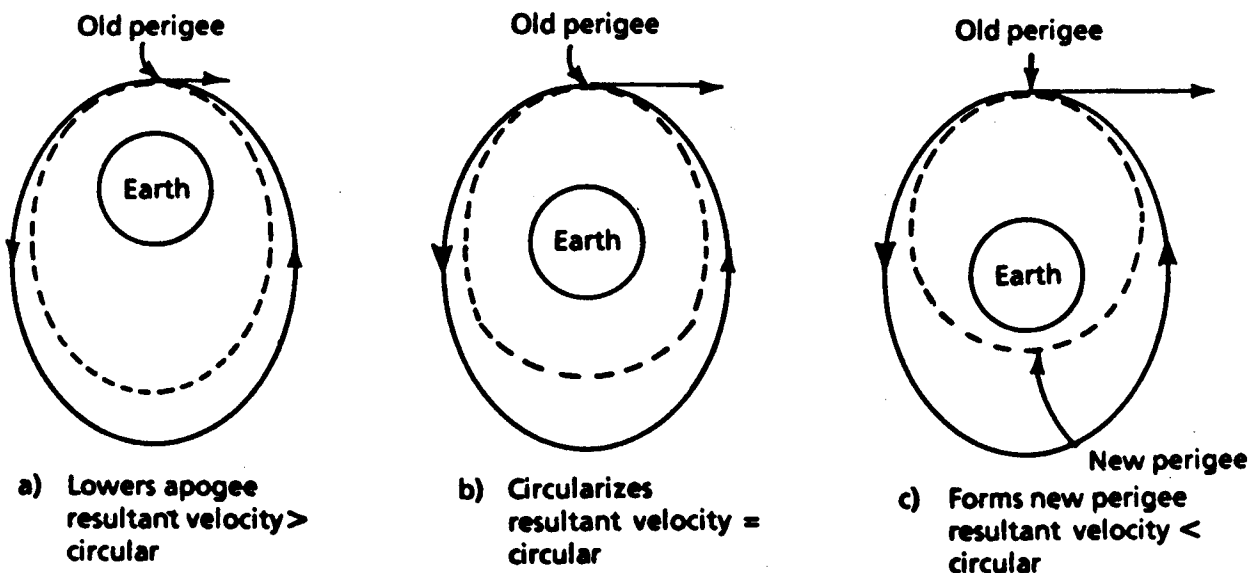
As with posigrade burns, if the original orbit were elliptical, the results of a retrograde burn would vary depending on the resultant velocity at the thrust point and where in the orbit the thrust took place. A retrograde burn at apogee would simply lower perigee in the resultant orbit, as shown in figure 1-13.

A retrograde burn at perigee will, of course, lower the orbit 180° away from the burn, and depending on the resultant velocity, will either lower apogee, circularize the orbit at the perigee height, or lower the orbit so much that the thrust point becomes the apogee of the new orbit. This depends on the resultant velocity at the thrust point (see figure 1-14).



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Figure 1-13.—Retrograde burn at apogee.



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Figure 1-14.—Retrograde burn at perigee.

The important point to remember is that a retrograde burn at apogee or perigee does not rotate the lines of apsides. A retrograde burn at a point in an elliptical orbit other than apogee or perigee will rotate the line of apsides, as well as decrease the semi-major and semi-minor axes. The rule is that a retrograde burn moves perigee farther away from the thrust point, with the resulting rotation of the line of apsides being clockwise or counterclockwise, depending on whether the thrust occurred while approaching apogee or perigee, respectively (see figure 1-15).

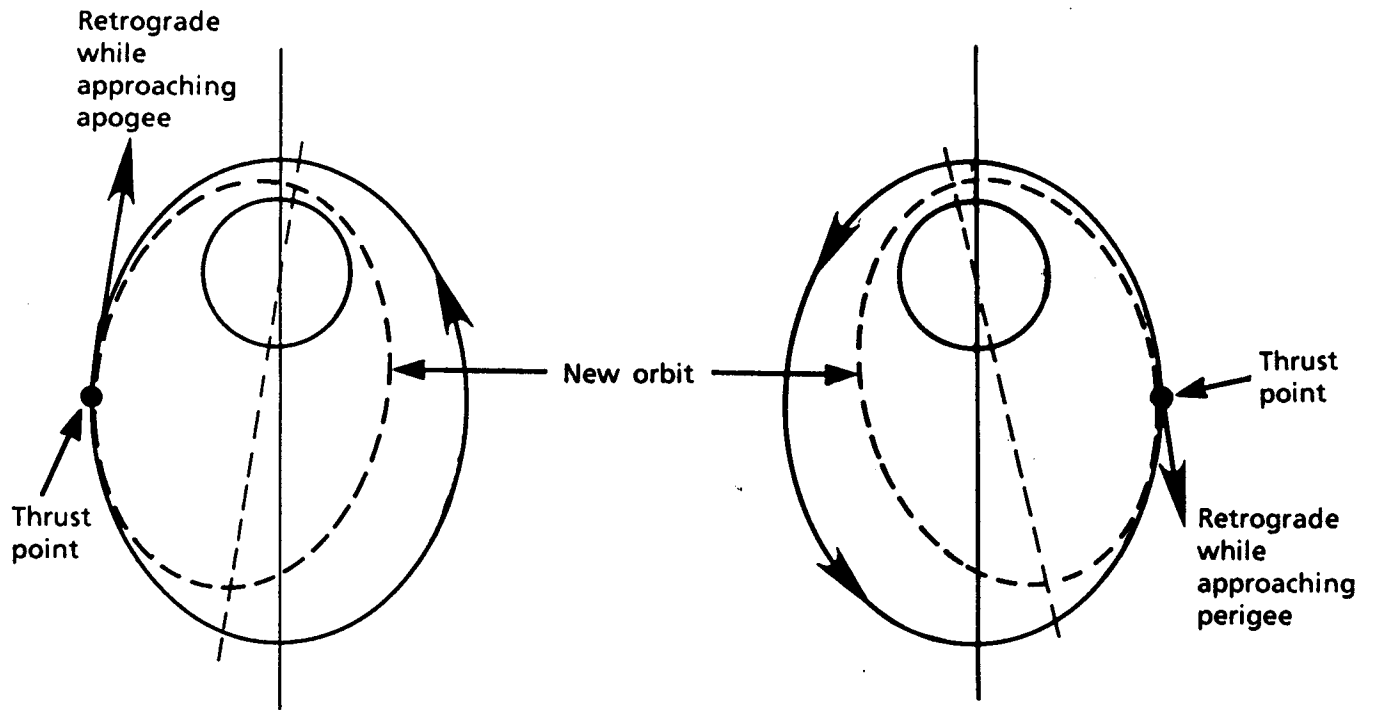


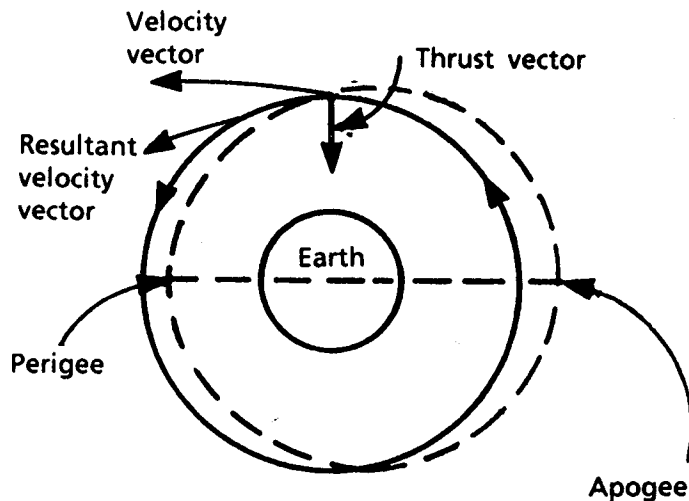
Figure 1-15.—Retrograde burn at point other than apogee or perigee.

Radial Burns

A radial burn is one in which the thrust is applied in a direction perpendicular to the spacecraft's velocity vector and in the orbital plane of the spacecraft. The name *radial* comes from the fact that if the spacecraft is in a circular orbit, a radial burn would be applied along the spacecraft's radius vector from the center of the Earth. Typically, the magnitude of a radial burn is significantly smaller than the spacecraft's orbital velocity (1 ft/s versus a velocity on the order of 25 000 ft/s). Furthermore, it is applied normal to the spacecraft's velocity vector and so does not significantly increase the velocity. Thus, the period of the new orbit is almost identical to that of the original orbit. In fact, for the timespan involved in a rendezvous (only a few days), a radial burn does not significantly

increase the period of the spacecraft's orbit. However, it does alter the line of apsides of the orbit, and it does affect apogee and perigee heights.

To simplify this discussion, assume the spacecraft is in a circular orbit. A burn applied radially inward moves the spacecraft toward the Earth, and this results in a negative flightpath angle. This means that the spacecraft is no longer in a circular orbit and, in fact, is approaching perigee. Perigee will be reached after approximately 90° of orbit travel. Furthermore, as the spacecraft is now approaching perigee, its speed will begin to increase. In the example in figure 1-16, a 1 ft/s radially inward burn was applied to a spacecraft in a 150-n. mi. orbit. This raised apogee approximately 900 feet and lowered perigee approximately the same amount.

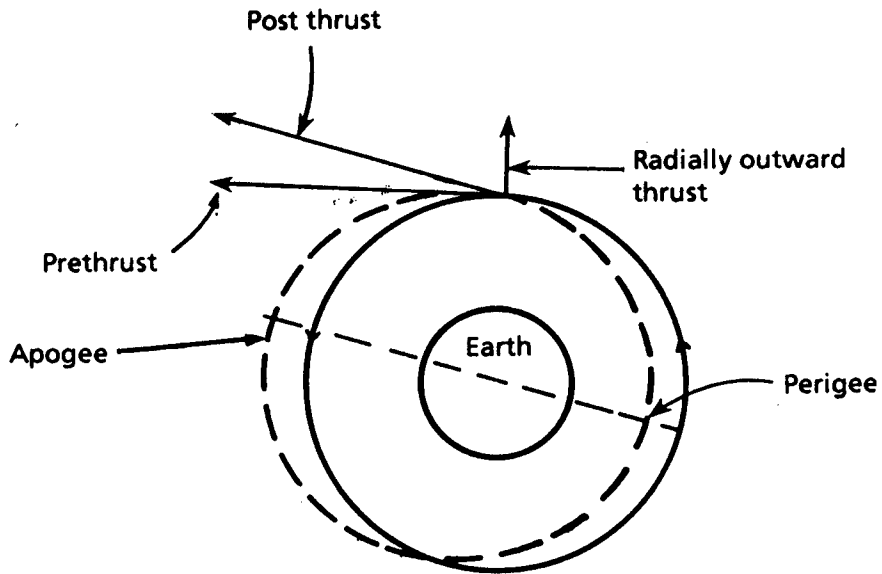


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Figure 1-16.- Radially inward burn in circular orbit.

Conversely, a thrust applied radially outward to a spacecraft in a circular orbit results in the spacecraft moving from the Earth with a positive flightpath angle. Thus, the spacecraft begins to approach apogee in its new orbit (figure 1-17). It will reach apogee approximately 90° of orbit travel ahead of the thrust point. Since the spacecraft is now approaching apogee, its speed will begin to decrease.

In either case, a radial burn applied to a spacecraft in a circular orbit will result in a line of apsides in the new orbit being established approximately 90° away from the thrust point.



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Figure 1-17.— Radially outward burn in a circular orbit.

Out-of-Plane Maneuvers

The first three kinds of maneuvers that have been discussed – posigrade, retrograde, and radial – are all in-plane maneuvers that affect only an orbit's shape and size, not its orbital plane. There are times, such as during rendezvous, when it is necessary to alter the inclination of the orbital plane of a spacecraft. To change the inclination of an orbit, the thrust vector must be directed out-of-plane. To examine the results of an out-of-plane thrust in more detail, first consider an orbit that does not lie in the equatorial plane, i.e., it has a non-zero inclination. The results of an out-of-plane thrust will depend on the direction and magnitude of the thrust, as well as whether it occurred at an ascending node, at a descending node, or off the line of nodes. In all three cases, however, recall that the thrust point is common to both the old and new orbits.

A thrust normal to the orbit plane at either the ascending or descending node will not alter the line of nodes. Instead, it simply rotates the orbit plane about the line of nodes, i.e., changes the inclination. Whether it increases or decreases the orbital inclination depends on whether the node is an ascending or descending node and the direction of the thrust, either on a northerly or southerly direction (see figures 1-18 and 1-19).

Logically, a northerly out-of-plane thrust at the ascending node would increase the inclination, while a southerly thrust would decrease it. The converse would be true at the descending node.

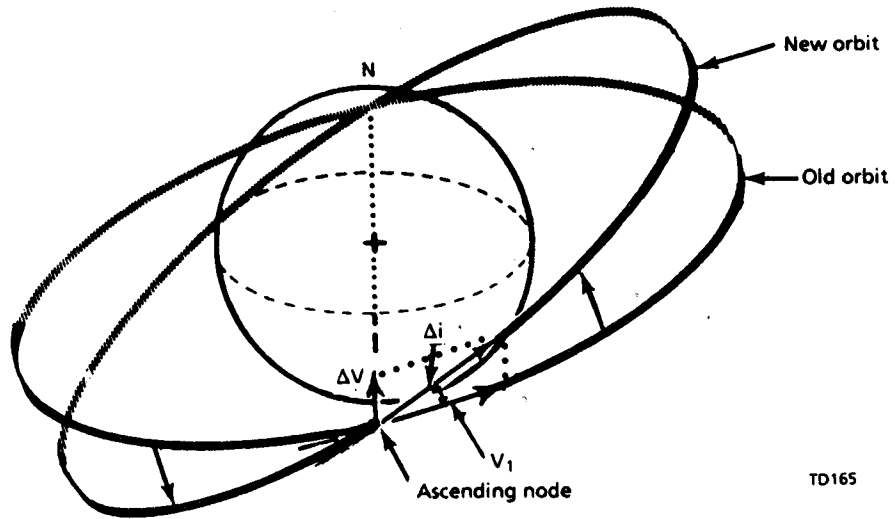
To give you a better idea of the magnitudes involved in plane changes, assume that the velocity of the spacecraft at the ascending node is V_1 and the inclination of the orbit is to be changed Δi degrees by burning out-of-plane at the ascending node (figure 1-18). The minimum ΔV required to perform this plane change would be $|V_1| \sin \Delta i$. Thus, if $|V_1| = 25\,000$ ft/s and $\Delta i = 1^\circ$, the minimum required ΔV would be 436.25 ft/s. As another example, if $|V_1| = 25\,000$ ft/s and $\Delta i = 5^\circ$, the necessary ΔV would be at least 2179 ft/s. As the total quantity of RCS and OMS propellant available for on-orbit operations yields less than 800 ft/s, it is obvious that only very small plane changes will be made on-orbit.

A thrust normal to the orbit plane at a point off the line of nodes will alter the line of

nodes and will change the orbital inclination. Essentially, it rotates the orbital plane about the line formed by the intersection of the prethrust and postthrust orbital planes. Note that this line will contain the thrust point and the point 180 degrees away from the thrust point in the original orbit (see figure 1-20.)

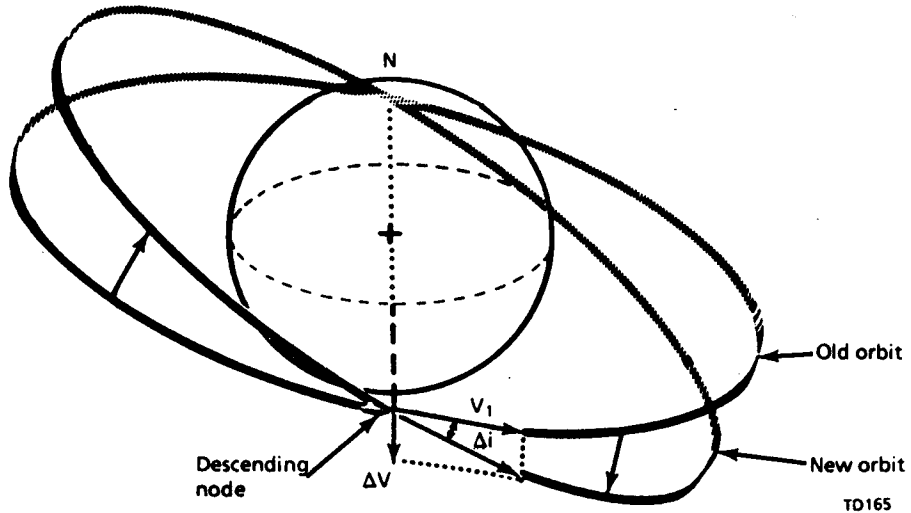
A general rule for the node shift is that when the spacecraft is over the northern hemisphere, a northerly thrust shifts the nodes east; whereas when the spacecraft is over the southern hemisphere, a northerly thrust shifts the nodes west. The opposite is true when evaluating out-of-plane thrusts in a southerly direction. The effect of such a thrust on the orbital inclination varies, depending on whether the thrust occurs generally in the same direction (north or south) that the spacecraft is traveling. Thus, if the spacecraft is traveling from the most southerly point of its orbit to the northern-most point of its orbit, a northerly out-of-plane thrust will increase the orbital inclination, whereas a southerly thrust would decrease it. Note that any out-of-plane thrust at the most southerly or northerly point of the orbit of a spacecraft will increase or decrease its orbital inclination, respectively.

The techniques previously described apply equally well when thrusting to change the orbital plane of a spacecraft to match another target orbital plane. This would be done during rendezvous.



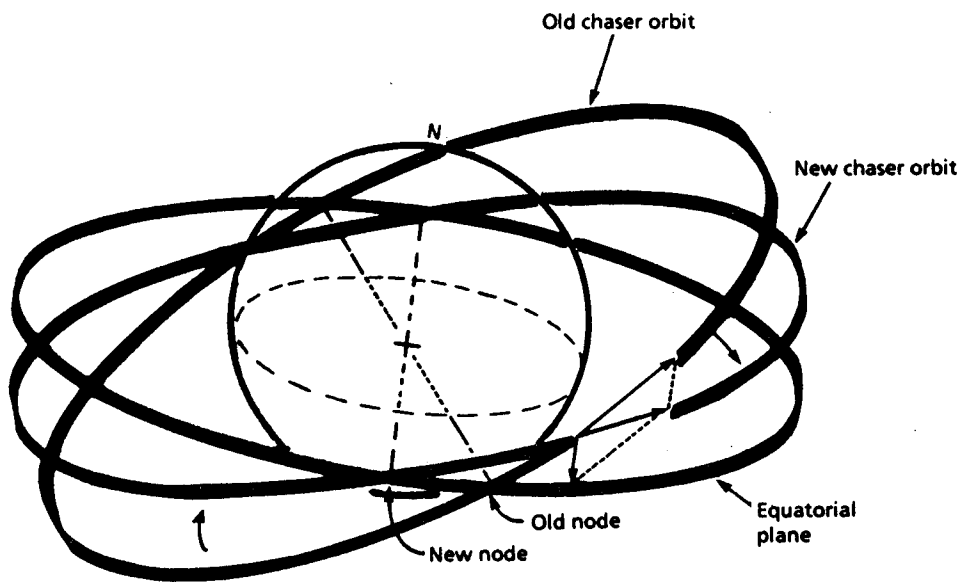
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Figure 1-18.- Northerly thrust at ascending node.



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Figure 1-19.- Southerly thrust at descending node.



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Figure 1-20.- Out-of-plane thrust offline of nodes.

Orbital Transfer

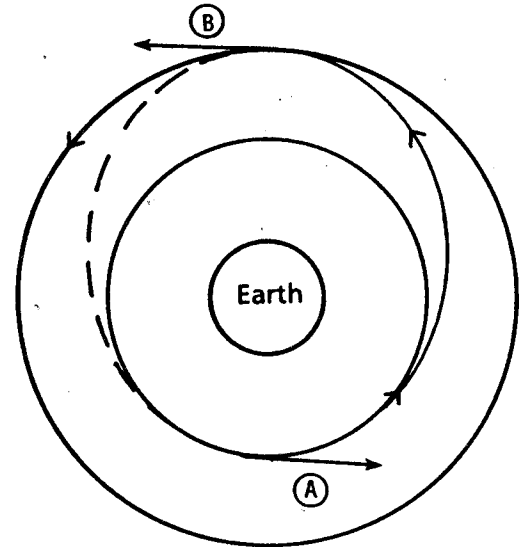
Using the ideas developed in the previous sections, it is possible to examine the problem of transferring between two orbits. Of course, if the two orbits have a point in common, it is possible to change orbits by making the appropriate thrust at the common point. If the two orbits do not have a point in common, two separate thrusts will be necessary: one to establish a transfer orbit which intersects the target orbit, and one to null the relative velocities at the intersection. Recall that an orbit is determined by the velocity (both direction and magnitude) at any given point. Also, recall that the thrust point is common to the prethrust and postthrust orbits. These two facts make it necessary to use two burns to transfer between two nonintersecting orbits. What techniques will be employed to make the orbit transfer will depend primarily on which is being optimized, fuel or time.

Hohmann Transfers

Since propellant is usually limited in space flight, a sequence requiring a minimum total ΔV is of particular interest. Therefore, the maneuver in which a spacecraft is transferred with minimum ΔV from one circular orbit into another, coplanar and concentric with the first, was studied by Hohmann in 1925. Hohmann proved that the minimum ΔV transfer in such a case is made by making two purely posigrade burns if going from a lower to a higher orbit, or with two purely retrograde burns if going from a higher to a lower orbit, assuming that the ratio of large to small orbit radii is less than 11.8 (as would always be the case for Shuttle flights). This maneuver is called a Hohmann transfer.

Suppose a spacecraft is in a circular orbit (A) and needs to go to higher circular orbit (B), using a Hohmann transfer. A posigrade burn can be applied at any point in the original orbit, as long as it is of the appropriate ΔV such that the resultant transfer orbit has its apogee as a point of orbit B 180° away from the thrust point. A second posigrade burn will be made 180° of orbit travel (half a period) later at the apogee of the transfer orbit, which will circularize the orbit to match orbit B (figure 1-21).

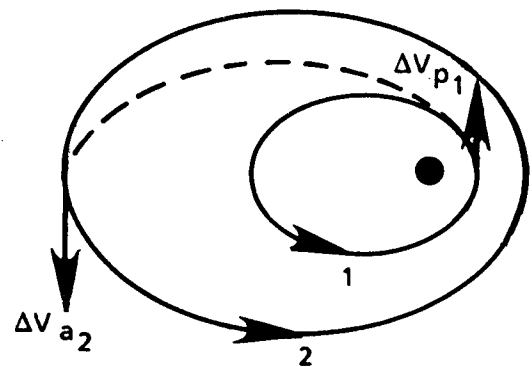
To transfer from a higher to a lower circular orbit, a similar procedure would be followed, with the exception that retrograde thrusts would be used. In any case, a Hohmann transfer requires one-half of the transfer orbit's period to accomplish. Thus, if the lower orbit is at 150 n. mi., the transfer would require at least 45 minutes to complete. For this reason, a Hohmann transfer is considered to be slow.



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Figure 1-21.— Hohmann transfer.

The principle of the Hohmann transfer can be applied equally well for a transfer between two coplanar coaxial orbits. (The two orbits share a common line of apsides.) The minimum energy, and thus minimum fuel, transfer between a lower and a higher orbit also involves two posigrade burns (as shown in figure 1-22) and conversely, the minimum energy transfer from a higher to a lower orbit requires two retrograde burns.



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Figure 1-22.— Transfer between coplanar coaxial elliptic orbits.

High Energy Transfers – Lambert/Clohessy-Wiltshire Targeting

As previously discussed, a Hohmann transfer should be done when fuel conservation is of more importance than time conservation. Also, it is somewhat limited in application as it is used to transfer between two coplanar orbits with a common line of apsides. It is, of course, possible to transfer from any orbit to another orbit with two correctly applied thrusts. It is also, in certain cases, possible to accomplish such a transfer in a specified period of time. In general, the necessary thrusts will not be purely posigrade or retrograde, and thus, such a transfer will be less fuel efficient than a Hohmann transfer. These non-Hohmann transfers are referred to as high-energy transfers.

The basic scheme behind any two-impulse orbital transfer technique is to (1) intercept the target orbit, and (2) match velocities at the intercept point.

In a Hohmann transfer, the intercept point must be at the target orbit's apogee or perigee, and the velocities are matched using posigrade or retrograde burns. In other kinds of orbital transfers (high energy), more latitude is allowed in both targeting steps (see figure 1-23)

As time considerations are of major importance in rendezvous, the most common targeting problem is twofold:

- A. Find the thrust (ΔV) necessary to transfer a spacecraft from a specified position (in its orbit) to another position (in the target orbit) in a specified amount of time (ΔT).
- B. Establish the necessary velocity at the intercept point to enter the target orbit.

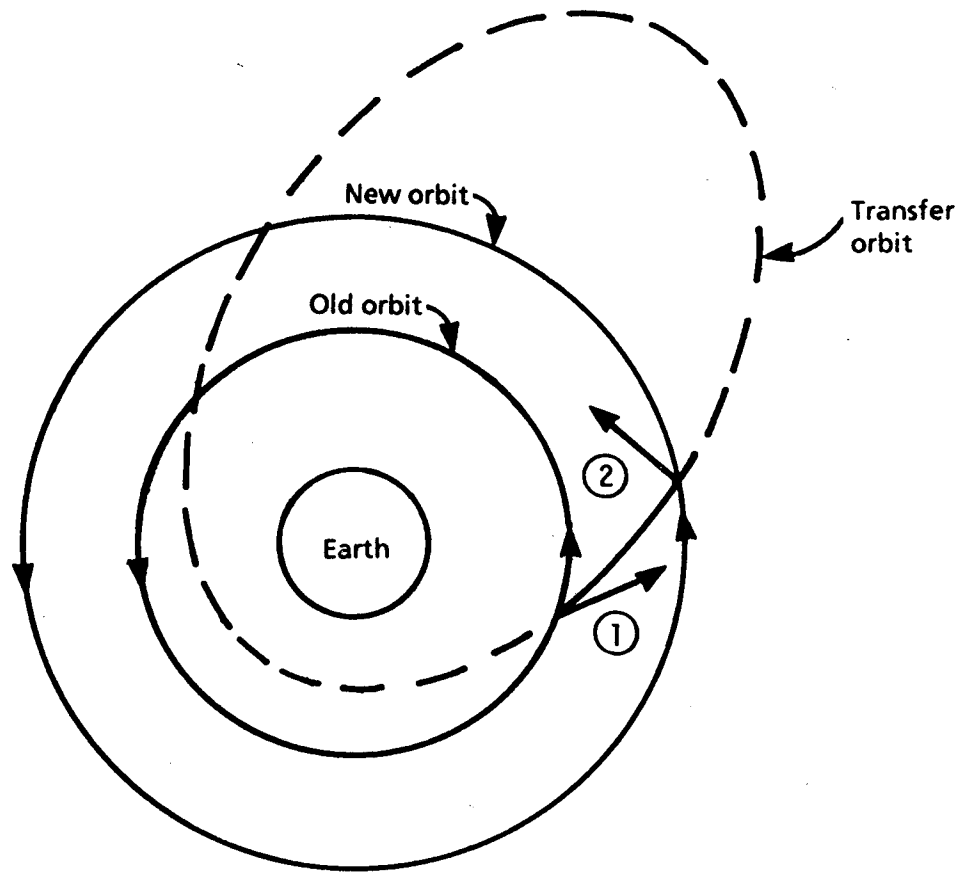
This problem may or may not have a practical solution. The two currently used methods (in the Shuttle onboard software) to solve this problem are Lambert targeting and Clohessy-Wiltshire (C-W) targeting.

Since no closed-form exact solution exists for this problem, some sort of approximation must be made. In the Lambert solution, a conic

transfer orbit is assumed and an iterative numerical approximation method is employed. The results of this form of targeting are very accurate; however, since it is complex it requires a slightly longer computation time. The onboard computers may require from 15 seconds to 2 minutes to compute the required ΔV . As it is very accurate, this technique will be used for a majority of all rendezvous maneuvers.

The C-W equations involve a simplification of the Lambert equations. Furthermore, they are derived in terms of the relative positions and state of the spacecraft (such as the Orbiter), with respect to a target-centered curvilinear coordinate system, as opposed to the inertial positions of the Orbiter and target, used in Lambert targeting. The C-W equations, therefore, result in a simpler and thus faster solution to the original problem, and, over short distances and timespans, their solution is accurate enough to accomplish the objective.

Which onboard targeting technique will be used will be determined preflight. A flag will be set in the software to specify which of the I-loaded maneuvers will be done with Lambert targeting and which will be done using C-W targeting. During flight, this flag can only be changed via a ground uplink command; or onboard, using the general purpose computer (GPC) memory read/write capability through the GPC MEMORY display (see appendix B for more information).



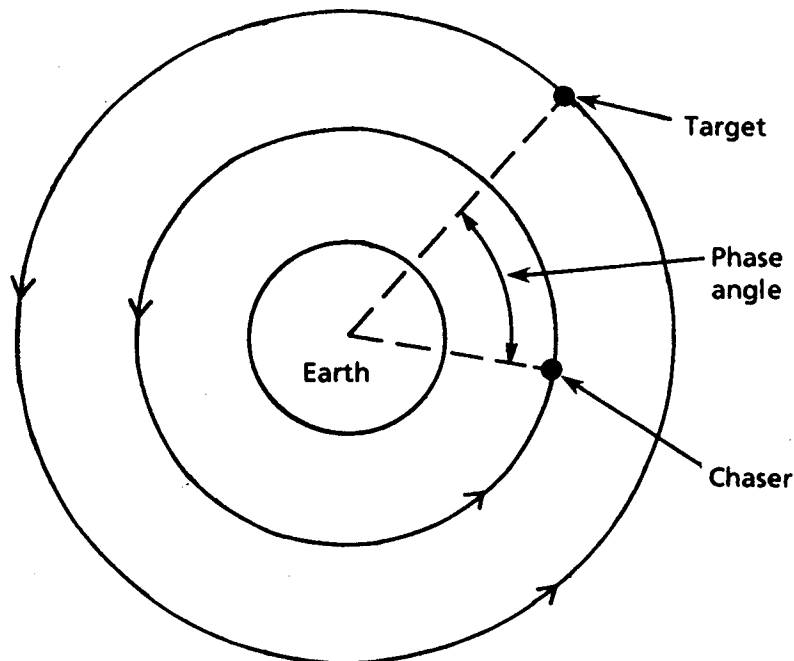
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Figure 1-23.- High energy transfer.

Phasing (Catchup Rates)

In the previous paragraphs concerning orbit transfers, the primary goal was to make the original spacecraft orbit the same size and shape as the target orbit. In rendezvous, however, an additional goal must be accomplished: the target vehicle must be at (or near) the targeted orbit intercept point. In order to accomplish this objective, the phase angle, as depicted in figure 1-24, must be adjusted. The phase angle is the angle between the line joining the chaser spacecraft and the center of the Earth, and the line joining the target vehicle and the center of the Earth.

If the chaser spacecraft and the target vehicle were in the same orbit, and a nonzero phase angle existed between them, the chaser would never catch up with the target without outside intervention. While thrusting directly in the direction of the target vehicle can bring the two vehicles together if the phase angle is small, it will also change the orbit of the chaser vehicle and thus may defeat the purpose.



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Figure 1-24. – Phase angle.

Early in the rendezvous sequence, a technique known as phasing is used. This technique is based on the fact that orbital period (and thus orbital rate) is contingent on the length of the semi-major axis. A smaller orbit has a shorter period and thus a faster average angular rate than a larger orbit.

A chaser vehicle in an orbit which is lower than the target vehicle's orbit will begin to catch up to the target vehicle; i.e., the phase angle will decrease.

Actually, this same technique may be used in reverse also. The chaser can be placed in a higher orbit than the target, and then the target will catch up to the chaser. The exact catchup rate is determined by the size of their orbits: If

a_t = semi-major axis (from the center of the Earth) of the target orbit

a_c = semi-major axis (from the center of the Earth) of the chaser vehicle orbit

d = $a_t - a_c$

Then the arc distance (ΔS) traveled by the chaser with respect to the target vehicle during each target revolution is approximately

$$\frac{\Delta S}{\text{rev}} = 3 \pi d$$

and the catchup rate in terms of radians per rev is

$$\frac{3 \pi d}{a_t} \times \frac{\text{radian}}{\text{rev of target}}$$

or

$$\frac{3 \pi d}{a_c} \times \frac{\text{radian}}{\text{rev of chaser}}$$

These rates can be converted to degrees by multiplying by

$$\frac{180}{\pi} \sim 57.3$$

As an example, suppose a target vehicle is in a 140 x 150-n. mi orbit, and the Orbiter is in a 100 n. mi circular orbit. Then the period of the target is 89.9 minutes, and the chaser period is 88.2 minutes.

$$a_t = 3444 + 140 + 150 \text{ n. mi.} = 2.181 \times 10^7 \text{ ft}$$

$2 (R_e = \text{radius of the Earth} = 3444 \text{ n. mi.})$

$$a_c = 3444 + 100 \text{ n. mi.} = 2.1435 \times 10^7 \text{ ft}$$

(1 n. mi. \sim 6077 ft)

$$d = 45 \text{ n. mi.} = 2.74 \times 10^5 \text{ ft}$$

The arc distance, ΔS , traveled by the chaser with respect to the target during every target revolution (every 89.9 minutes) is approximately

$$\frac{\Delta S}{\text{rev}} = 3 \pi d = 424.1 \text{ n. mi.} = 2.58 \times 10^6 \text{ ft}$$

and the catchup rate is

$$\frac{3 \pi d}{a_t} = 0.11817 \text{ radians/rev of target}$$

$$= 6.77^\circ/\text{rev of target}$$

or

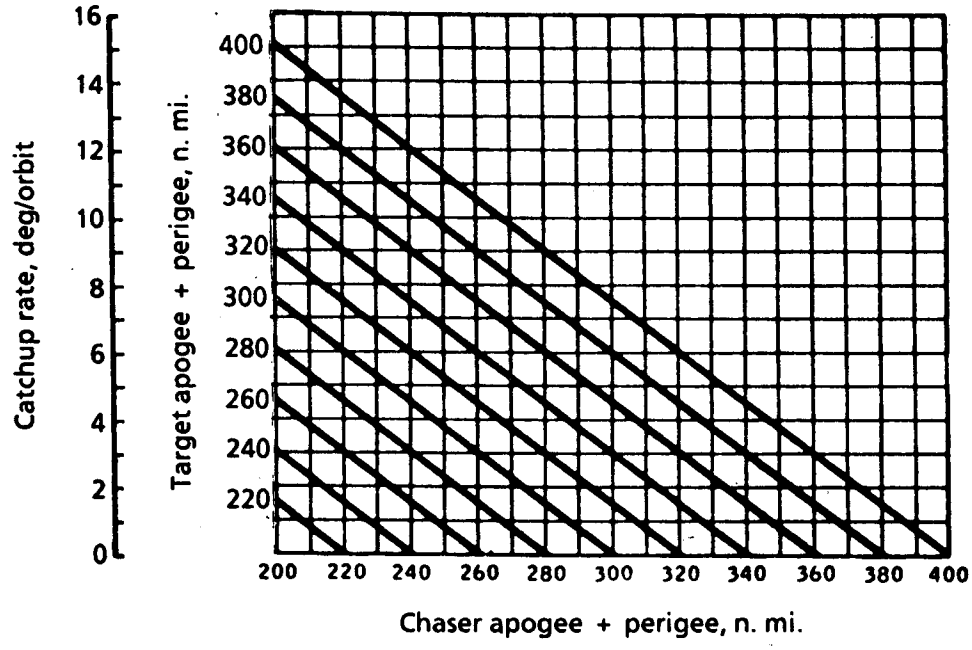
$$\frac{3 \pi d}{a_c} = 0.11967 \text{ radians/rev of chaser}$$

$$= 6.86^\circ/\text{rev of target}$$

As apogee and perigee heights are typically given above the surface of the Earth, the following chart (figure 1-25) may be less complicated than the previous formulas in determining catchup rates for the near-Earth orbits. To use the chart, start with the size of the chaser orbit (in terms of the sum of the apogee and perigee heights above the surface of the Earth). At the bottom of the chart, move straight up until you intersect the diagonal line representing the size of the target orbit (also in terms of the sum of its apogee and perigee and perigee heights above the surface of the Earth).

Then go left and read the catchup rate. As an example, if a target vehicle is in a circular orbit 160 n. mi. above the Earth's surface and the chaser is in a circular orbit at 120 n. mi., their catchup rate is approximately 6 deg/rev.

With these ideas in mind, it is clear that during a rendezvous, it may be necessary to maintain a significant difference in orbit size between the target and chaser orbit in order to establish the proper catchup rate. This technique is called phasing.

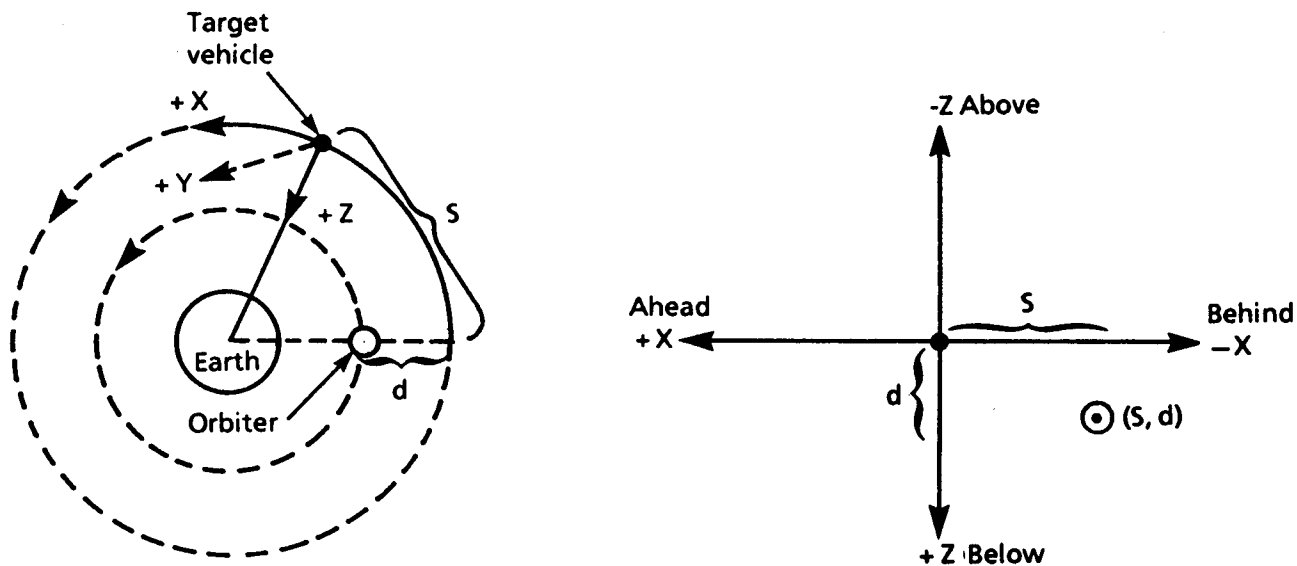


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Figure 1-25.- Catchup rates.

Relative Motion Plots

Of particular importance during rendezvous operations is the concept of the relative motion between a target and the chaser vehicle. A plot of the relative motion of the vehicle can be used to simplify explanations of certain rendezvous and proximity operations maneuvers. A relative motion plot uses a target-centered coordinate system in which the Z-axis rotates with the target and is positive-directed radially toward the Earth, the X-axis is curvilinear and positive in the direction of orbit motion, and the Y-axis is out-of-plane and completes the right-hand coordinate system (figure 1-26).



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Figure 1-26.— Relative motion plot definition.

As the relative motion plot is normally used when the phase angle between the target and chaser is relatively small, the coordinate systems used in a relative motion plot can be thought of as an local vertical/local horizontal (LVLH) coordinate system centered at the target. Thus, if the chaser was directly below the target on the radius vector to the Earth's center ($R\text{-bar}$), it would appear on the $+Z$ axis of the relative motion plot. Similarly, if the chaser was ahead of the target on the target's velocity vector ($V\text{-bar}$), it would appear on the $+X$ axis on the relative motion plot. Some examples of typical relative motion plots are shown in figures 1-27 and 1-28. To simplify matters, in each case, assume the target is in a circular orbit (150 x 150 n. mi.).

Example 1: Figure 1-27 shows the chaser in a lower circular orbit (100 x 100 n. mi.) than the target, starting at time A. Notice that the chaser maintains a constant ΔZ with the target, while the trailing displacement (in the X direction) decreases with time as the chaser catches up with and eventually passes the target.

Example 2: Figure 1-28 shows the chaser in lower elliptical orbit (100 x 125 n. mi.), starting at time A. The peaks and valleys in the relative motion plot coincide with the lower orbit's apogee and perigee, respectively. Notice again that the chaser passes the target in their respective orbits due to its shorter period.

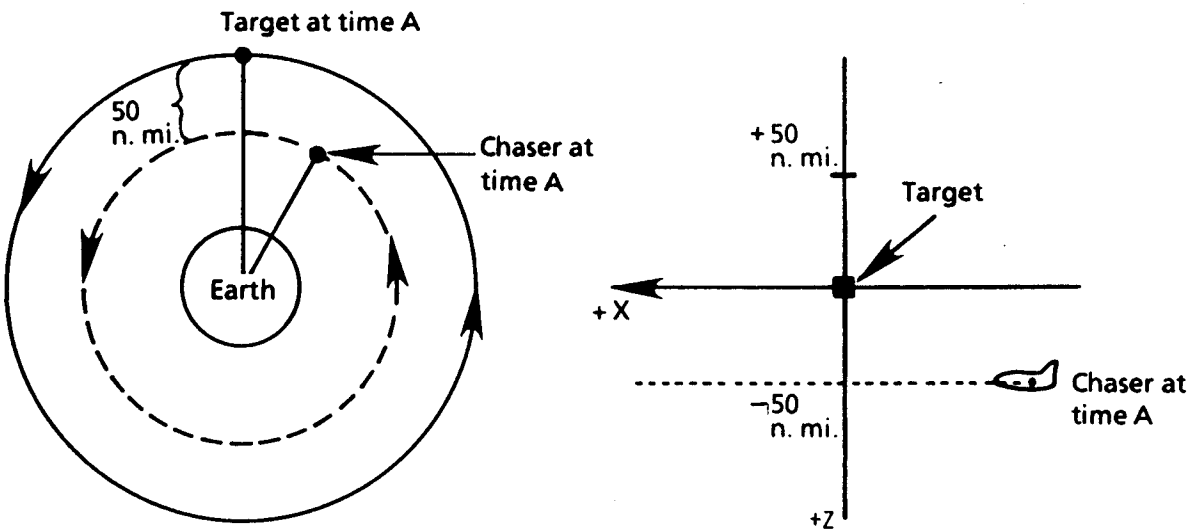


Figure 1-27.- Relative motion plot, example 1.

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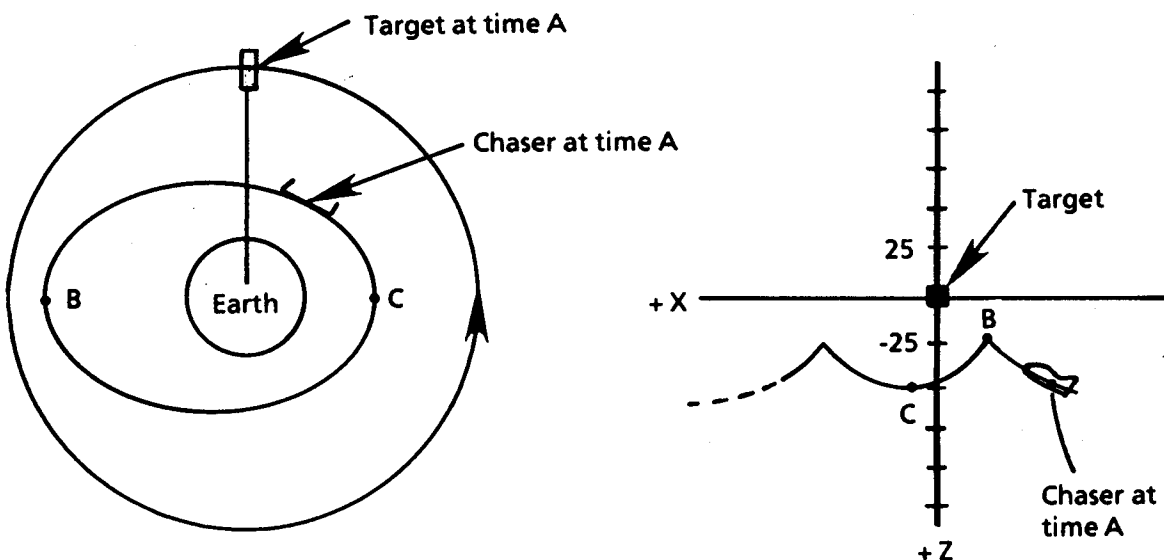


Figure 1-28.- Relative motion plot, example 2.

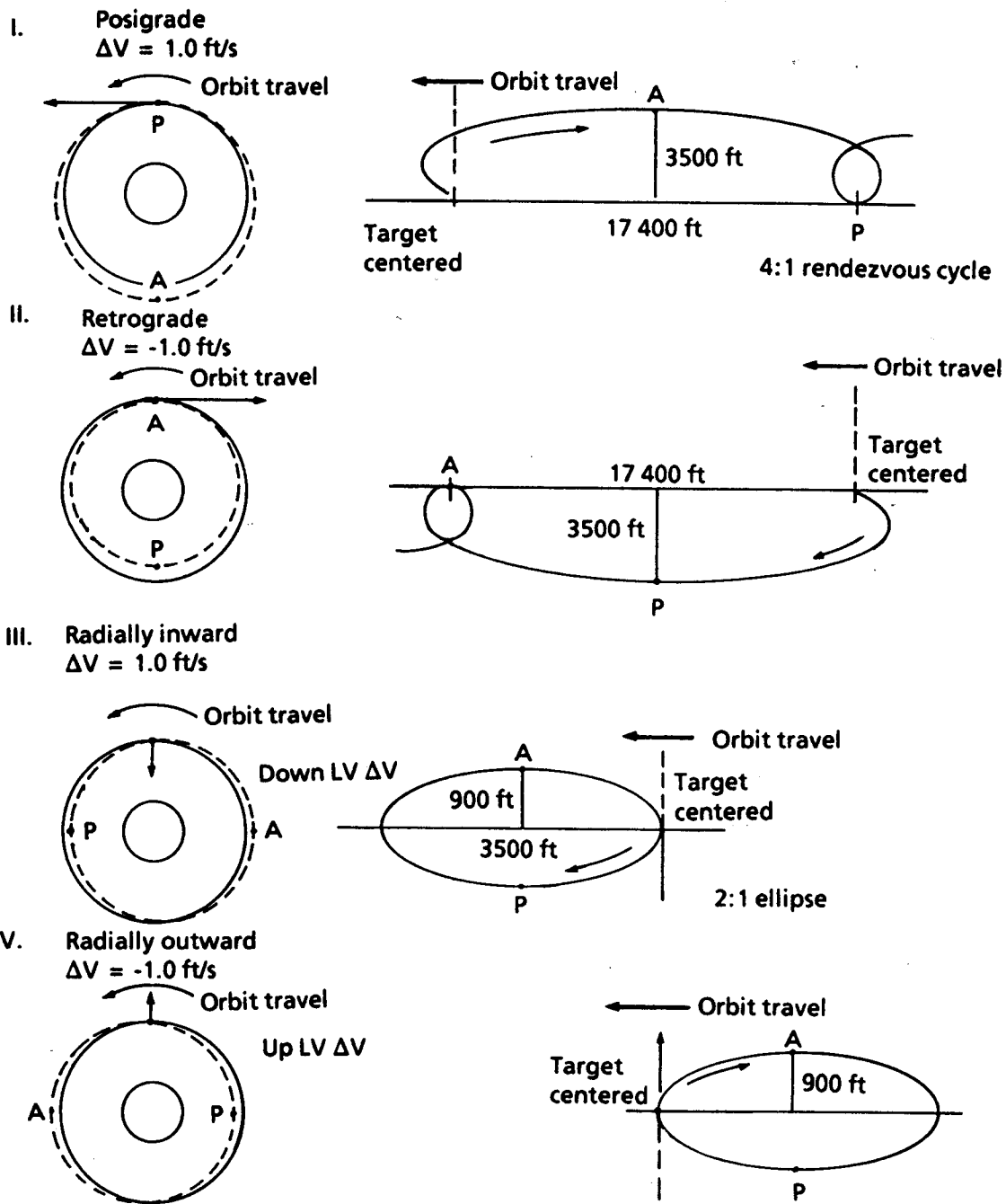
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Relative motion plots are helpful in understanding the effects of the different kinds of in-plane single impulse maneuvers described in the Orbital Maneuvers section. The following examples illustrate the results of posigrade, retrograde, radially inward, and radially outward burns. In each case, the target and chaser vehicles are initially assumed to be together in a circular 150-n. mi. orbit. A burn of 1 ft/s is then made by the chaser vehicle in the appropriate direction. Each maneuver will be described individually and illustrated in figure 1-29.

- I. **Posigrade burn:** A 1-ft/s posigrade burn will place the chaser in a higher orbit, with perigee occurring at the burn point; thus, the chaser will never go below the target. The immediate effect is for the chaser to move ahead of and above the target. However, as it is now approaching apogee and is in a larger orbit (with corresponding longer period) than the target, it will begin to fall back behind the target. This continues until the chaser again approaches perigee, at which time its speed again increases relative to the target, and the chaser loops back toward the target. Thereafter, the chaser will fall back again and continue this type of motion.
- II. **Retrograde burn:** A 1-ft/s retrograde burn forces the chaser to a lower elliptical orbit, with apogee established initially at the burn point. Thus, the chaser will never go above the target. The immediate effect is for the chaser to move behind and below the target. However, since the chaser is in a smaller orbit (with corresponding shorter period) and is approaching perigee, it will soon begin to fall ahead of the target. It continues this motion until it is again approaching apogee. At this time, its speed decreases relative to the target and the chaser loops back toward the target. As it passes apogee, it will again show an increase in relative velocity and will continue this type of motion.
- III. **Radially inward burn:** A radially inward burn of 1 ft/s for the chaser will result in a slightly elliptical orbit, with perigee occurring approximately 90° ahead of the burn. A radial burn of this magnitude is so small compared to the original circular velocity ($\sim 25\,000$ ft/s) that it only slightly

raises the orbit (it alters perigee and apogee ~ 900 feet out of approximately 3600-mile radius), and alters the period (~ 90 minutes at this altitude) a correspondingly small amount. Thus, for all practical purposes, a radial burn does not affect the period of the chaser.

- IV. **Radially outward burn:** A radially outward burn of 1 ft/s for the chaser will result in a slightly elliptical orbit, with apogee occurring 90° ahead of the burn. As described in III, it essentially does not alter the period of the chaser.



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Figure 1-29.— Summary of relative motion plots for 1 ft/s burns made in a circular orbit.

Spacecraft Attitude and Pointing

Up until now, only the position and velocity of a spacecraft with respect to a target vehicle have been discussed. Procedurally, it is also important to be able to specify the attitude orientation of the spacecraft with respect to the target: nose forward, payload bay forward, etc. To do this, a body pointing vector and an angle called omicron are specified. For this discussion, the spacecraft will be the Orbiter.

Body Pointing Vector

A body vector is a unit vector which is defined in a pitch-yaw sequence in the Orbiter's body axis coordinate system. (For a description of the Orbiter's body axis coordinate system, see appendix A.) For example, the Orbiter body vector defined by a 0° pitch followed by a 0° yaw is the + X body axis, whereas the body vector defined by a 90° pitch (up) followed by a 0° yaw is the - Z axis. In certain conditions, it will be desirable to point a specified body vector toward a specified target. For example, it may be necessary to point the Orbiter's -Z axis toward a target (or star) prior to using the -Z star tracker. In this case, the -Z axis would be called the body pointing vector. It is important to remember that the body pointing vector does not have to be one of the Orbiter body axes. It can be an arbitrary axis based on the body axes (figure 1-30). The GPC software necessary to point the Orbiter is called *universal pointing*, and is a part of on-orbit guidance, and will be controlled by the crew via a CRT display: UNIV PTG (Universal Pointing).

Omicron

Omicron (OM) is an angle used to specify the orientation of the Orbiter about the body vector which is pointing at the target. It is used with the tracking option in the UNIV PTG display.

Omicron Definition

Omicron is the angle between two particular planes. Except when the body pointing vector (BPV) is the + or -Y axis, the

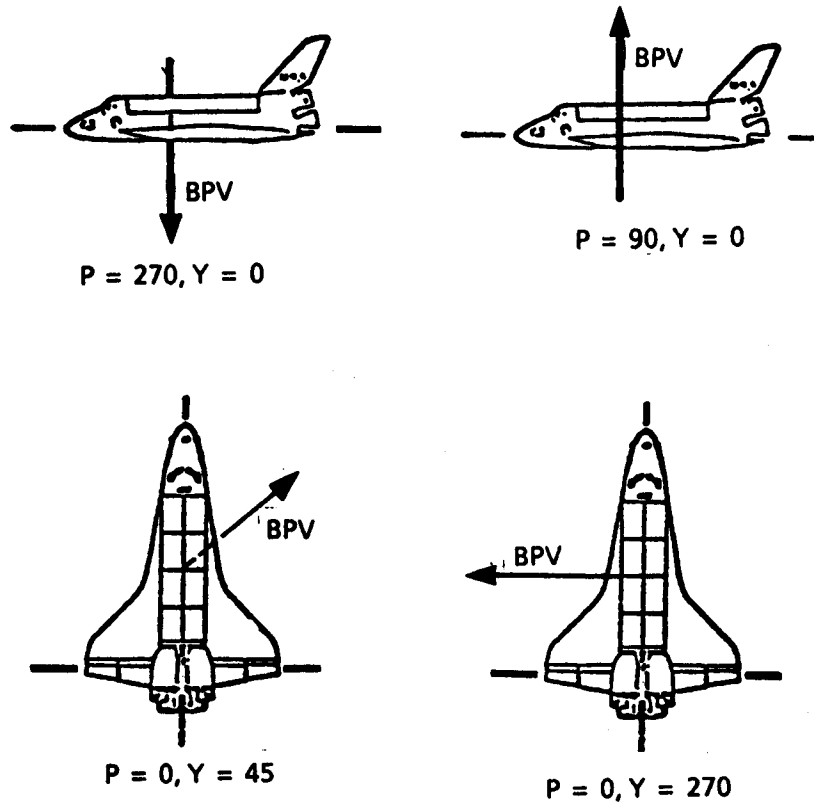
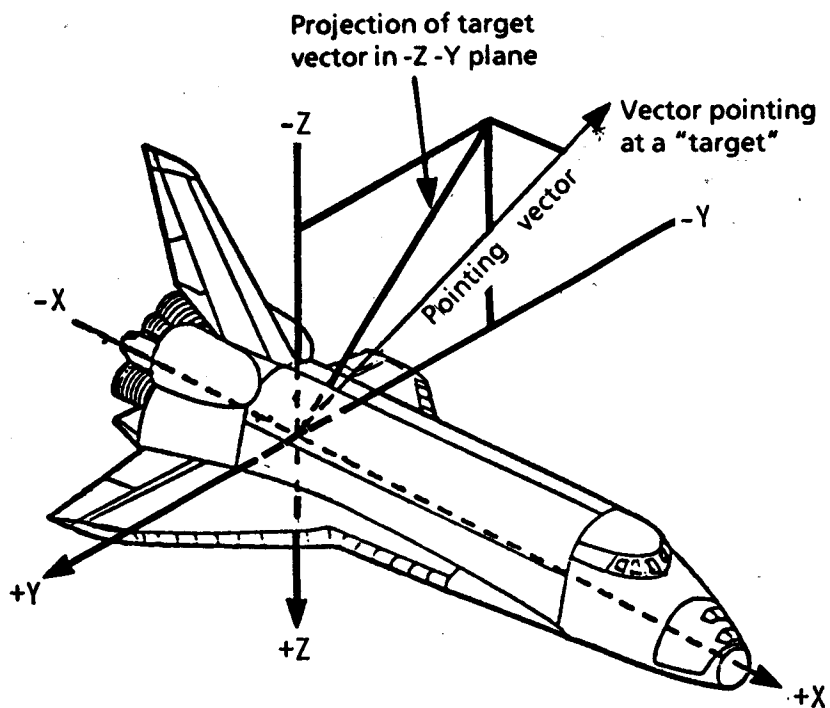
first plane is defined by the BPV and the negative angular momentum vector ($-\vec{h}$), and the second plane is defined by the BPV and the Orbiter's + Y axis. The negative angular momentum vector is identified by first finding the angular momentum vector using the right-hand rule; i.e. wrap your fingers in the direction of orbital travel. The $-\vec{h}$ will be opposite the direction the right-hand thumb is pointing. If the BPV is the + or -Y axis, then the -Z body axis is substituted for the + Y axis in the definition of the second plane; i.e., the second plane is defined by the BPV and the Z-axis of the Orbiter. If the BPV is the negative angular momentum vector, then the descending node vector (the vector from the center of the Earth to the descending node of the Orbiter's orbit) is substituted for $-\vec{h}$ in the definition of the first plane.

Thus, omicron is the angle between the first and second planes, defined above, measured clockwise with respect to the BPV. Note that omicron can be visualized as the angle defined when the first plane is rotated clockwise about the BPV into the second plane.

Another way to define omicron is to say it is the angle between the vector formed by the cross product of the BPV and $-\vec{h}$ and the vector formed by the cross product of the BPV and the Orbiter + Y axis. Once again, substitutions are made in the cases where the BPV = $\pm Y$ or when BPV = $-\vec{h}$. Also, omicron is again measured as positive, resulting in a clockwise rotation from the first vector into the second vector with respect to the BPV.

Neither definition offers quick and ready insight of omicron. A more easily understood description follows:

- Define the BPV
- Determine the Orbiter's orientation with an omicron of 0° . To do so:
 - Point the Orbiter BPV at the target (for example, the center of the Earth)
 - Point the Orbiter's + Y axis in the direction of the negative angular momentum vector ($-\vec{h}$)
- Rotate the omicron = 0° configuration clockwise (looking down the BPV) about the BPV the specified number of degrees.



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Figure 1-30.— Body pointing vectors in a pitch-yaw sequence.

Omicron can now be defined as the angle of rotation about the BPV through a specified number of degrees from a reference attitude (omicron = 0°) to a desired attitude.

There is one important thing to remember: if the BPV is the +Y or the -Y axis, place the Orbiter's -Z axis along $-\bar{h}$.

Detailed Omicron Examples

In the following examples, $-\bar{h}$ is the negative angular momentum vector of the Orbiter. In the figures below, $-\bar{h}$ is perpendicular to the orbit plane (into the paper).

Example 1: Target = long duration exposure facility (LDEF), and the Orbiter is on the target's negative velocity vector, trailing the target.

BPV = +X axis

First, find the omicron = 0° case. To do this, point the Orbiter's +X axis at the target and place the Orbiter's +Y axis along the $-\bar{h}$ vector (figure 1-31).

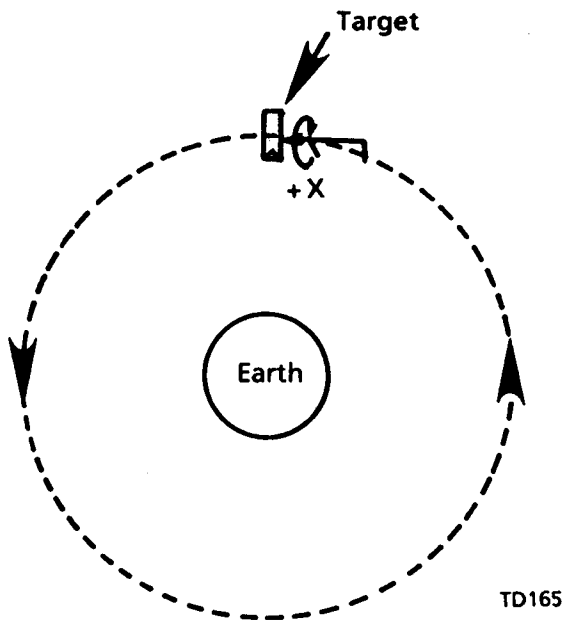


Figure 1-31.- Example 1: OM = 0°.

To find the orientation with an omicron of 180°, rotate the configuration in figure 1-31, 180° clockwise about the BPV (figure 1-32).

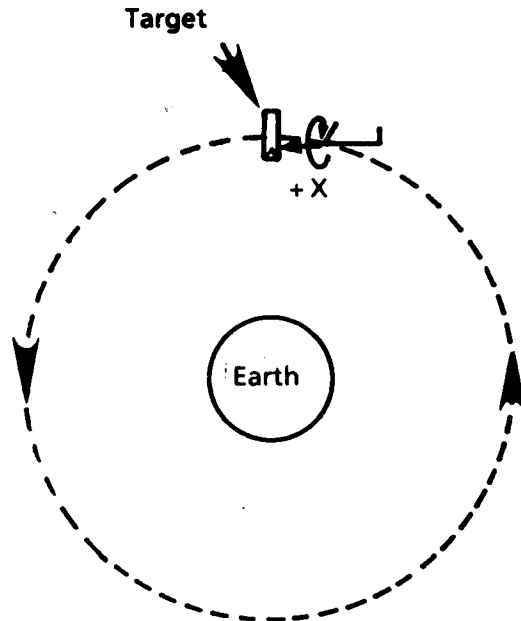


Figure 1-32.- Example 1: OM = 180°.

Example 2:

Target = Earth center

BPV = +Z Orbiter axis
(P = 270°, Y = 0°)

First, find the omicron = 0° case. To do this, point the Orbiter's +Z axis at Earth's center and place the Orbiter +Y axis along the $-\bar{h}$ vector (figure 1-33).

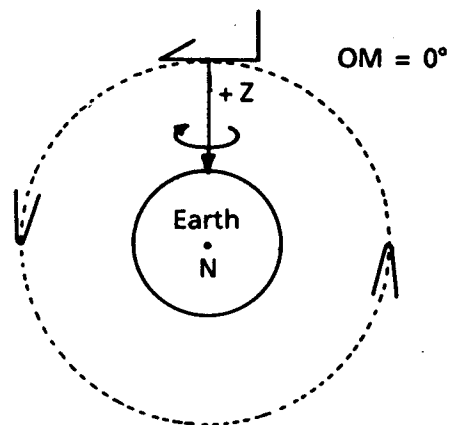
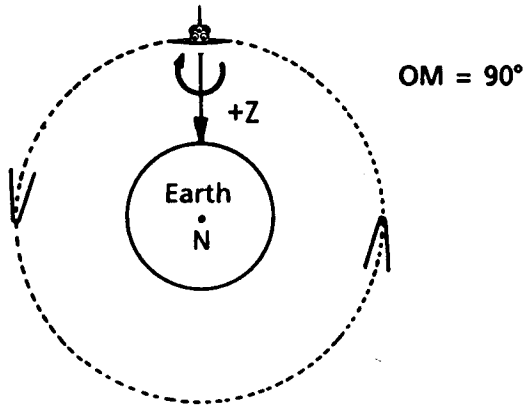


Figure 1-33.- Example 2: OM = 0°.

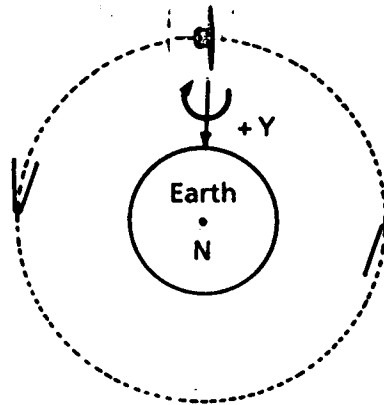
Now, to find the orientation with another value of omicron, such as $OM = 90^\circ$, rotate the above Orbiter configuration 90° clockwise about the BPV (figure 1-34).



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Figure 1-34.— Example 2: $OM = 90^\circ$.

To find $OM = 270^\circ$, rotate the Orbiter about the BPV 270° clockwise (figure 1-36).



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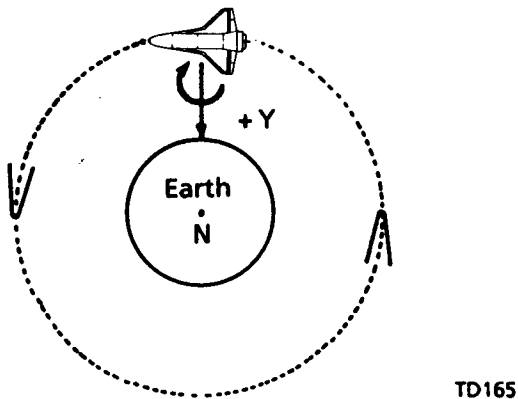
Figure 1-36.— Example 3: $OM = 270^\circ$.

Example 3:

Target = Earth center

BPV = +Y Orbiter axis
($P = 0^\circ, Y = 90^\circ$)

In this case, first find the $OM = 0^\circ$ configuration. Point the Orbiter's +Y axis toward Earth's center and point the Orbiter's -Z axis along the negative angular momentum vector (-h) (figure 1-35).



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Figure 1-35.— Example 3: $OM = 0^\circ$.

Summary example:

Fill in the square in figure 1-37 from the chart in figure 1-38.

Target = Earth center

BPV = body pointing vector, defined in a pitch-yaw sequence.

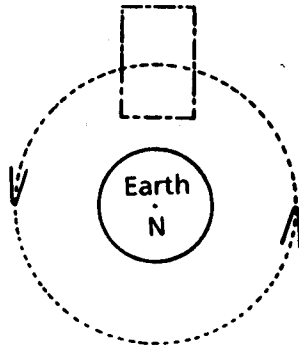


Figure 1-37.- Summary example.

BPV \ OMICRON	OMICRON						
	1	0°	2	90°	3	180°	4
+X P=0, Y=0							
-X P=180, Y=0							
+Z P=270, Y=0			 REAR		 FRONT		
-Z P=90, Y=0			 FRONT		 REAR		
+Y P=0, Y=90			 REAR		 FRONT		
-Y P=0, Y=270			 FRONT		 REAR		

Figure 1-38.- Summary example chart.

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Stationkeeping

Stationkeeping is the technique used to maintain the Orbiter at a desired relative position, attitude, and attitude rate with respect to the center of mass of the target vehicle. In such a stable configuration, time is available to obtain navigation information and to determine and set up the subsequent approach maneuvers. The most common positions at which stationkeeping occurs are on the velocity vector of the target (V-bar stationkeeping) or on the target's radius vector of the target to the center of the Earth (R-bar stationkeeping). Note that V-bar stationkeeping can be accomplished either ahead of (+ V-bar) or behind (-V-bar) the target (figure 1-39).

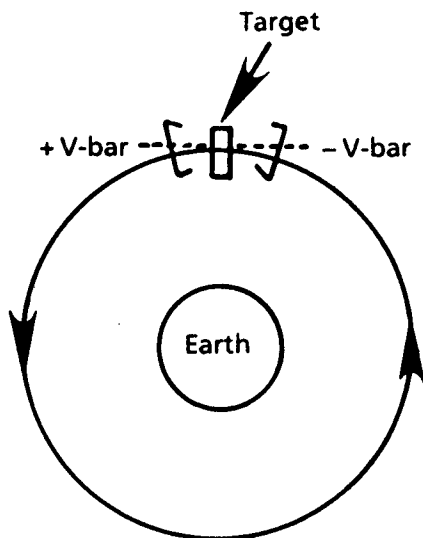


Figure 1-39.- V-bar stationkeeping.

Similarly, R-bar stationkeeping can occur either below (+ R-bar) or above (-R-bar) the target, as shown in figure 1-40. The only completely stable stationkeeping position, however, is on the V-bar of the target. In any other configuration, orbital mechanic effects will result in the Orbiter and target changing their relative positions over an extended period of time. The stationkeeping position will typically be maintained by the crew using the crew optical alignment sight (COAS) and/or rendezvous radar data, although the capability will exist to use onboard software targeting to maintain the stationkeeping position.

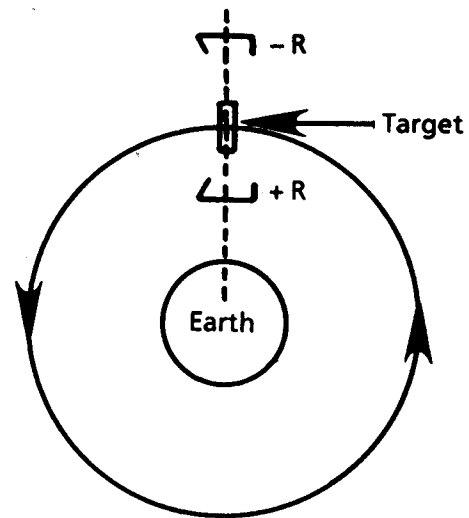


Figure 1-40.- R-bar stationkeeping.

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Lighting Considerations

One of the major driving considerations during the rendezvous phase of a mission involves the correct lighting for navigation and target tracking. While the early rendezvous maneuvers are ground targeted, the later maneuvers will be targeted onboard, using an appropriate navigation sensor. Whether the star tracker or rendezvous radar will be used depends on the status of the target vehicle (active or passive with respect to an active transponder) and the distance to the target. For a passive target, star tracker data will be available at approximately 300 n. mi. if the lighting conditions are correct. Rendezvous radar data, on the other hand, will not be available until 12 to 15 n. mi. The important lighting consideration for the star tracker is the requirement that the target be in the sunlight and not occulted by the Sun, Moon, or Earth's lighted horizon. In order for the crew to monitor the final maneuvers visually, the target must be in the sunlight. Also, the Orbiter must be positioned such that the crew is not being forced to look at (or near) the Sun, the Earth's horizon, or the Moon while looking at the target. All of these considerations will affect rendezvous procedures, as described in the Rendezvous Maneuvers section (section 3).

Launch Window

A launch window is defined as a timespan during which a launch may be made satisfying all mission objectives and constraints. The mission constraints include satisfying the launch and landing lighting conditions and establishing the correct orbit plane. For example, in a rendezvous mission, it is imperative that the Orbiter closely match the orbit plane of the target at launch, as changing the orbit plane after launch is very fuel expensive. Depending on the length and type of rendezvous sequence performed, it may also be necessary to adjust the launch time to provide a reasonable phase angle at orbit insertion.

Contingent on the applicable mission constraints, the launch window may vary from seconds to days. Typically, it will involve a short period of time each day for a certain number of days.

Summary (Section 1)

- Apogee** – The point in a spacecraft's orbit farthest from the Earth.
- Perigee** – The point in a spacecraft's orbit closest to the Earth.
- Apside** – Either apogee or perigee.
- Line of apsides** – The line containing apogee and perigee.
- Local vertical** – The line from an orbiting spacecraft to the center of the Earth.
- Local horizontal** – The line through an orbiting spacecraft perpendicular to local vertical.
- Flightpath angle** – The angle between local horizontal and an orbiting spacecraft's velocity vector.
- Ascending node** – The node formed as the spacecraft travels from south to north across the equatorial plane.
- Descending node** – The node formed as the spacecraft travels from north to south across the equatorial plane.
- Node** – The intersection point between a spacecraft's orbit with another plane.
- Line of nodes** – The line containing the ascending and descending nodes.
- Wedge angle** – The angle between two intersecting orbit planes.
- Semi-major axis** – Line segment between the center of an ellipse and either apogee or perigee.

Orbital inclination

- Angle between the equatorial plane and orbital plane.

Hohmann transfer

- A minimum fuel orbital maneuver that requires two posigrade burns to transfer a spacecraft between two coplanar circular orbits.

Phase angle

- The angle between the line joining the chaser spacecraft and the Earth and the line joining the target vehicle and the Earth.

Launch window

- A timespan during which a launch may be made that will satisfy all mission objectives and constraints.

Launch azimuth

- Angle between due north and the direction in which the spacecraft is traveling.

If an orbiting spacecraft has a positive flightpath angle, it is approaching apogee; if it has a negative flightpath angle, it is approaching perigee.

The maximum velocity of a spacecraft is at perigee; its minimum velocity is at apogee.

A posigrade thrust at apogee or perigee will not shift the line of apsides.

A posigrade burn will raise every point of the original orbit except the thrust point.

A retrograde burn lowers every point of the orbit except the thrust point.

A radial inward thrust applied to a spacecraft in a circular orbit will establish a perigee approximately 90° ahead of the thrust point.

A radial outward thrust applied to a spacecraft in a circular orbit will establish an apogee approximately 90° ahead of the thrust point.

A thrust applied normal to the orbit plane at either the ascending or descending node will not alter the line of nodes and will change the orbital inclination.

The rendezvous maneuvers will be Lambert targeted.

A body pointing vector is defined with respect to the Orbiter's body-axis coordinate system via a pitch/yaw sequence.

The only completely stable stationkeeping position is on the target's V-bar.

Practice Questions

Given the following information, work the problems. (Refer to the text as necessary.)

$$\mu = 1.408 \times 10^{16} \text{ ft}^3/\text{s}^2$$

D_e = diameter of the Earth = 6888 n. mi.

$$1 \text{ n. mi.} = 6077 \text{ ft}$$

1. (a) What is the period of a spacecraft in a 100 x 150 n. mi. orbit?

- (b) In this orbit, what is the velocity of the spacecraft at apogee?

- (c) At perigee?

2. (a) What is the period of a spacecraft in a 150-n. mi. circular orbit?

- (b) What is its velocity?

3. What is the velocity of a spacecraft in a 200-n. mi. circular orbit?

4. Using the information given in the preceding section, if a target vehicle is in a 200-n. mi. circular orbit and the chaser is in a 140 x 160-n. mi. elliptical orbit, find the catchup rate in degrees per target revolution between the two orbits.

Answers

Given the following information, work the problems. (Refer to the text as necessary.)

$$\mu = 1.408 \times 10^{16} \text{ ft}^3/\text{s}^2$$

$$D_e = \text{diameter of the Earth} = 6888 \text{ n. mi.}$$

$$1 \text{ n. mi.} = 6077 \text{ ft}$$

1. (a) What is the period of a spacecraft in a 100 x 150 n. mi. orbit?

$$a = \frac{100 + 150 + 6888}{2}$$

$$= 3569 \text{ n. mi.} = 21\,688\,813 \text{ ft}$$

$$\text{Period} = t = 2\pi \mu^{-1/2} a^{3/2} = 5\,348.5 \text{ sec} \\ = 89.14 \text{ min}$$

- (b) In this orbit, what is the spacecraft's velocity at apogee?

$$V_a = \mu \left(\frac{2}{r_a} - \frac{1}{a} \right)^{1/2}$$

where r_a = apogee height from center of Earth

$$\text{so } r_a = (150 + 3444)(6077) \text{ ft}$$

$$= 21\,840\,738 \text{ ft}$$

$$\text{Thus, } V_a = 25\,309 \text{ ft/s}$$

- (c) At perigee?

$$V_p = \mu \left(\frac{2}{r_p} - \frac{1}{a} \right)^{1/2}$$

$$\text{where } r_p = (100 + 3444)(6077) \text{ ft}$$

$$= 21\,536\,888 \text{ ft}$$

$$\text{Thus, } V_p = 25\,668 \text{ ft/s}$$

2. (a) What is the period of a spacecraft in a 150-n. mi. circular orbit?

$$t = 5404.8 \text{ sec}$$

$$= 90.08 \text{ min}$$

- (b) What is its velocity?

$$V = \left[\frac{\mu}{r} \right]^{1/2}$$

$$\text{where } r = (150 + 3444)(6077) \text{ ft}$$

$$= 21\,840\,738 \text{ ft}$$

$$\text{Thus, } V = 25\,390.3 \text{ ft/s}$$

3. What is the velocity of a spacecraft in a 200-n. mi. circular orbit?

$$V = 25\,215.5 \text{ ft/s}$$

4. Using the information given in the preceding section, if a target vehicle is in a 200-n. mi. circular orbit and the chaser is in a 140 x 160-n. mi. elliptical orbit, find the catchup rate in degrees per target revolution between the two orbits.

$$\text{Catchup rate} = \frac{3\pi d}{a_t} \times \frac{180}{a_t}$$

$$= 7.41 \text{ deg/rev of target}$$

$$\text{where } d = 50 \text{ n. mi. and } a_t = 3644 \text{ n. mi.}$$

Section 2: Navigation Instruments for Rendezvous/Proximity Operations

Introduction

The navigation instruments used in providing Rendezvous/PROX OPS information to the crew are the star tracker, crew optical alignment sight (COAS), rendezvous radar, and the payload bay closed circuit television (CCTV). (Refer to detailed workbooks on each ITEM.)

Star Tracker

The Orbiter star tracker (STRK) is an image dissector electro-optical tracking device which is used to obtain precise angular measurements of selected stars and sun-illuminated orbiting objects (targets). These measurements are used to determine the Orbiter's attitude in inertial space, and this data is used to align the inertial measurement units (IMUs). Also, the STRK provides angular data from the Orbiter to a target being tracked.

Two STRKs, -Y and -Z, are located next to the IMUs on the Orbiter's navigation base. They are slightly inclined off the vehicle axis for which they are named. The -Y axis STRK line of sight is approximately 10.5° from the Orbiter -Y axis, and the -Z axis STRK line of sight is approximately 3° from the Orbiter -Z axis (see figure 2-1).

The STRKs total field of view is $10^\circ \times 10^\circ$ square. The STRKs can also be commanded to search a $1^\circ \times 1^\circ$ square field within the field of view. The STRKs total error in measurement of star or target angles does not exceed 1 arc minute (1 sigma). Star or target magnitude measurement errors do not exceed an absolute maximum of 0.6 magnitude.

The STRKs will acquire and track sunlit targets of 3.0 or greater magnitude. The target must be no brighter than -7.0 magnitude. Target size must be no greater than 8 arc minutes.

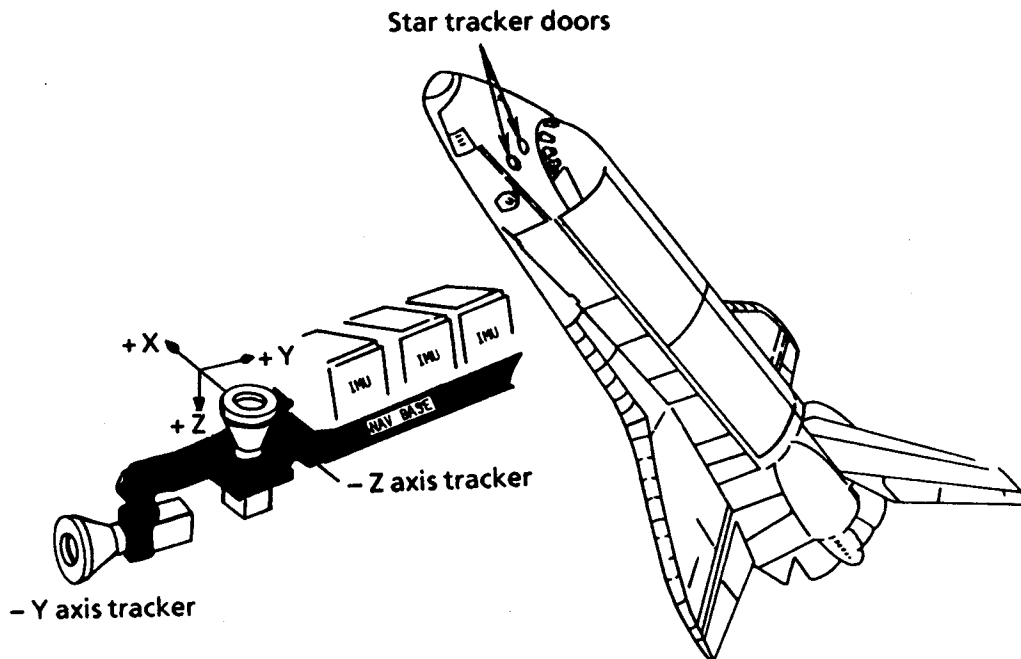


Figure 2-1.-- Star tracker location.

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The STRKs normally will not be used to track a target or star when the Sun is within 30° of the STRK line of sight, the Moon is within 8° of the centerline, the lighted Earth's horizon is within 12° of the centerline, the Earth's dark horizon is within 2° of the centerline, or if the target is in darkness. In any of the above situations, the STRK centerline is said to be occulted (by the Sun, Moon, or Earth).

The primary use of the STRKs is to provide accurate data on star locations to be used in aligning the IMUs. Also of importance in rendezvous is the capability of the STRK to locate a sunlit target in its field of view, provide accurate angle information to navigation, and then to track the target. This is called the target track mode for the ST. For a target the size and magnitude of -7.0, the target may be acquired at up to 300 n. mi. if the lighting is correct.

The STRKs are controlled onboard, using the S TRK/COAS CNTL SPEC (SPEC 22). Before commanding the selected STRK to begin tracking a target, the Orbiter will be positioned so the selected STRK is pointed toward the target (or navigation

estimate of where the target is). This will be done in GNC Major Mode 201 (MM 201), using Universal Pointing (UNIV PTG) in the TRACK mode. (An appropriate target ID and body vector must be entered to the UNIV PTG display, as described in section 5.) Then, the target track (TGT TRK) ITEM can be selected on the STRK/COAS CNTL SPEC (see figure 2-2). At this time, navigation will send the coordinates of the target, based on its calculations. The STRK will go to these coordinates and will search a 1° x 1° square, searching for the target for up to 4 seconds. If the target is not found because of onboard navigation errors or incorrect pointing of the STRK axis, the STRK will begin searching its entire field of view (10° x 10°) for 20 seconds before giving up. If the target is not found, or it is occulted, an appropriate message will be driven (NO TARGET, OUT FOV, OCCULT, etc.). If the target is found, the STRK begins a track scan centered about the position at which the target is found. The track scan is a 12 arc minute cruciform pattern which is executed every 42 milliseconds. The target will continue to be tracked until the target leaves the field of view (FOV), or until it is commanded by the crew to stop tracking via the display.

2011/022/	S TRK/COAS CNTL		2 001/19: 23: 03
			000/00: 00: 00
S TRK CNTL	-Y	-Z	S TABLE
SELF-TEST	1	2	1
STAR TRK	3	4	2
TGT TRK	5	6	3
BREAK TRK	7	8	TRK ID
TERM/ IDLE	9*	10*	33
			Δ MIN
			51
			ANG DIF
			0.0
			ERR
			0.00
			0.00
			0.00
			SEL
			17
			18
			19
			S TABLE CLR
			20
S TRK	-Y	-Z	COAS
RECD ID	11	12	REQD ID
TRK ID	0	0	21
S PRES	0	0	11
ΔANG	+0.32	+0.19	ΔDEG X
THOLD	13	14	-2.7
SHUTTER	CL	CL	Y
MAN OP	15	16	-1.1
STATUS	OUT FOV	OUT FOV	SIGHT MODE
			22
			ACCEPT
			23
			CAL MODE
			24
			DES
			25*
			POS
			+X 26
			-Z 27*
			ΔBIAS
			0.00
			0.00
			UPDATE
			28
			29

Figure 2-2.-- S TRK/COAS CNTL typical display .

4222. ART, 1

Crew Optical Alignment Sight

The crew optical alignment sight (COAS) is an optical device with a reticle projected on a combining glass focused on infinity, which is used to sight stars or a target manually. The COAS is used to align the IMUs when their misalignment is greater than the STRKs can handle, when the STRKs have failed to verify that the correct star or target is being tracked by the -Z tracker, and for improving the relative Orbiter/target state vector during the

rendezvous phase. During PROX OPS the crew may also obtain the range and line-of-sight angle (LOSA) to a target, as well as obtaining a qualitative idea of the range rate and target normal LOS rate.

The COAS may be mounted in two locations defined by the vehicle axis along which it sights, as shown in figure 2-3:

- + X above the commander's window
- -Z next to starboard overhead window.

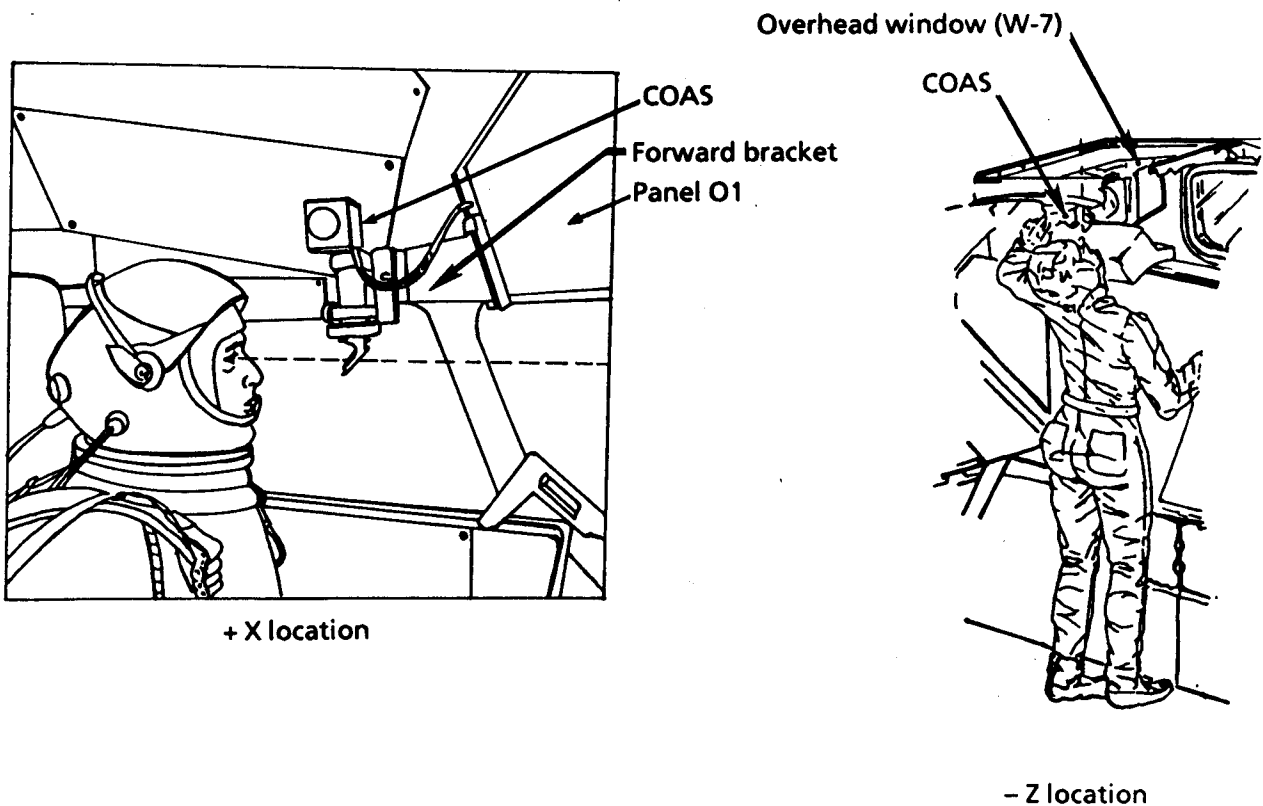


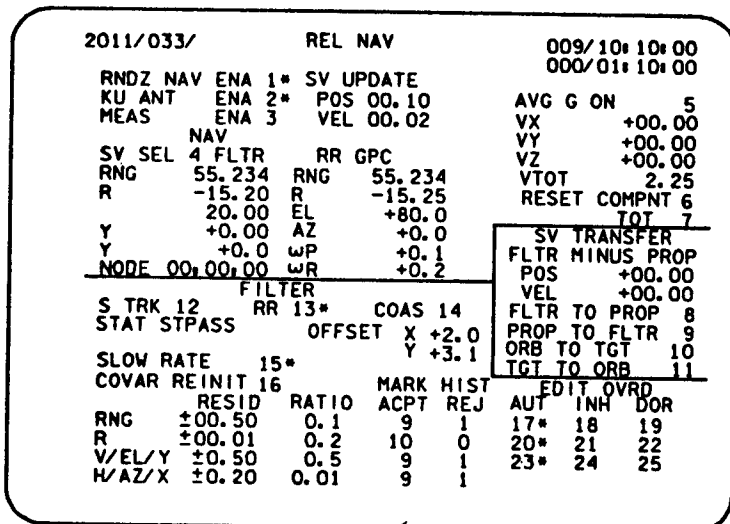
Figure 2.3.- COAS locations.

Before COAS data can be incorporated into rendezvous navigation, several steps must be taken. The COAS must be mounted in either the +X or -Z position. Rendezvous nav must be enabled (ITEM 1) and COAS selected for angle data (ITEM 14). Next, the S TRK/COAS CNTL SPEC (figure 2-2) must be called. Inputs to the display are then made to specify the target REQ ID = 1 (ITEM 21) as well as identifying to software the position in which the COAS is mounted (+X or -Z). An input must also be made to enable the COAS sight mode (ITEM 22). In this mode, the ADI ATT REF pushbutton is used to mark on the target. If the -Z STRK is already tracking an object (target or star) a readout (X, Y) will appear on this display and the REL NAV display (figure 2-4) to inform the crew of the STRK determined location of the object in the COAS FOV. The (X, Y) direction is the same as the Orbiter body axes. In any case, the crew must ensure that the target is in the

COAS FOV. Manual maneuvers are then made so that the target will cross the center of the reticle (see figure 2-4a).

When the target crosses the center, the crewmember depresses the ATT REF button; i.e., takes a mark. This causes the software to store the gimbal angles of the IMUs. However, the software will not process these angles until the data has passed a nav filter test (ratio less than 1 and ITEM 23 enabled) or the mark is forced with ITEM 25. By using ITEM 25, several marks may be taken until the crewmember is satisfied that the target was properly centered at the instant the ATT REF button was depressed.

To be accurate, forced marks must also pass a nav filter test. The nav filter test accepts a mark when its ratio as displayed on SPEC 33 is less than the last marks ratio plus 0.2.



4221. ART, 1

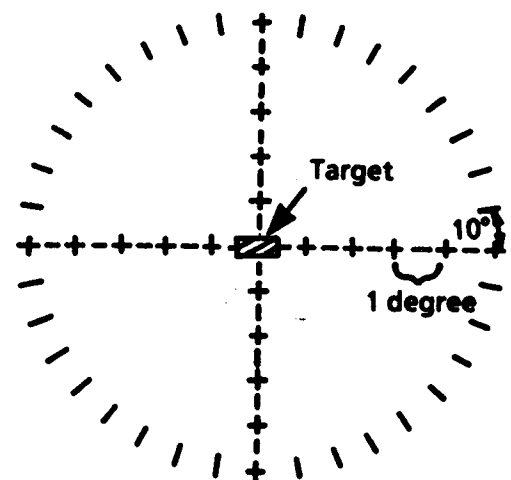


Figure 2.4.- REL NAV display, spec 33.

Figure 2.4(a).- The COAS reticle pattern.

During the PROX OPS mission phase, the pilot can determine the range to the target by centering the target in the reticle and determining its subtended angle (see figures 2-5 and 2-6). An angle versus range chart is then used to determine the relative range. This kind of chart incorporates the known diameter of the target (d) and the subtended angle in the COAS (θ).

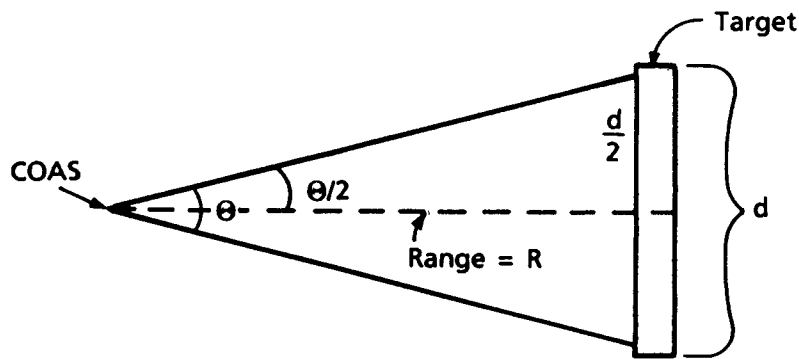
Using trigonometry, the range to the target is given by the equation

$$R = \text{Range} = \frac{d}{2} \cot \frac{\theta}{2}$$

In figure 2-6, $\theta = 4^\circ$, and the diameter, d, of the long duration exposure facility (LDEF) in this orientation is 14 feet. Thus,

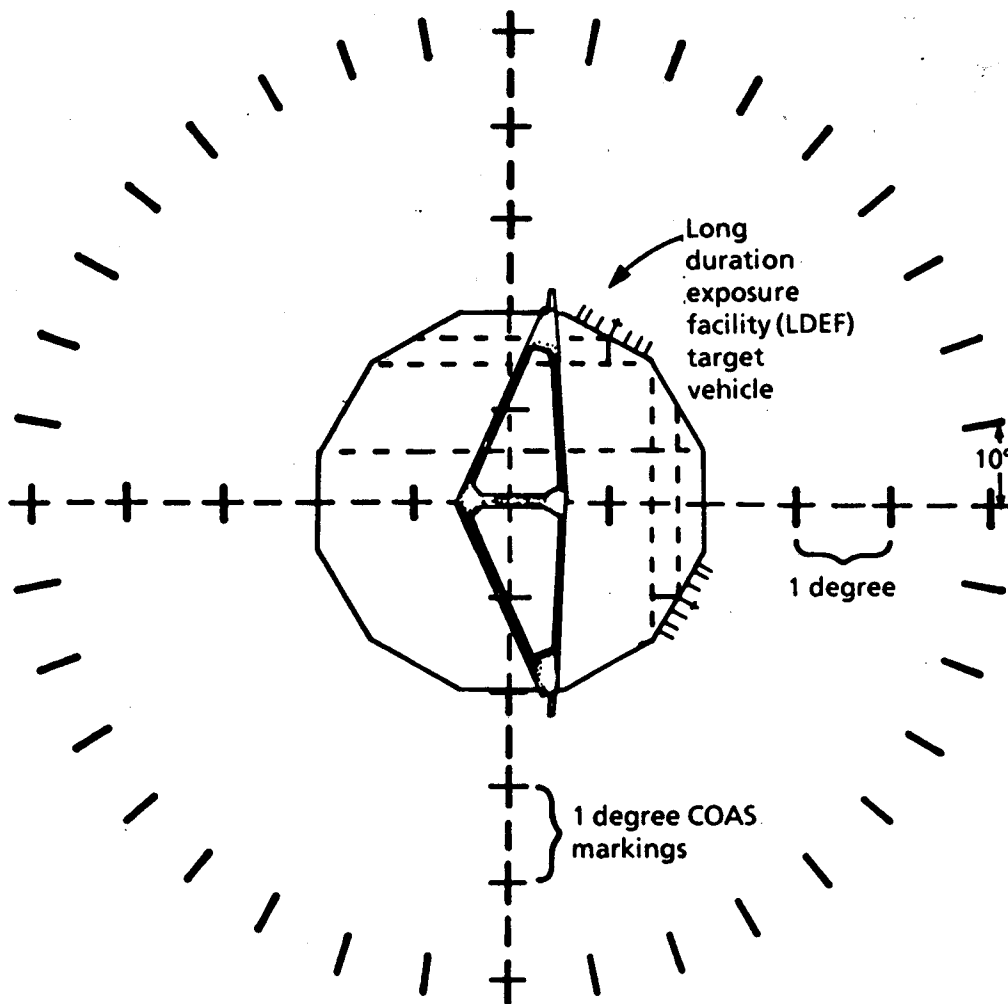
$$R = \frac{14}{2} \cot \frac{4^\circ}{2} = 200 \text{ feet}$$

The range rate of the target can be determined qualitatively (approaching, stationary, departing) by monitoring the change in the subtended angle. If the target is not centered in the COAS, the angular displacement of the target LOS about the X (or -Z) axis can be determined. This angular displacement can be used to determine a qualitative idea of the normal LOS rates by monitoring the movement of the target across the COAS FOV.



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Figure 2-5.- COAS aid in ranging.



d = Target vehicle diameter (ft).

θ_s = Subtended angle in COAS (deg)

$$R = \text{range} = \frac{d}{2} \left[\frac{1}{\tan \frac{\theta_s}{2}} \right] \text{ (ft)}$$

In example shown:
 $d = 14$ ft and $\theta_s = 4^\circ$
 therefore $R = 200$ ft

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Figure 2-6.- COAS ranging technique.

Rendezvous Radar

The rendezvous radar (RR) provides automatic target detection, acquisition, and tracking. It assists in reducing target location uncertainties during rendezvous maneuvers. The radar tracks a target automatically to determine range, range rate, roll (azimuth), and pitch (elevation) relative to the Orbiter, as well as inertial target angle rates.

The RR operates as part of the Ku-band system. It should be noted that the Ku-band system is a dual purpose system which will be used as either a wide band communications system for data interchange with the ground through Tracking and Data Relay satellite (TDRS) or as an RR. The functions of Ku-band communications and radar are not available simultaneously. The antenna associated with this system is contained in a deployable assembly, which is mounted on the starboard sill longeron. It is rotated in a roll-pitch Euler angle sequence (also known as azimuth-elevation, respectively). Because of the design, there are no pure azimuth or elevation movements of the antenna.

Both active and passive targets can be detected by the RR. With an active target; i.e., one with a compatible transponder, the target can be acquired at up to 300 n. mi. A passive target is acquired using skin tracking and depending on its size can be tracked up to 27 n. mi. It is thought, for example, that a target having a 1-square meter radar cross-section can be acquired at 12 to 15 n. mi.

The RR controls are on panel A1U (see figure 2-7) and allow the following operational modes and options:

- POWER – To go from OFF to STANDBY or ON takes about three minutes.
- MODE - RDR COOP – Assumes transponder on target.
MODE-RDR PASSIVE – Assumes skin track only.
MODE-COMM – No radar function.
- RADAR OUTPUT-HIGH, MED, LOW – In the GPC (auto) mode, the radar reduces the power level as range decreases. However, if lock is broken, it automatically returns to the highest power of switch setting. Auto downmoding is required only to protect the antenna and will not necessarily protect the target. To protect the target, the manual downmoding mode must be used.

Therefore, in practice, during an approach it is advised to manually switch from high to medium power and from medium to low power at appropriate ranges (500 and 200 ft have been proposed). This will keep induced currents below 20 volts per meter on the target.

- ROTARY SELECT-GPC – This is "GPC acquisition," a computer driven mode that will automatically point the radar search (for 60 out of every 80 seconds) and track a target. The angle search depends on expected range: $\pm 30^\circ$ at 8 miles, $\pm 20^\circ$ at 12 miles, $\pm 5^\circ$ at 300 miles. Two things determine where the antenna points. If RNDZ NAV and Ku are enabled (ITEM 1&2 on REL NAV SPEC 33) GNC will provide antenna pointing angles to SM for the rendezvous target instead of TDRS. If MIN RDR RNG (ITEM 2 SM OPS 201) is enabled, the antenna will point and search about the Orbiter's -Z axis.

ROTARY SELECT-GPC DESIG – The antenna is automatically pointed at the target. There is no scan. Range/rate data is provided if the state vector estimate is

close enough and a strong enough signal is returned.

ROTARY SELECT-AUTO TRACK – This does a spiral scan about the pointing direction at the time search is initiated, at all ranges, to a width of $\pm 30^\circ$.

ROTARY SELECT-MAN SLEW – The RR must be pointed correctly manually; there is no scan, but all ranges are searched within the 1.8° cone of the radar beam. If a target is acquired, standard range/rate data is provided. To break track on a target, go to MAN SLEW, then slew the RR away a few degrees until the TRACK talkback (tb) goes barberpole (bp).

- SEARCH (momentary contact) – This command only works if the rotary switch is in AUTO TRACK and no target is currently being tracked. The RR then begins a spiral search pattern to a width of $\pm 30^\circ$.
- CONTROL-PANEL – The panel switches have absolute control.

CONTROL-COMMAND – Functions may be altered from the ground via the ground control interface logic (GCIL) despite the current position of the switch in question.

- SLEW – These switches allow manual slewing of the antenna. Two rates are allowed: low (4 deg/sec) and high (20 deg/sec). Elevation and azimuth are defined in an Orbiter body frame along the -Z axis, with elevation corresponding to Orbiter pitch (+ is toward the nose) and azimuth corresponding to Orbiter roll (+ is toward the right).
- SIGNAL STRENGTH meter – This shows how much power is being reflected back and is a function of range, transmitter power, and target cross-section.
- Talkbacks – TRACK is gray if tracking, bp otherwise; SCAN WARN is gray if the beam is within about 5° of the Orbiter body, bp if otherwise; SEARCH is gray if searching, bp otherwise.

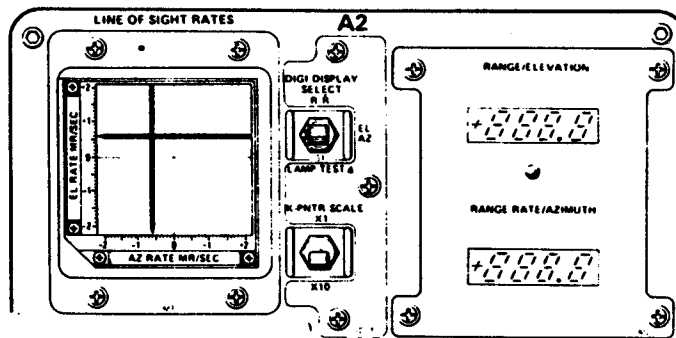
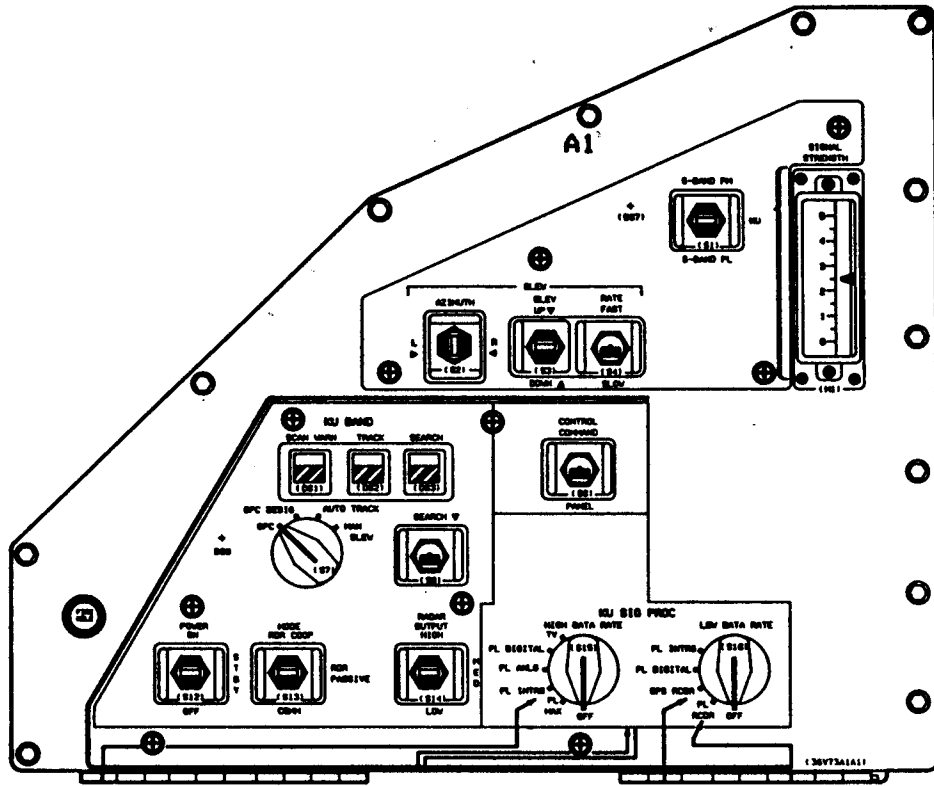


Figure 2-7.- Rendezvous radar displays and controls (panels A1 and A2).

In addition to the REL NAV display (spec 33), RR data is displayed on panel A2 using a pair of LEDs which represent either range (kft) and range rate (fps) or (depending on position of the DIGI-DIS SELECT switch) elevation and azimuth angles. The elevation and azimuth angles are the LOS angles with respect to the Orbiter's - Z axis. The elevation and azimuth inertial LOS target angle rates (in milliradians per second) are always displayed on a two-D meter (scale is selectable, x1 or x10); the "milliradian" unit is useful because multiplying milliradians per second by range (in kft) gives a very close estimate of normal to line of sight (NLOS) rate in feet per second.

The elevation inertial line of sight needle is also very useful in setting up and maintaining a stationkeeping position on the target's velocity vector.

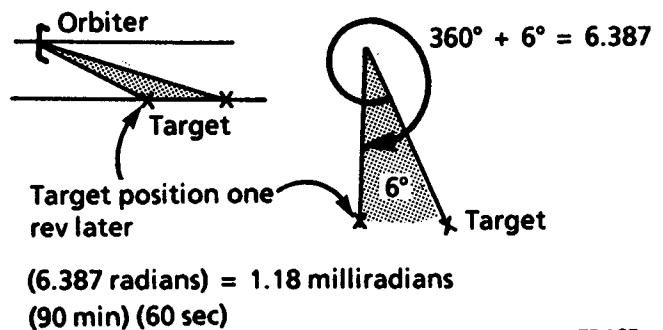
When the Orbiter is in the same orbit as the target, the antenna will sweep out 360° in one orbital period.

$$\frac{2\pi}{\text{Orbiter's orbital period}} (90 \text{ min}) \frac{1 \text{ min}}{60 \text{ sec}} = 1.16 \text{ milliradians/sec}$$

If the Orbiter is in a higher orbit than the target, the elevation rate needle will display a value higher than 1.16. Since the target is in a lower/faster orbit, the antenna has to sweep through an angle greater than 360° in one rev (figure 2-8).

Conversely, if the Orbiter is in a lower orbit than the target, the antenna will sweep out an angle less than 360° in one rev. This results in a value less than 1.16 milliradians/sec.

Sign convention is determined by the antenna's movement through inertial space with respect to the Orbiter's body axes. So, if the Orbiter was trailing the target in the above examples, the sign convention would be negative. We can observe this by keeping the Orbiter's attitude inertially fixed while the antenna tracks a target (figure 2-9).



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Figure 2-8.- Determining inertial LOS rates.

Closed Circuit Television (CCTV)

The crew can use the CCTV to determine range and range rate. The CCTV zoom capability, as well as the conveniently located TV monitor, make the CCTV a desirable substitute for the COAS.

Range Determination

Range can be obtained by reading the tilt angle of the fwd (A) and aft (B) cameras while keeping the target centered in the CCTV monitor.

A chart would be used based on the relationship of range and tilt angle between the two cameras.

Using figure 2-10 as an example:

- Camera A at 70° of tilt
- Camera B at 70° of tilt.

Plotting these two values on the CCTV ranging chart (figure 2-11) places the target at ≈ 80 feet.

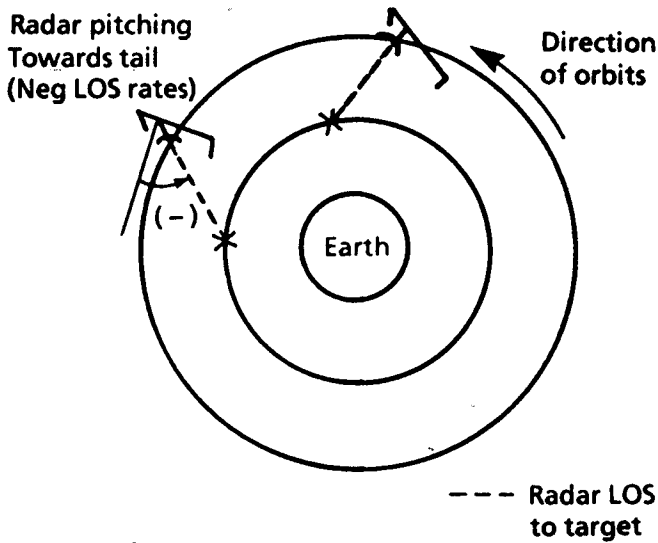
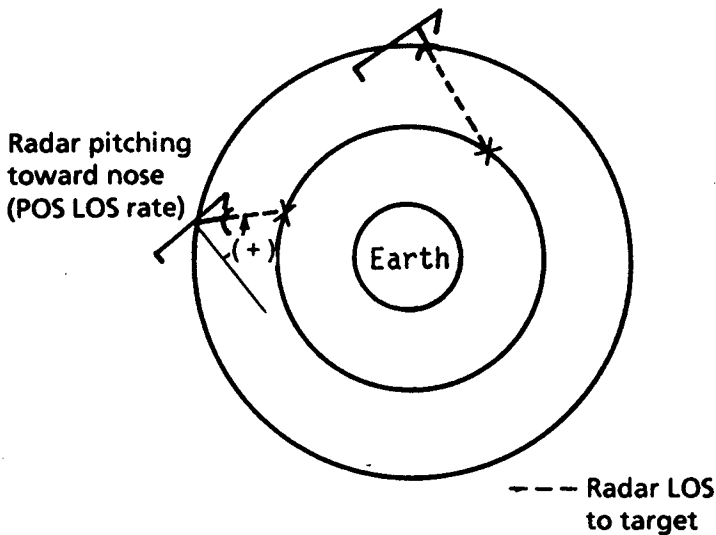


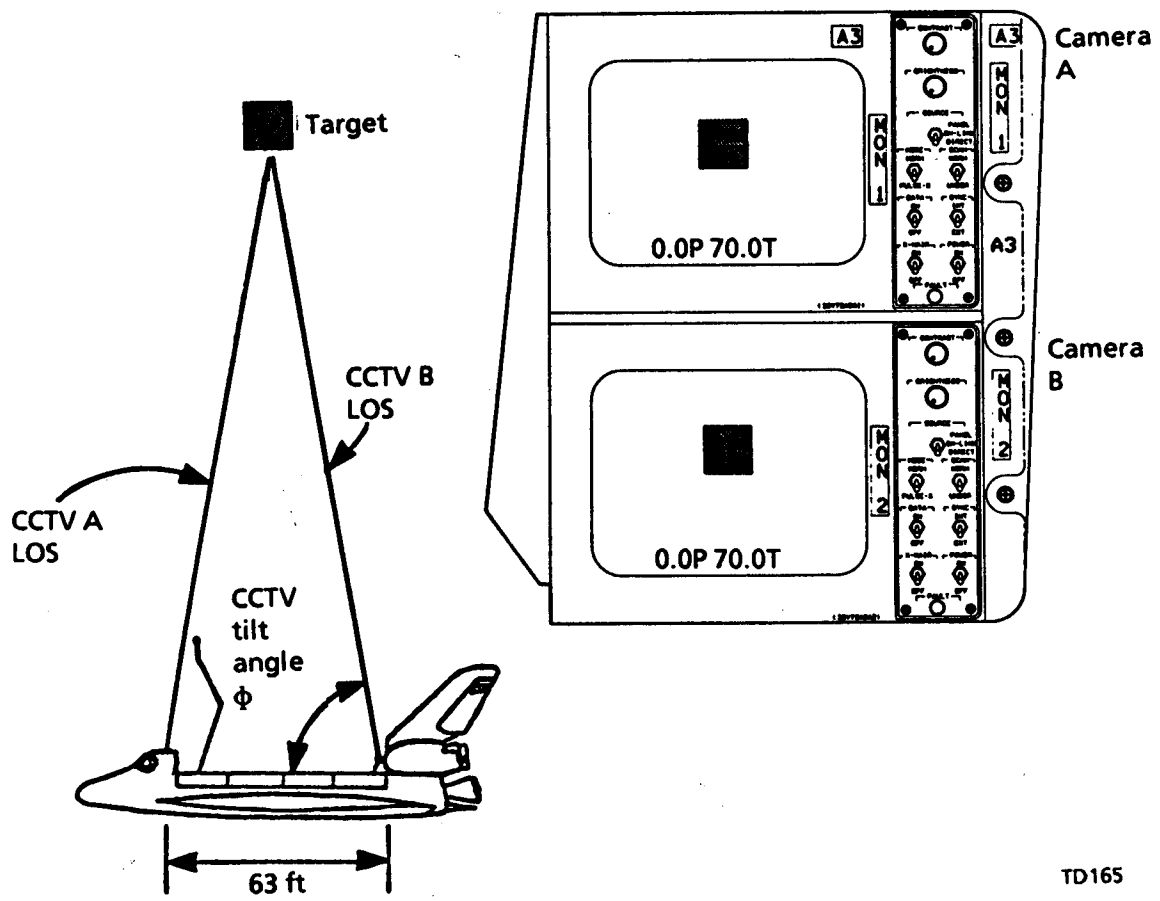
Figure 2-9(a).- Trailing target.



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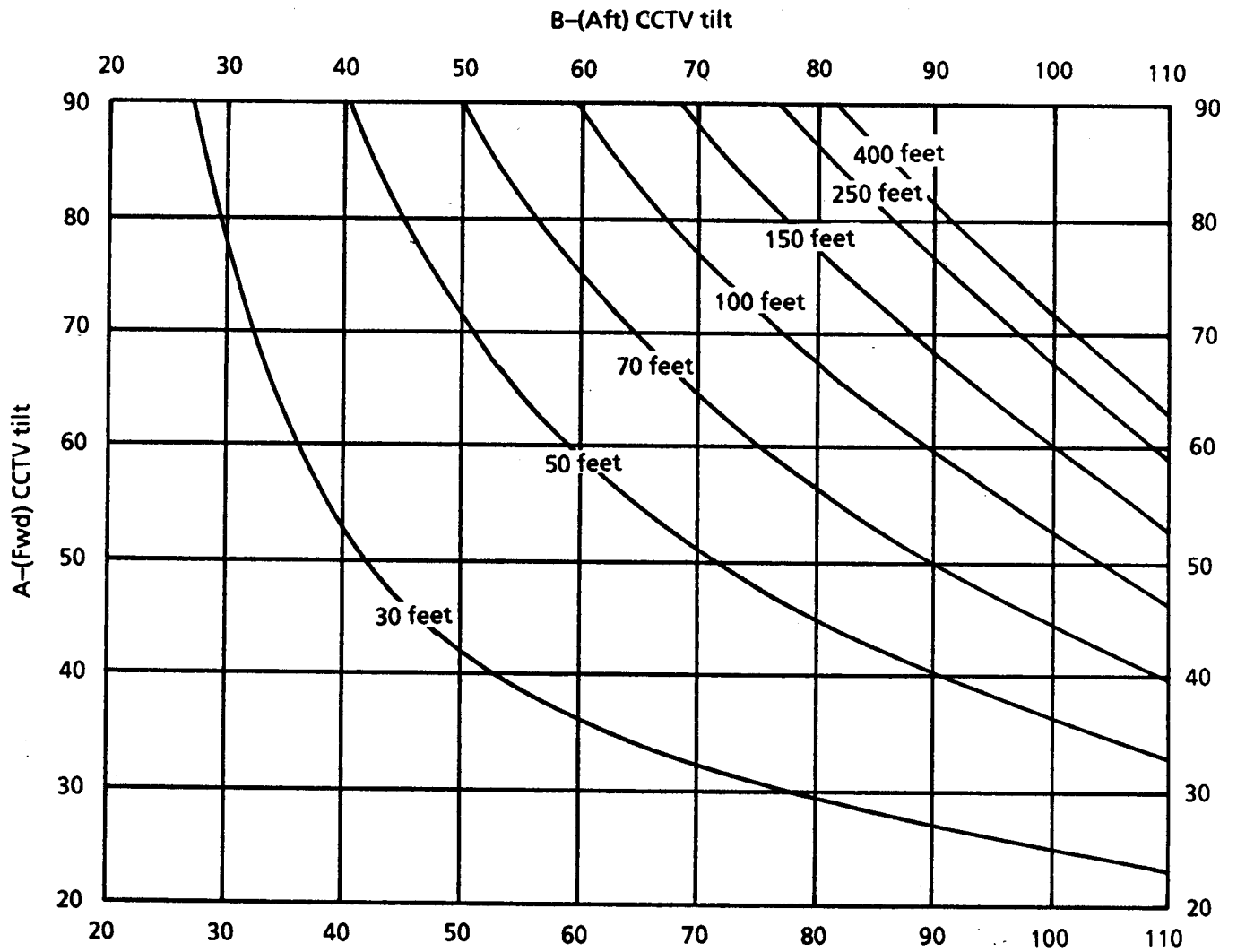
Figure 2-9(b).- Leading target.

Figure 2-9.- Orbiter tracking target.



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Figure 2-10.- Use of CCTVs in ranging.

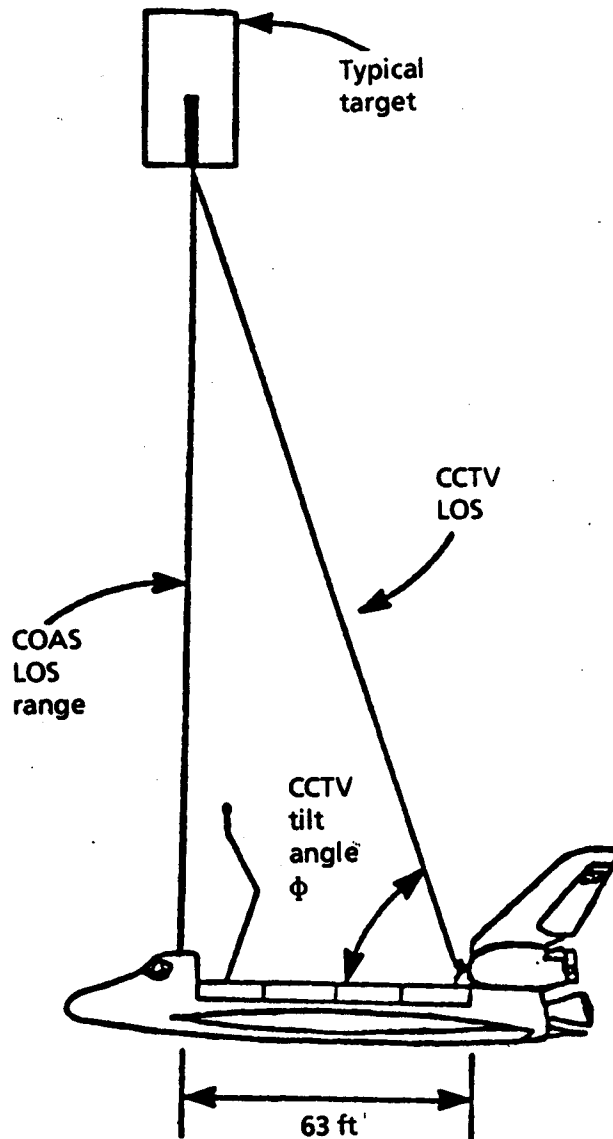


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Figure 2-11.- CCTV ranging chart.

Range Rate Determination

Range rate along the - Z COAS LOS can be measured qualitatively by monitoring target movement across the aft bulkhead camera CCTV monitor. This capability can be enhanced by using the CCTV zoom feature to magnify small relative motion (figure 2-12).



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Figure 2-12.-- Use of CCTV in determining range rate.

CCTV/Overlay

One more way of using the CCTVs in estimating range and range rate, is to use it in conjunction with a calibrated ranging ruler, (see figure 2-13). For a payload the size of the Solar Maximum Mission satellite (SMM), range can be estimated from about 30 to 8000 ft, with a mean error of about 3 percent.

Procedurally, the crew configures the CCTV cameras for either full or no zoom, with underscan. The target is approximately centered, then the bottom line of the applicable range ruler is placed atop the screen image of one end of the optimum axis. Range is read where the other end of the axis falls. A rough estimate of range rate may also be obtained by using the rulers and a stopwatch.

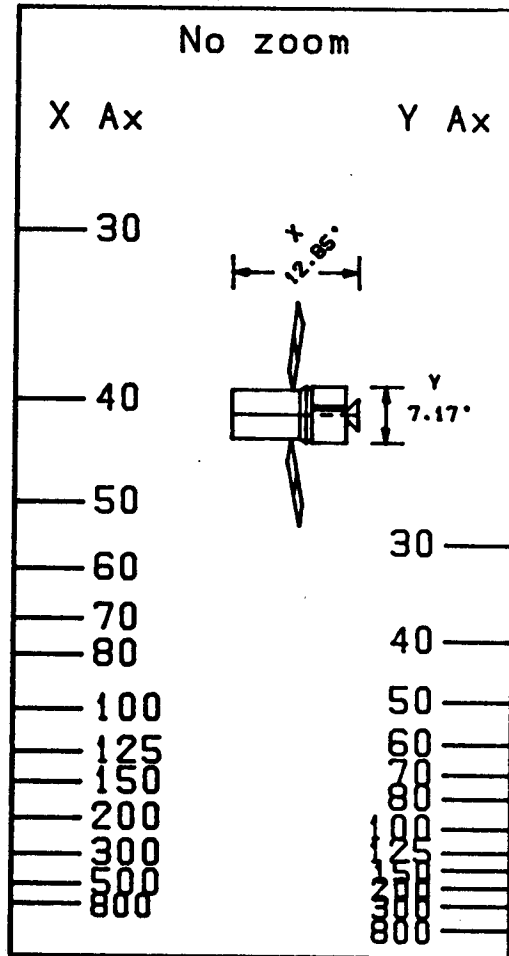
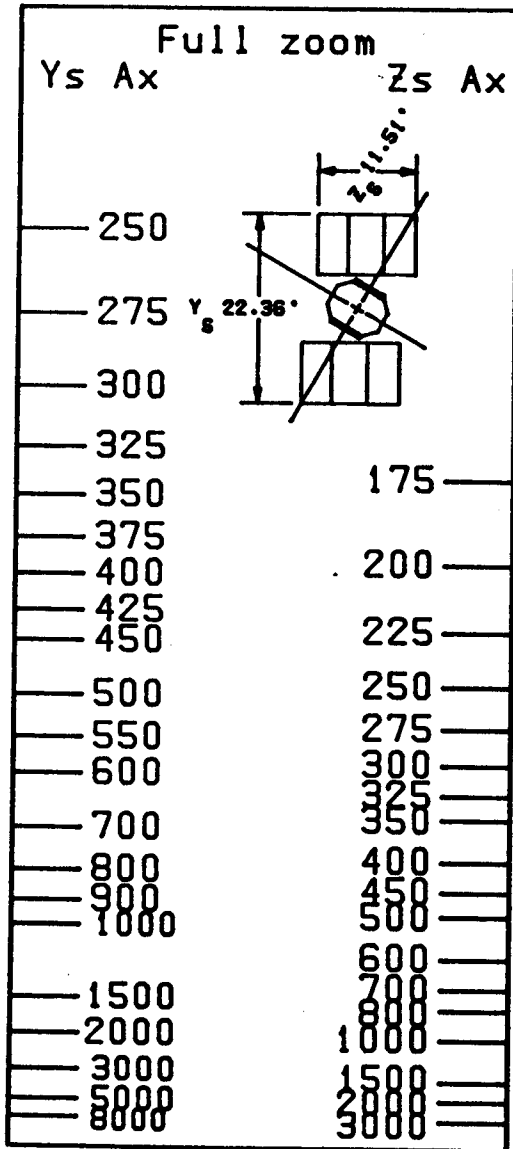


Figure 2-13.- Ranging rulers.

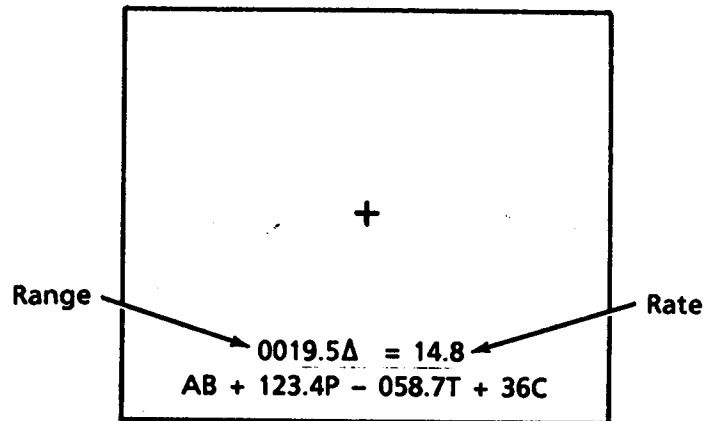
Laser Auto Focus Device

Another ranging device that was tested for the first time on an earlier STS flight was the laser auto focus device.

Originally conceived as part of an auto focusing system for the CCTVs, it was mounted on a payload bay TV camera. The device was used to display range and range rate data on the CCTV monitor as shown in figure 2-14. The

laser beam is parallel with and 4 inches below the TV centerline. The effective range is from 25 to 1000 ft. As there is no automatic lock-on or interface with flight software; all pointing and lock-on has to be controlled by manually using the CCTV that the device is attached to.

The laser is powered on/off by tilting down and panning left at the same time. The laser range finder is not currently planned for use on any rendezvous mission.



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Figure 2-14.- Laser range/rate display on CCTV.

Summary (Section 2)

The purpose of the star tracker is to provide angular data from the Orbiter to a target being tracked and to determine the Orbiter attitude in inertial space for aligning the IMUs.

The purpose of the COAS is to sight stars or a target manually, align the IMUs if the STRKs have failed, incorporate angle data into RNDZ NAV and visually determine range and LOS angle to a target.

The RR provides range, range rate, roll (azimuth), and pitch (elevation) information about the target to the GNC GPCs.

The RR can detect a passive target at about 15 n. mi. and a target with an active compatible transponder at 300 n. mi.

Section 3: Rendezvous Maneuver Sequence

Introduction

A rendezvous can be thought of as the act of bringing two vehicles together. In the context presented, one vehicle is referred to as the target vehicle and the other as the chaser vehicle. The target vehicle is assumed to be in a known stable orbit. A series of maneuvers will be performed by the chaser to ensure that it will rendezvous with the target at a desirable position in the target's orbit and within a prescribed amount of time. The desired target position at rendezvous depends primarily on lighting conditions, and the prescribed amount of time can range from hours to days. In practice, the scheduled rendezvous maneuvers will leave the Orbiter in a stable position 800 ft from the target. The actual final approach to the target is considered a part of proximity operations (PROX OPS).

The reader may see the letter "N" preceding many of the rendezvous burn acronyms referred to in other literature as meaning normal or nominal. Neither is correct. The N was originally designed as a variable to number which rev in the mission the burn would be performed on. The N has since become an accepted part of the acronym and a number has simply been added to the end of the acronym to sequentially number burns used more than once.

In order to accomplish a rendezvous mission, several items must be taken into consideration:

- A. A target vehicle's orbital altitude may be greater than can be reached in powered flight using the Orbiter's main engines and solid rocket boosters. Thus, main engine cutoff (MECO) will occur below the target's orbit, and subsequent burns will be required to force the Orbiter's orbit to intercept the target's orbit.
- B. Achieving an appropriate phase angle between the target and the Orbiter at orbit insertion either will restrict the launch window or will

force the inclusion of a phasing maneuver which would allow the Orbiter to "catch up" with the target.

- C. The launch window and the direction of the launch are designed so that the Orbiter will be approximately within the orbit plane defined by the target vehicle. The time of launch within the launch window, as well as errors which develop during the powered phase of ascent, may result in the Orbiter and target being in slightly different orbit planes. The rendezvous sequence, then, must include the potential for an out-of-plane corrective burn.
- D. Of particular importance are the lighting conditions during the final portions of the rendezvous phase. As the final maneuvers are to be performed visually by the crew, both vehicles must be in the light. Furthermore, they must remain in the light long enough to allow completion of the rendezvous and PROX OPS phases, as well as satisfy the mission objectives (deploy/retrieve payload, etc.). In a similar way, many of the onboard targeted maneuvers are scheduled to occur with the target in the light, so that star tracker marks may be taken and incorporated into the navigation software before burns are executed.
- E. The rendezvous sequence must have the flexibility to compensate for the errors inherent in the navigation, guidance, and flight control systems.
- F. The rendezvous sequence also must not violate any other mission rules. Thus, the launch window must allow for the correct lighting conditions at launch, and, in case of aborts, enough reaction control system (RCS) fuel must remain onboard to

accomplish the mission, as well as to return to the landing site in case of (OMS) failure.

- G. Lastly, with all these items considered, the resultant launch window must be of a realistic duration.

In view of all the above considerations, a general sequence of rendezvous maneuvers has been developed. While all the maneuvers need not be performed during each mission, the software capability will be available in order to add any of the maneuvers, as necessary, to the rendezvous sequence. Note that the goal of these maneuvers is a successful rendezvous of the chaser (the Orbiter) and target vehicles under the appropriate conditions (i.e.; lighting), and thus, the earlier maneuvers are designed to provide the correct conditions for the final maneuvers.

Rendezvous Maneuvers

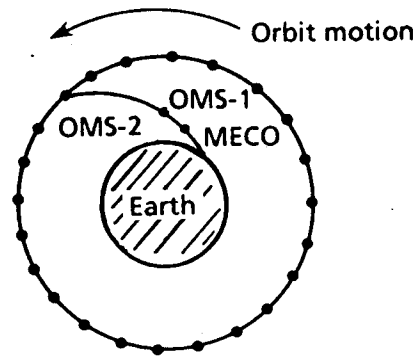
It is not possible to give a generic rendezvous profile since the Shuttle missions involving rendezvous will have considerable differences. Some missions will involve re-rendezvous with a payload released earlier in the mission while others will be designed to go from launch to a rendezvous with an orbiting satellite. The following paragraphs will consider two baseline profiles that encompass a majority of the rendezvous maneuvers.

The following profile includes the maneuvers from orbit insertion (OMS 1) and ending with the first braking maneuver which places the Orbiter in a stationkeeping position near the target from which the final approach is made. The braking maneuvers and the final approach to the target are considered part of PROX OPS.

OMS 2

Following MECO, the Orbiter is in approximately a 80 x 40 n mi. elliptical orbit. The OMS 1 burn is used to raise the Orbiter to an orbit with an apogee >95 n. mi. This burn is called the insertion burn. The rendezvous sequence is considered to begin with the next maneuver, OMS 2. This burn is designed not only to raise the orbit to a safe level, but it is

planned as a coarse phasing burn to control the catchup rate. The opportunity also exists at this time to circularize the orbit. A circularization burn would be used to adjust the lighting conditions for some future event (figure 3-1). OMS 2 is accomplished with a ground targeted Hohmann transfer and thus is performed at the apogee of the post-OMS 1 orbit.



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Figure 3-1.- OMS 2 burn.

Direct Insertion

For rendezvous taking place in high orbits, it is more fuel efficient to delete the OMS 1 burn and use the main propulsion system (MPS) propellant to create an apogee near the target's orbit. This technique is called direct insertion. The apogees that can be achieved vary from nominal to 300 n. mi.

Phase-Adjust Maneuver (NC 1)

Following the OMS 2 burn, the Orbiter is in an orbit which is considerably lower than the target's orbit. Thus, there is a significant catchup rate between the two vehicles. Depending on the phase angle between the Orbiter and target at insertion, it is possible to raise the Orbiter's orbit to adjust and decrease the catchup rate and, if OMS 2 resulted in a circular orbit, make the two orbits coaxial (share a common line of apsides). This maneuver is called the NC 1 maneuver (figure 3-2). Also, it is possible to lower the orbit at this time to provide for additional catchup capability for larger insertion phase angles. For a 4-day rendezvous, this technique allows for a successful rendezvous with any insertion phase angle. The NC 1 maneuver is a ground targeted

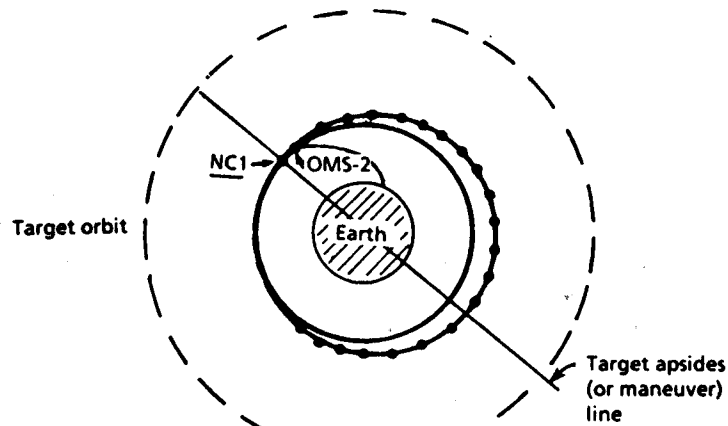


Figure 3-2.— An NC 1 burn.

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OMS burn based on post-OMS 2 ground tracking of the Orbiter and the target. Although it is possible to bypass an NC 1 burn (leave the Orbiter in the OMS 2 orbit until phasing is accomplished), the burn is typically performed to adjust the catchup rate in order to optimize the lighting and ground coverage conditions for the remainder of the rendezvous.

Plane Change (PC)

Depending on the magnitude of the wedge angle between the Orbiter's and the target's orbital planes, it may be necessary to insert a plane change (PC) maneuver. If this is necessary, a ground targeted out-of-plane burn

will be made at a node between the two planes in order to make them coplanar. This maneuver can be inserted whenever desired.

Height Adjustment (NH)

After the necessary phasing is accomplished following NC 1, the phase angle will be the correct size such that the Orbiter can rendezvous with the target in a given timeframe. The next goal is to raise the Orbiter's apogee to within 10 n. mi. of the target's orbit. This is done with an OMS burn performed at the apogee of the post NC 1 orbit (figure 3-3). With respect to the Earth's surface, the NH, like the NC burn, is a horizontally executed maneuver.

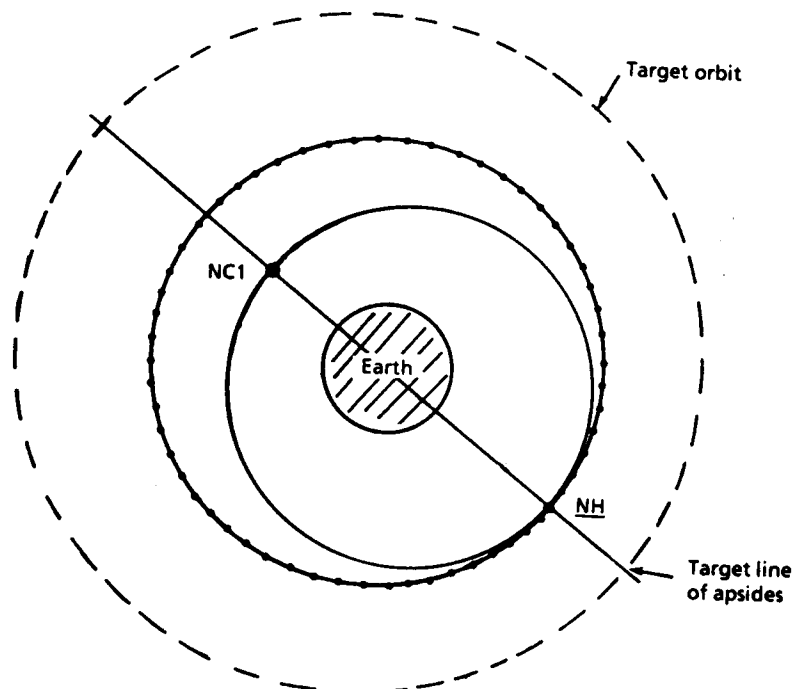


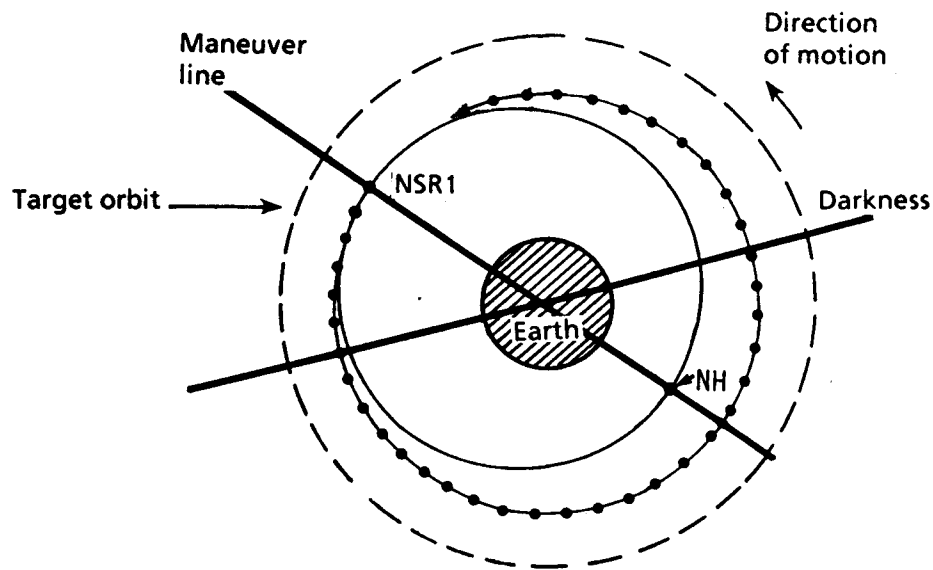
Figure 3-3.— An NH burn.

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Coelliptic Maneuvers Normal Slow Return (NSR)

The NSR burn sets up coelliptic orbits between the target and Orbiter (figure 3-4). It is performed using a ground targeted solution and is generally preceded by an NC and NH combination or by a NCC maneuver.

Placing the Orbiter in an orbit coelliptic with the target is beneficial in obtaining the appropriate distance or lighting conditions necessary for the remainder of the rendezvous. The catchup rate between the target and Orbiter is diminished because of the decreased ΔH between the vehicles and thus the Orbiter can wait for the most opportune condition from which to begin the next maneuver.



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Figure 3-4.- An NSR 1 burn.

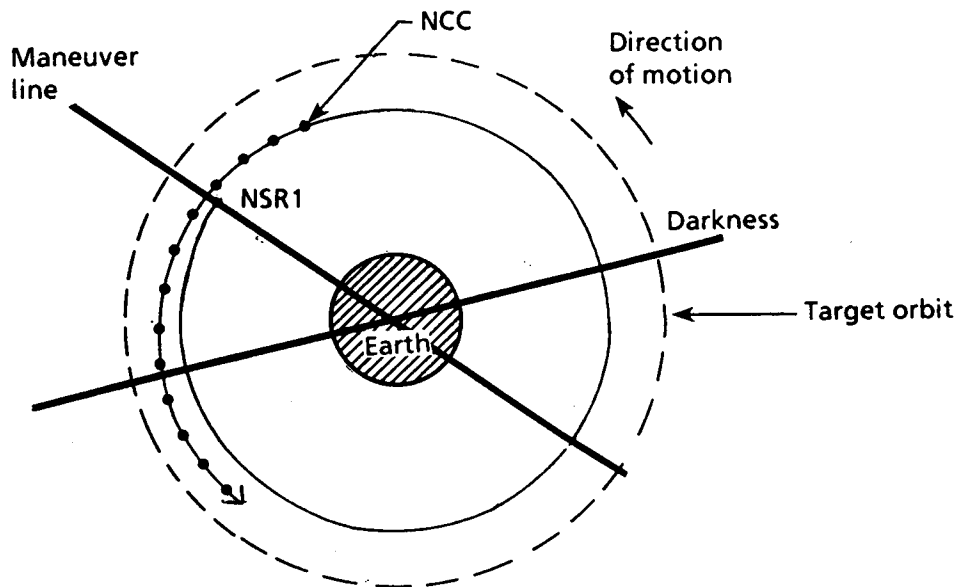
Corrective Combination (NCC)

The NCC maneuver is the first onboard targeted burn based on onboard sensor data (STRK, RR, or COAS) (figure 3-5). Using Lambert targeting, the maneuver is targeted for a desired Orbiter-target phase and differential altitude or an Orbiter-target offset position at a future NSR maneuver. Using external sensor data, a non-zero wedge angle between the Orbiter and target may be discovered. Combining an NCC maneuver with an NSR 2 burn, the capability is provided to correct for any non-zero wedge angle, as well as to raise the Orbiter's orbit. If a planar correction is necessary, one of the burns must occur at a node between the Orbiter and target planes. To establish more flexibility, the first burn, NCC, is designed to establish a node 37 minutes later (at what will be the NSR 2 point). The amount of time (37 minutes) was chosen to allow time for the crew to enable navigation marks to be taken on the target using the RR if the target is active.

If the target has an active transponder, RR data may be obtained at about 300 n. mi. and

be incorporated in navigation. However, if the target is passive, or in case of an RR failure, STRK data will be used to update the onboard computed Orbiter/target relative states. Thus, the lighting prior to NCC must be correct (target in the sunlight), such that STRK target tracking can take place for several minutes. Time must also be allowed for the onboard software to target the first maneuver (NCC), the crew to make comparisons with ground solutions and, if required, maneuver to the correct burn attitude before the burn. With these considerations in mind, NCC is planned to occur at a point such that at NCC - 40 minutes (and for several minutes thereafter), the target will be in light.

The crew must enable rendezvous navigation via the REL NAV SPEC (see Relative Navigation Display section) in MM 201, which enables the use of external sensor data (ST/RR/COAS) in the GNC software. Also, the crew will initiate onboard targeting using a burn target set and the ORBIT TGT display (Orbit Targeting Display section). The maneuver is executed using the ORBIT MNVR EXEC display (Orbit MNVR EXEC Display section), which is the OPS display in MM 202.

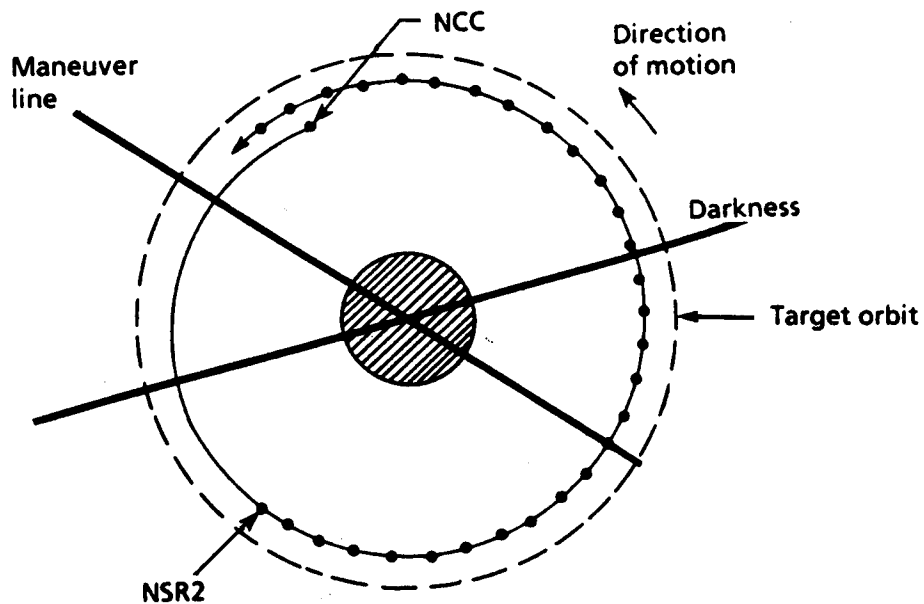


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Figure 3-5.- An NCC burn.

Second Coelliptic Maneuver (NSR 2)

NSR 2 is the second maneuver in the NCC/NSR corrective combination maneuver set and is designed to null any out-of-plane velocity between the Orbiter and target at the node established at 37 minutes past NCC (figure 3-6). NSR 2 is also targeted to hit the nominal point for the next maneuver, terminal initiation (TI), in an amount of time which would make the resultant orbit coelliptic with the target's orbit, assuming small dispersions errors in navigation and targeting. This maneuver is onboard Lambert targeted, with the crew initiating the onboard targeting software through the ORBIT TGT display and executing the burn through the ORBIT MNVR EXEC display.



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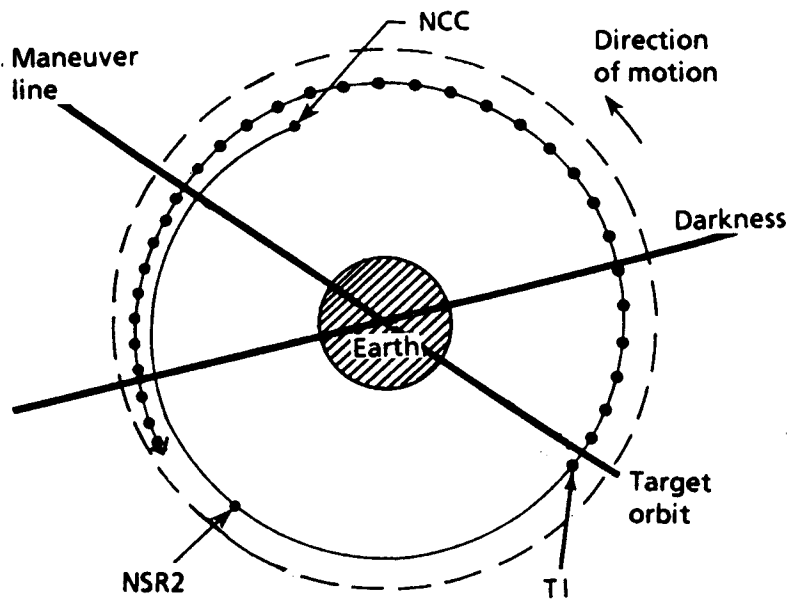
Figure 3-6.- An NSR 2 burn.

Terminal Initiation (TI)

The TI burn is designed to change the approach from one of phasing to a direct intercept (figure 3-7). The burn occurs approximately 5 minutes before orbital noon at the apogee of the post NCC orbit. It is targeted onboard for a direct transfer to the target in 320 degrees of target orbital travel. The choice of a 320-degree transfer is to ensure a 'hot' closing trajectory at intercept. This will allow

some margin for dispersed low energy trajectories by targeting so that the Orbiter will pass through the target.

Any delay in the rendezvous profile would occur at TI. A ground computed PEG 7 burn solution would be used to cause an Orbiter/target separation of 1 n. mi./rev. A recomputed onboard targeted TI solution would be used to resume the rendezvous after a one rev delay. After a 24-hour delay, the rendezvous would begin with a recomputed NC burn.



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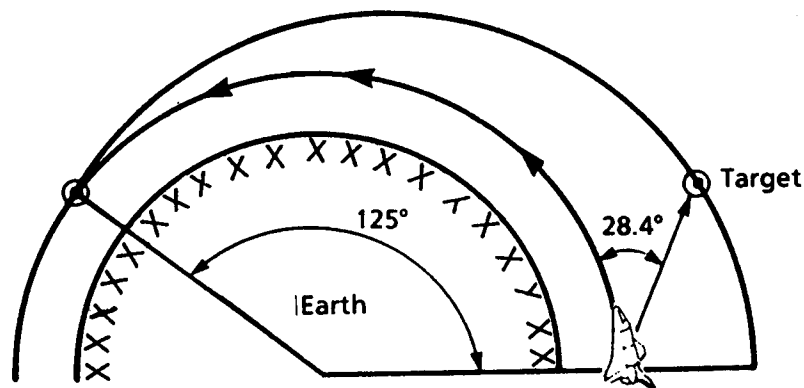
Figure 3-7.- TI maneuver.

Midcourse Correction Maneuvers (MC)

The midcourse correction maneuvers will be performed as necessary to ensure a correct intercept trajectory from TI to the target. Using rendezvous radar to update navigation, four midcourse correction burns are planned. The first burn, MC 1, is executed approximately 8 minutes prior to sunset. This allows the opportunity to take STRK marks before and

after the burn in case the radar is failed. The burn is Lambert targeted and performed with RCS, axis by axis, in $-Z$ track attitude.

The second burn, MC 2, places the orbiter on an intercept trajectory for the target such that interception occurs in 125 degrees of orbit travel. The cue to begin this maneuver is the establishment of a specified elevation angle (28.4) between the Orbiter's local horizontal and LOS to the target vehicle. (See figure 3-8).



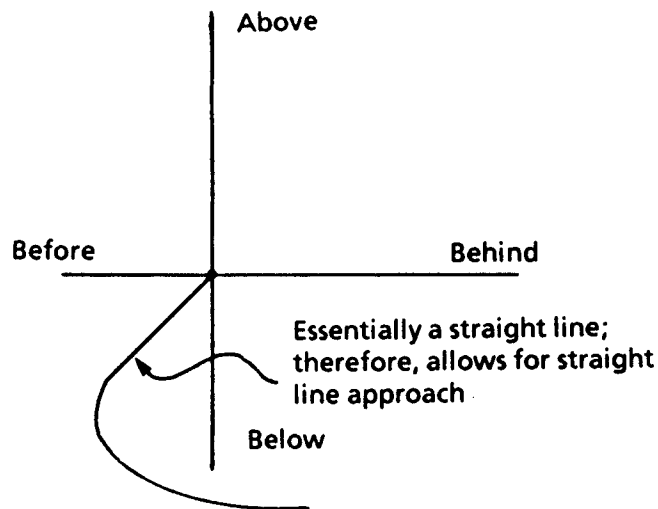
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Figure 3-8.- MC 2 initiation.

An elevation angle of 28.4 degrees and interception in 125 degrees of orbit travel (approximately 32 minutes) results in certain desirable conditions at target intercept. Examination of the relative motion plot from MC 2 to intercept using these values (figure 3-9) reveals that this allows for a low LOS angular rate and a low terminal approach velocity. More important, the slope of the LOS becomes constant prior to interception, and thus, an inertial LOS is established, which simplifies the final maneuvers.

The MC 2 maneuver is an onboard Lambert targeted burn. It is planned to occur in the middle of a night pass to ensure that the lighting will be correct for the final approach maneuvers. RR marks will be taken prior to the burn. If the radar is unavailable, MC 2 will be the last targeted midcourse burn.

If needed, MC 3 and MC 4 will be executed 10 and 20 minutes, respectively, after MC 2. Both burns are targeted for target intercept and performed with RCS axis by axis.



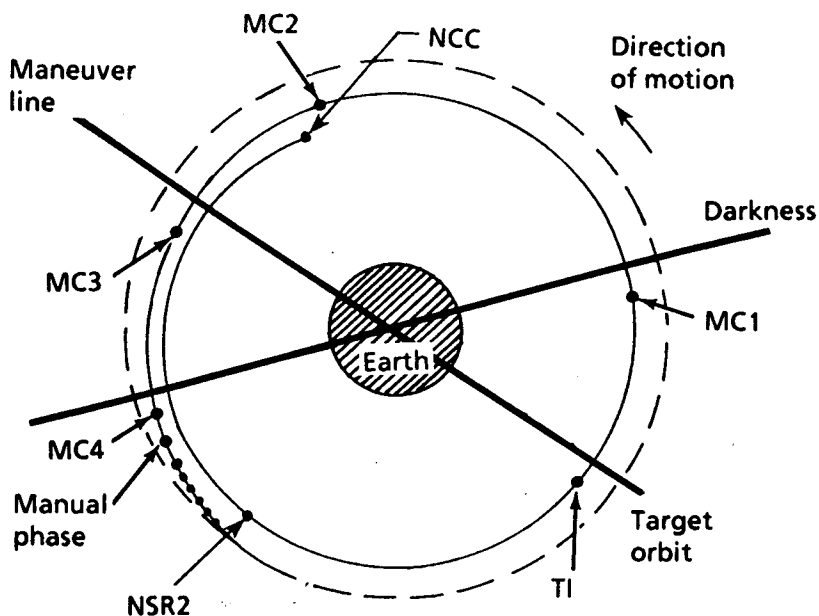
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Figure 3-9.- Relative motion plot for 130° of travel.

Manual Line of Sight Control Terminal Phase Finalization (TPF)

The trajectory established at TI (and corrected as necessary with the MC maneuvers) is designed to intercept the target. The TPF maneuvers include a series of breaking gates which establish a specified closing velocity at fixed distances from the target. The TPF phase begins 22 minutes after MC 2, when the Orbiter is put into inertial attitude hold (DAP:MAN/DISC/DISC/DISC) (figure 3-10). The visible target

is kept within the COAS field of view and stationary to maintain a zero LOS rate, while the radar is used to establish the breaking gates (no greater than 4 ft/s at 2000 ft and 2 ft/s at 1500 ft). At the end of the TPF, the Orbiter should be in a stable stationkeeping position at a specified distance from the target. The distance will vary according to mission requirements. Further maneuvers, such as to other offset positions, are considered a part of PROX OPS, which will be discussed in section 4, Proximity Operations Maneuvers. In some documentation, the TPF maneuvers are considered part of PROX OPS.



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Figure 3-10.- MCC maneuvers.

Summary (Section 3)

NC 1 (phasing):

Posigrade Hohmann transfer from post-OMS 2 apogee or circular orbit

Performed on TGT line of apsides

Post-NC 1 orbit stay time dependent on true anomaly to target

PC (plane change):

Ground targeted

Performed at node between the two planes

NH (height adjustment):

Posigrade Hohmann transfer

Performed at apogee of post-NC 1 orbit

Raises post-NC 1 perigee to an altitude at or near the target's orbit

NSR 1 (first coelliptic):

Posigrade and coplanar

Preceded by an NC and NH combination or by an NCC maneuver

Places the orbiter in an orbit coelliptic with the target

NCC (corrective combination):

First onboard targeted burn

Lambert targeting

Targets for a desired Orbiter/target phase and differential altitude or an Orbiter/target offset position at a future NSR maneuver

Establishes NSR 2 node 32 minutes later

NSR 2 (second coelliptic):

Aplanar

Results in a coelliptic orbit with the target

TI (terminal initiation):

Lambert targeting

Performed five minutes before noon at the apogee of the post-NCC orbit

Results in TGT intercept after 320 degrees of orbit travel

MC 1 (first midcourse correction):

Lambert targeting and coplanar (nominally zero)

Performed 8 minutes before sunset

Executed using RCS axis by axis

MC 2 (second midcourse correction):

Lambert targeting

Burn is performed on an elevation angle of 28.4 degrees

Results in target intercept after 125 degrees of orbit travel

MC 3 and MC 4 (third and fourth midcourse correction burns)

Lambert targeting (nominally zero)

Performed when radar data is available

Executed 10 and 20 minutes respectively after MC 2

Manual phase:

Performed in daylight

Begins 22 minutes after MC 2 when the Orbiter is put in inertial attitude hold

Braking gates used to control closing rates

THC used to keep Orbiter stationary in COAS field of view

Section 4: Proximity Operations Maneuvers

Introduction

The proximity operations (PROX OPS) maneuvers are designed to take the Orbiter from its final, stable position following the rendezvous maneuvers to an appropriate position for remote manipulator system (RMS) capture of a payload, or to satisfy some other payload or mission objective. The PROX OPS maneuvers may include transitions to specified offset positions, stationkeeping, approaches, flyarounds, and finally, after all mission objectives are satisfied, the separation of the Orbiter to a stand-off position. In certain documentation, the TPM and TPF maneuvers covered in the TI and MC Maneuvers sections are also considered a part of PROX OPS. First, certain considerations affecting PROX OPS maneuvers will be examined, and then the basic PROX OPS maneuvers will be described.

PROX OPS Considerations

The position in which PROX OPS leaves the Orbiter after the final approach and the techniques used to maneuver the vehicle to this position depend on several considerations. One of these concerns the effects of the RCS plume on the target (plume impingement). The concern here is with target contamination and the resultant body rates and torques (overpressure) on the target. Another consideration concerns RCS fuel usage, particularly from the forward pod. This consideration will also influence the choice of braking techniques and stationkeeping positions near the target, and at times it may conflict with the plume impingement concerns. Another consideration is whether it is more efficient to compute the required maneuvers using onboard targeting software, or to have the crew manually execute the maneuvers using "out-the-window" data; e.g., COAS, RR digital readouts, etc. In all cases, crew visibility and procedural simplicity impact the selected PROX OPS maneuvers.

Plume Impingement

One of the primary factors in the selection of braking methods and close-in stationkeeping positions for the Orbiter is that of reaction control system (RCS) plume impingement on the target vehicle. The problem includes both contamination of the target and the resultant induced body rates and torques on the target.

Recall that the primary reaction control system (PRCS) thrusters generate 870 pounds of thrust with an MMH/N₂O₄ propellant system. These engines can either fire continuously or operate in an 80-msec pulse mode. The RCS engines are located in the forward and aft modules of the Orbiter (figure 4-1).

Note that the PRCS jets are labeled according to the direction of the resultant thrust; i.e., the -X RCS jets are located in the forward part of the Shuttle and produce a thrust in the -X direction (aft).

Certain types of payloads may be sensitive to the hydrazine molecules in the PRCS plume. For example, a payload containing delicate optical equipment could be damaged by an intense blast made by the PRCS at a short distance. In order to avoid these problems, the PRCS typically will be in a pulse mode near the target to avoid continuous firings. Which jets will be fired, as well as the distance and orientation of the Orbiter with respect to the target, may be adjusted to reduce the contamination problem.

In varying degrees, all payloads will be sensitive to the overpressure and resultant body rates/torques which result following a direct PRCS firing in close proximity to the target. The degree of impact depends on the relative distance between the target and payload, which PRCS jets are fired, and the mass and affected surface area of the target. It also depends on whether the target vehicle has an active control system which compensates for the applied torques and body rates.

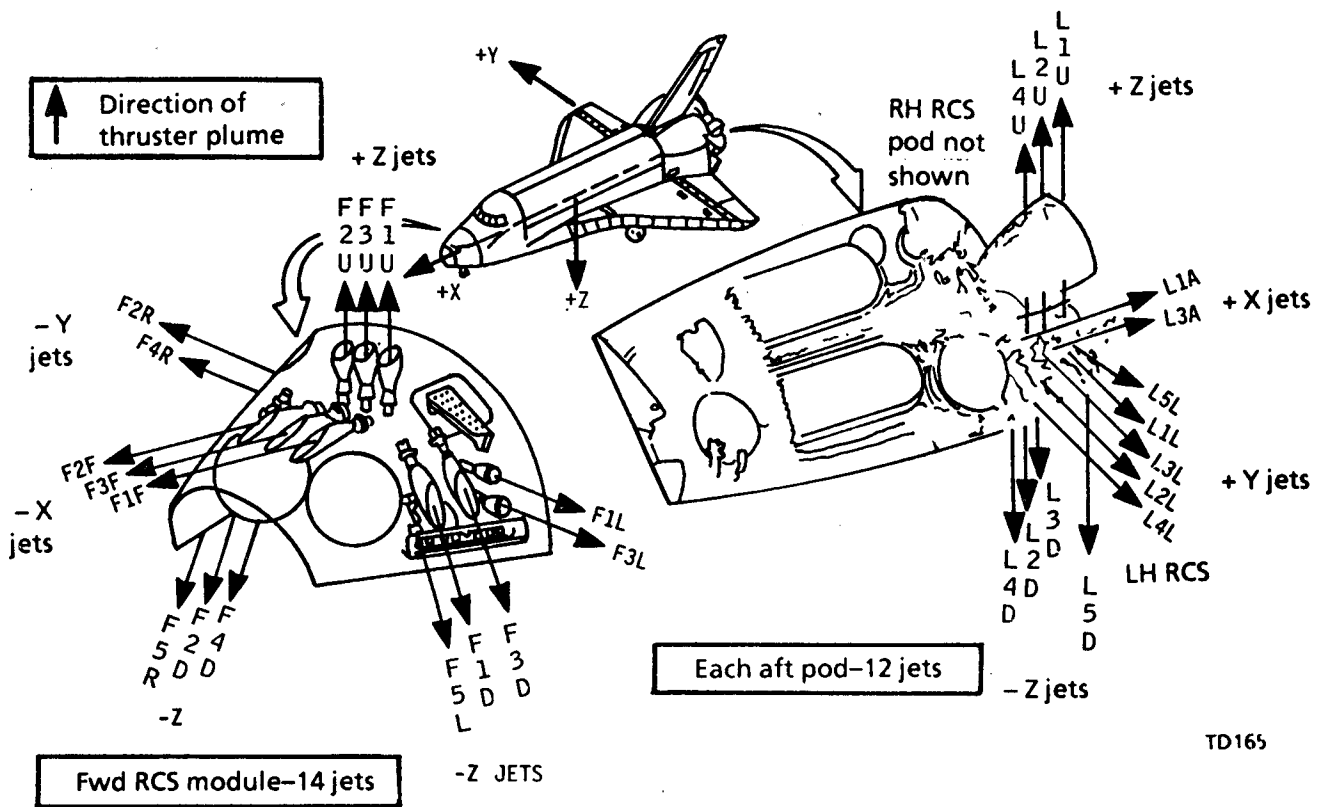


Figure 4-1.- RCS thruster identification.

To alleviate the plume problems, it is desirable to minimize direct firings at a target by using different braking schemes (see the Braking Techniques section). Unfortunately, the best alternate braking method has a higher RCS fuel cost. The other way to alleviate the problem is to identify a "safe" distance from the target (the contamination/overpressure (C/O) sphere) and not allow any direct PRCS firings at the target within that distance. This distance may range from 25 to 1000 feet, depending on the target.

RCS Fuel Conservation

Due to the number of maneuvers that may be required to accomplish a rendezvous mission and the mission rule which states that enough RCS fuel must be available during the deorbit maneuver in order to reenter using the RCS jets in case of an OMS failure, it is necessary to be conservative in the use of RCS fuel. Unfortunately, one of the best braking techniques with respect to plume impingement is also the most costly with

respect to RCS fuel consumption (see Low Z section).

Additionally, there is a more severe RCS fuel problem with the forward propellant than the aft, as it is possible to crossfeed OMS tank propellant to the aft RCS jets. Thus, special braking techniques have been developed which minimize the usage of the forward RCS jets; i.e., they use the aft RCS jet firings. One of the techniques is called orthogonal braking.

Braking Techniques

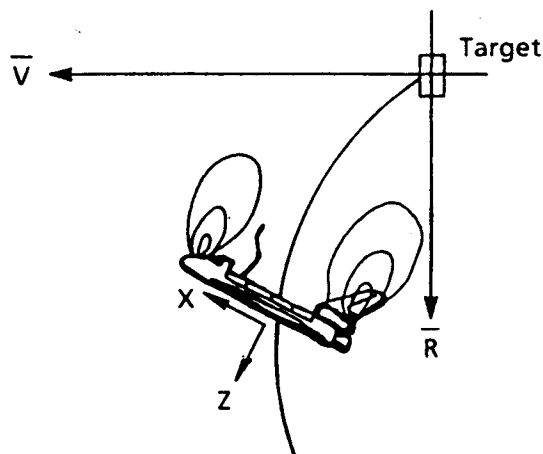
In order to achieve the goals of reducing plume impingement and saving fuel (especially the forward tank propellant), as well as ensure that the Orbiter can successfully reduce its relative velocity with respect to a target, several braking techniques have been established. Three of them rely on the appropriate use of the PRCS jets, while one relies on orbital mechanics to provide a braking action.

Normal Z-axis Braking

The normal Z-axis PRCS braking technique uses the upward firing + Z PRCS thrusters for braking. These thrusters provide for an Orbiter acceleration in the + Z direction (figure 4-2).

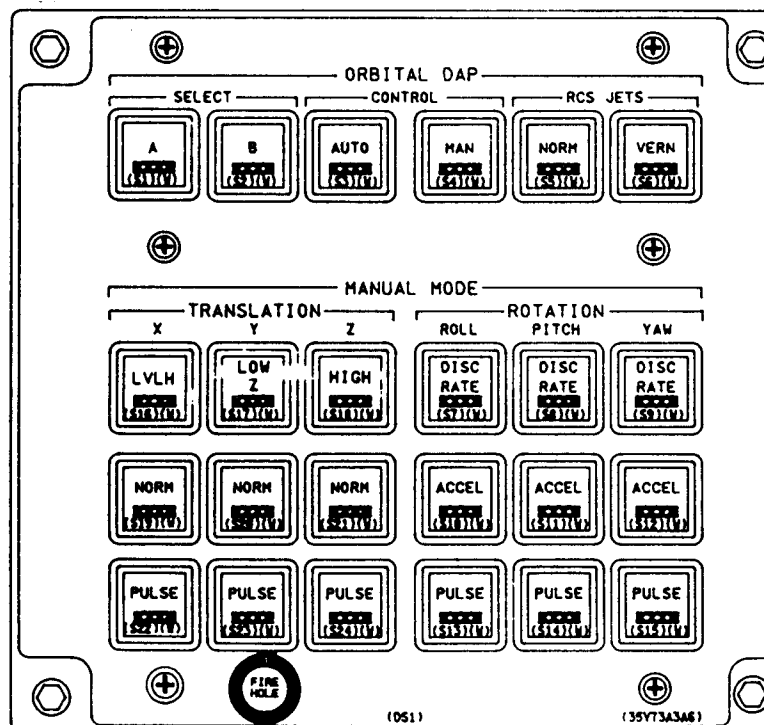
This technique is used when the target is approaching along the Orbiter's -Z axis. Good target visibility is maintained through the overhead window for the crew. Also, the RR can be utilized in this configuration. The + Z thrusters can be used to null precisely any final closing velocity between the Orbiter and

target by providing approximately 0.45 ft/sec^2 acceleration in the + Z direction, assuming a 180 000-pound Orbiter. The propellant cost for this amount of acceleration is relatively low: 30 lb/ft/sec for a 180 000-pound Orbiter. However, this technique maximizes plume impingement on the target, as the + Z PRCS jets will fire directly at the target. Thus, this technique will only be used outside of a target's C/O sphere of influence, or in case of an emergency (such as an imminent collision with the target). This type of thrusting is crew selectable via the NORM Z pushbutton on the Orbital digital autopilot (DAP) panel on panels C3 (figure 4-3) or A6.



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Figure 4-2.- Normal Z-axis braking.



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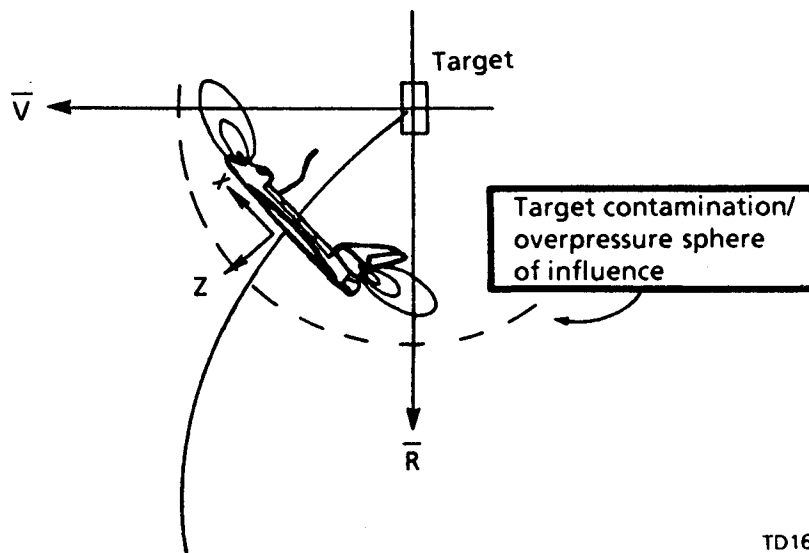
Figure 4-3.- Panel C3: Orbital DAP panel.

Low Z Axis Braking (\pm X PRCS)

The low Z-axis braking technique utilizes the cant of the aft firing thrusters (+ X PRCS) and the scarf (nozzle extension) of the forward firing thrusters (-X PRCS) in order to produce a + Z Orbiter acceleration (see figure 4-4). In this mode, the orbital DAP nulls any residual force in the X-axis direction. This technique, like the normal Z-axis braking technique, is used when the target is approaching along the Orbiter's -Z axis. This technique also allows good crew and radar visibility of the target. Furthermore, it minimizes plume impingement on the

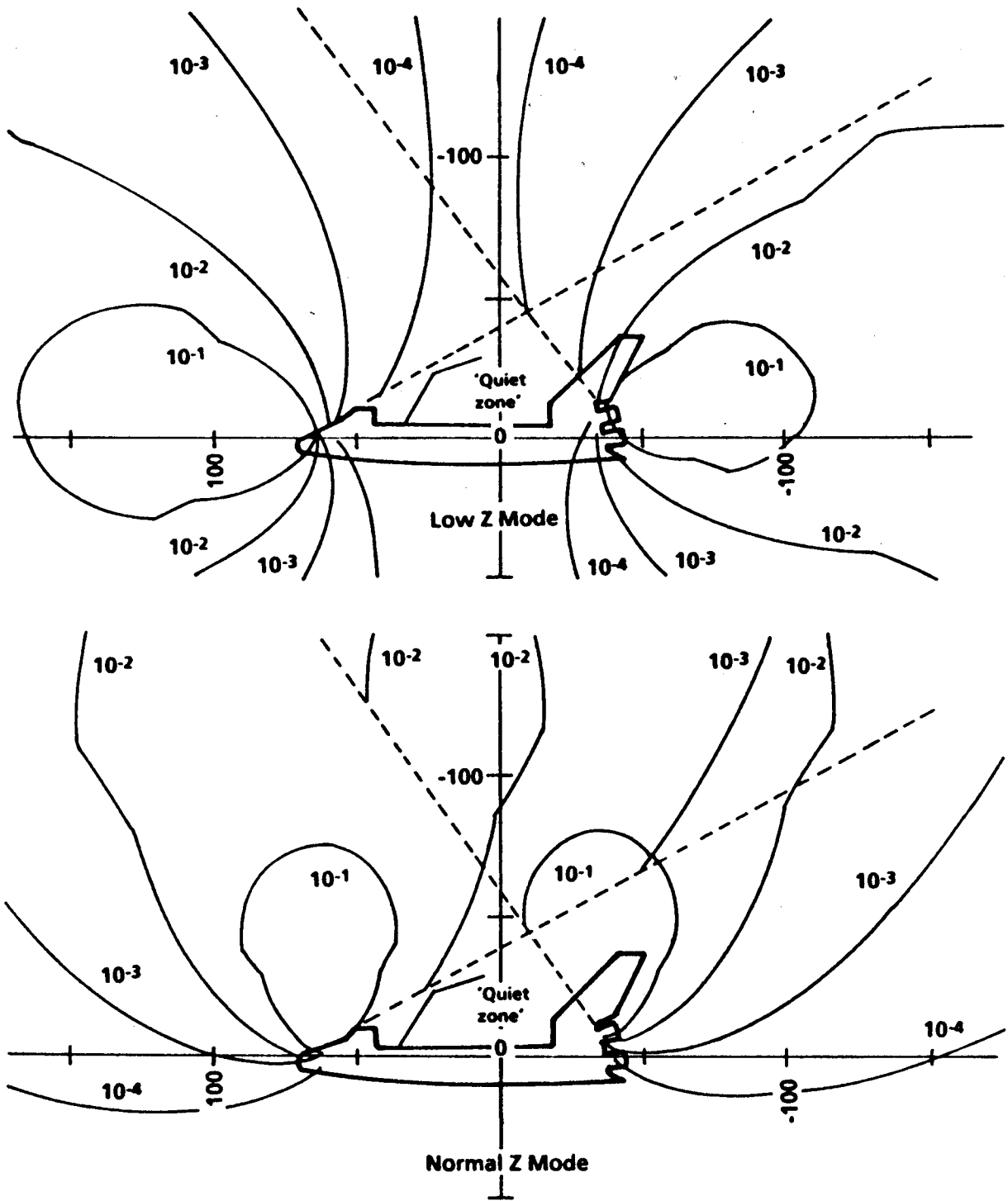
target (see figure 4-4A), while providing a 0.05 ft/sec^2 acceleration on the Orbiter in the + Z direction. Unfortunately, the RCS fuel expenditure in this mode (281 lb/ft/sec) is significantly greater than in the normal Z mode due to the "wasted" fuel expelled in the + X and -X directions. Thus, this mode typically will be used only within the target's C/O sphere.

This mode is crew selectable with the LOW Z pushbutton on the Orbital DAP panel on panels C3 or A6.



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Figure 4-4.- Low Z-axis braking.



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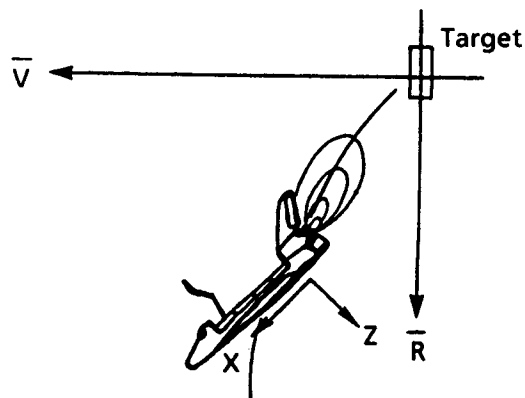
Figure 4-4A.- RCS plume overpressure for LOW Z and NORM Z mode (lb/ft²).

The + X Axis Braking/Orthogonal Braking Technique

In order to minimize forward pod propellant consumption, the aft + X PRCS thrusters alone can be used to provide the necessary braking force. While it is possible to use this technique in the situation where the target is approaching along the Orbiter's -X axis as in figure 4-5, it results in plume impingement problems, as well as poor target visibility (for both crew and radar). The acceleration provided by the + X thrusters is 0.3 ft/sec^2 and the fuel consumption is 30 lb/ft/sec .

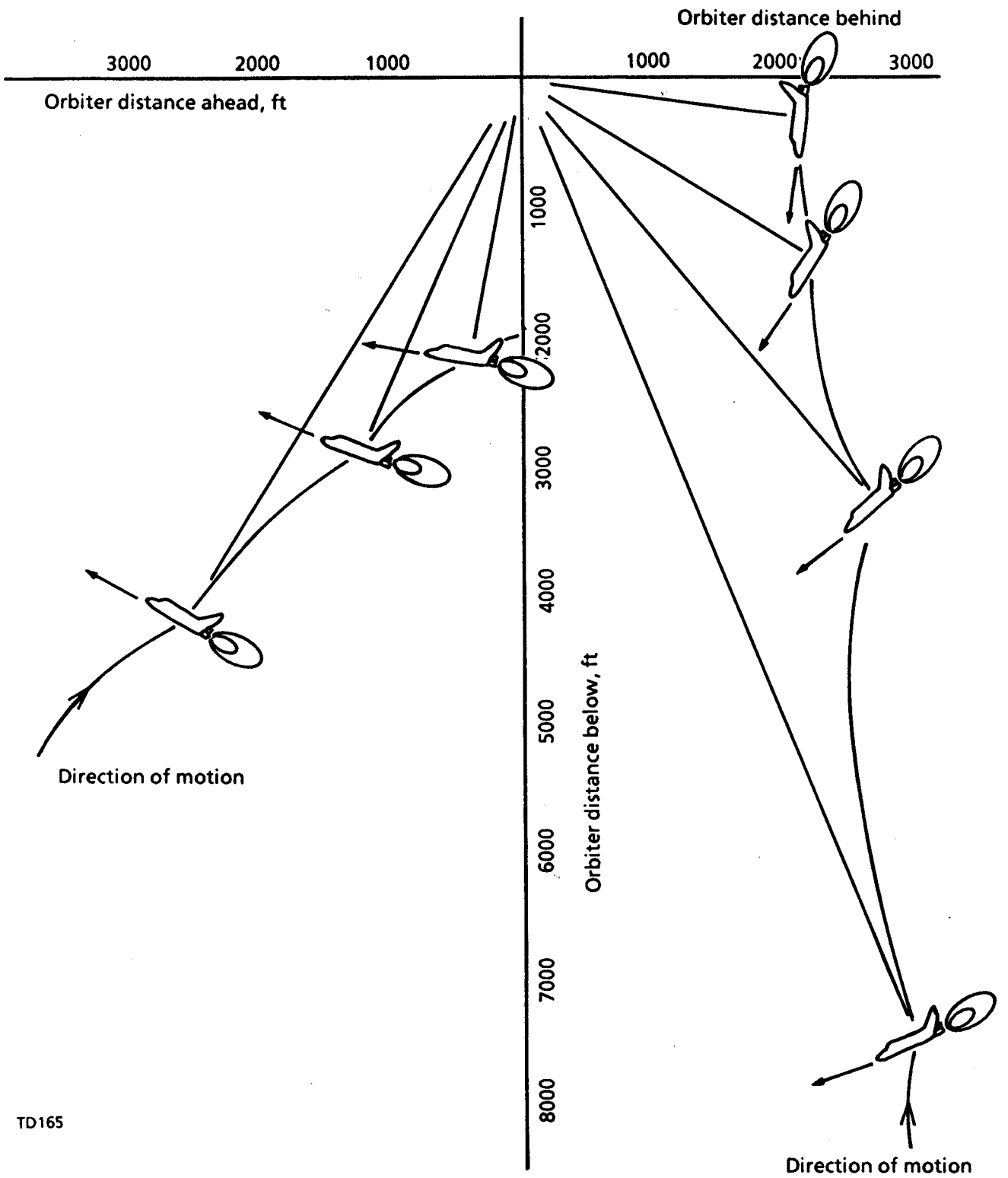
Another use of + X axis braking is in a technique called orthogonal braking. In order to reduce the possibility of plume impingement on the target, the + X burns are made perpendicular to the LOS to the target vehicle. In this technique, orbital mechanics

effects, supplemented by the + X jet firings, result in the Orbiter braking with respect to the target, as shown in figure 4-6. The braking scheme is accomplished by placing the Orbiter in the target track mode, with the -Z axis pointed toward the target. Trajectory analysis is done on the ground preflight to determine at which points, with a specified time interval between them (approximately 5 seconds); in the intercept trajectory the Orbiter's velocity vector is perpendicular to the target LOS. These positions and times for the braking maneuvers are part of the I-load. As it is fairly difficult to visually determine when the Orbiter's velocity vector is orthogonal to the target LOS (the -Z axis), the onboard targeting software typically will be used to update the I-loaded positions and times, and to compute the magnitude of the thrust necessary to intercept the specified offset position using only RCS firing perpendicular to the target LOS.



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Figure 4-5.- The + X axis braking.



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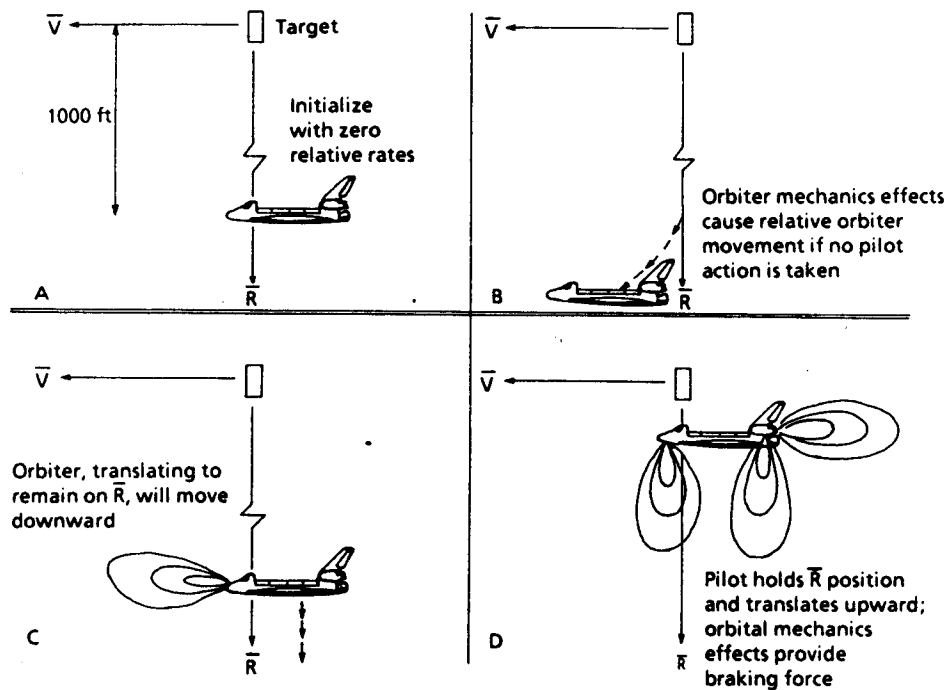
Figure 4-6.- Orthogonal braking.

Utilization of Orbital Mechanics in Braking

A braking action between the Orbiter and a target may be achieved without firing the Orbiter's RCS jet directly toward the target. This can be achieved by utilizing the different orbital rates of the Orbiter and target if their orbits are different sizes. To utilize orbital mechanics effects in this way, the Orbiter will be placed either in an orbit larger than or smaller than the target's orbit. To simplify this discussion, assume that the Orbiter is in an orbit which is smaller than the target's orbit and that both the Orbiter and target are near circular orbits. Also, assume that the Orbiter is near and "behind" the target. As the Orbiter is in a lower orbit than the target, it is "below" the target and is catching up to the target. To enhance target visibility (by the crew and radar), the Orbiter will be tracking the target with its $-Z$ axis. When the Orbiter crosses the target's radius vector to the center of the Earth (R -bar), the relative rates between the Orbiter and target are nulled. This requires a retrograde burn and results in the Orbiter being in a new orbit at less than circular velocity. Thus, the Orbiter will begin to

approach perigee, its speed will increase relative to the target, and it will also begin to fall below the target. (See figure 4-7.) The Orbiter is maintained on R -bar by firing retrograde, which results in the Orbiter moving down the target's R -bar. The approach to the target is then made using radially inward burns ($-Z$) with the necessary $+ or -X$ RCS firings to keep the Orbiter on the target's R -bar. This will be discussed more in the R -bar approach section. If the Orbiter was in a higher orbit than the target, a similar procedure could also be accomplished.

The effective orbital mechanics braking force depends on the difference in orbital size between the target and Orbiter (ΔH), as well as the distance between the two vehicles. As an example, with a 1000-ft ΔH , the resultant acceleration is approximately 0.0033 ft/sec^2 , and at 100 feet, the acceleration is approximately 0.00033 ft/sec^2 . Care must be taken so that the relative range rates and ranges between the target and Orbiter are such that orbital mechanics can compensate for the effective closing forces. Otherwise, RCS firings must be made toward the target to null the relative velocity.



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Figure 4-7.- Orbital mechanics effects in braking.

Maneuver Targeting

The PROX OPS maneuvers (transitions to offset positions, approaches, flyarounds, separations, etc.) can be targeted either visually, using RR, COAS, or closed circuit television (CCTV) data (this is called the "out-the-window" approach), or automatically (using the onboard targeting software). Which technique will be used depends on whether one technique yields greater benefits in terms of accuracy, flexibility in case of errors, RCS fuel savings, crew time savings, etc.

Visual (Out-the-Window) Targeting

To target the PROX OPS maneuvers visually, the crewmember is usually required to maintain the target in a specified relative position with the COAS. Also, it is necessary to establish a specified velocity, R , relative to the target (using the RR) at a specified distance, R , from the target (using the RR). These specified positions and velocities are called braking gates. The required maneuvering is executed using the translational hand controller (THC).

Software-Assisted Targeting

The other method of targeting the PROX OPS maneuvers involves the use of the onboard targeting software. While either Lambert or C-W targeting could be used at this time, C-W targeting generally is faster than the Lambert method, and is accurate enough for use at the range found during the PROX OPS phase. Thus, C-W targeting is usually chosen for the PROX OPS maneuvers. Each of the planned PROX OPS maneuvers is determined preflight and the required parameters (relative positions, base time, TIG times, etc.) are included as part of the original software I-load. In order to compute the ΔV 's necessary for a maneuver, the crewmember calls the ORBIT TGT SPEC (Relative Navigation Display section), inputs a target number associated with the desired maneuver, inputs any necessary parameters (if the I-loaded values are no longer correct) and commands the software to compute the necessary ΔV 's. These numbers are displayed to the crew and are passed to a buffer in guidance. Then, another CRT display, ORBIT MNVR EXEC (the OPS display in GNC OPS 202, Universal Printing display section) is called. With this display, the necessary burn

attitude can be ascertained (typically with the Orbiter's X-axis along the thrust vector) and the Orbiter can automatically be maneuvered to this attitude. The burn can then be executed automatically by the software (OMS) or manually by using the THC (RCS) and burning out the velocity to go (VGO) which is shown on this display.

Use of the onboard software in this way has both advantages and disadvantages as compared with the visual out-the-window mode. It appears that the onboard automatic targeting is more accurate and thus requires less RCS propellant. Also, it would be difficult to execute the orthogonal braking scheme (which minimizes plume impingement and forward pod propellant usage) visually, whereas it is accomplished easily using the onboard software. How flexible it is with respect to pilot error (such as missing a burn) and other problems (such as losing the RR) is still being examined. Therefore, which techniques will be used is a function of the mission rules for that particular flight.

Proximity Operations Maneuvers Descriptions

The PROX OPS phase is composed of several types of maneuvers: transition to offset position (at which stationkeeping may be performed), final approach to the target, (direct, R-bar, V-bar), close-in stationkeeping (inertial, matched rate, flyaround), and a final separation maneuver which transfers the Orbiter to a specified stand off position. The two general headings under which all of these maneuvers come are approach techniques and stationkeeping techniques.

Approach Techniques

An attempt has been made to standardize the techniques used to make an approach to a target (or to a stationkeeping position in close proximity to the target). The methods which have been studied include a direct approach, a V-bar approach (from behind or ahead), and an R-bar approach (from above or below). Each technique has advantages and disadvantages with respect to operations feasibility, propellant consumption, and plume impingement.

Direct Approach

A direct approach is made when the trajectory established at TI (designed to intercept the target at a specified offset position) is followed up to target or offset interception. In general, the final approach is from below and in front of the target (figure 4-8), in order to enhance the lighting conditions at intercept. As these final maneuvers are usually made shortly after entering the daylight pass, this technique reduces the necessity of the crew to look into the Sun.

In the direct approach technique, it is necessary to reduce the ΔV between the target and the Orbiter, using a series of braking thrusts according to a predefined table of acceptable Orbiter/target ranges and range rates. Also, sufficient closing velocity must be maintained to ensure an intercept. This makes it necessary to null a significant relative ΔV close to the target, which, of course, leads to plume impingement concerns. In general, the

normal + Z PRCS will be used for braking until it is within the target's C/O sphere and then the low-Z ($\pm X$) braking mode could be used to null the final relative velocities while alleviating the plume impingement problem. Recall, however, that the low-Z mode is very RCS fuel expensive, and it does not provide a very great deceleration. This type of approach can be made either visually or by using the onboard software targeting.

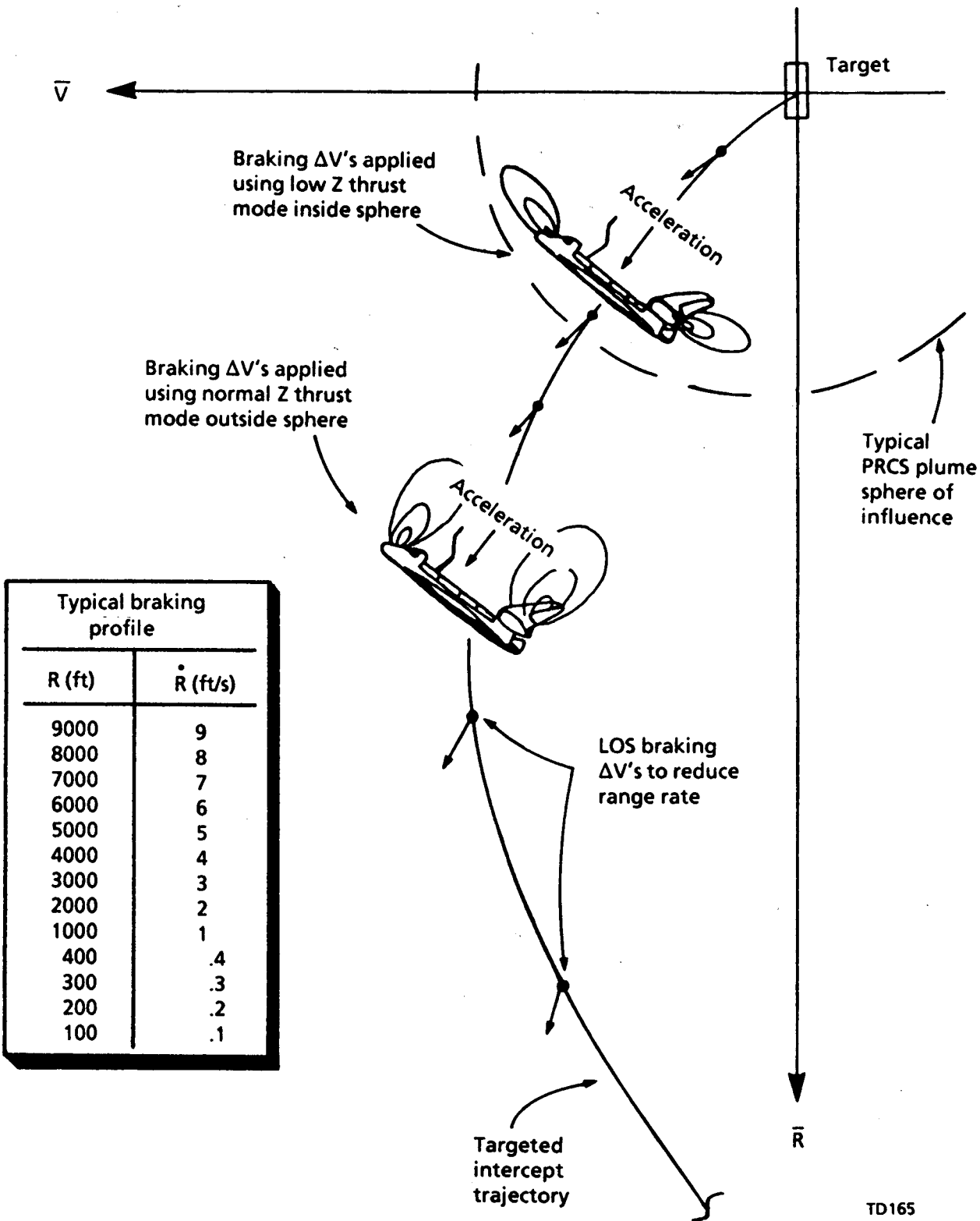


Figure 4-8.- Direct approach to close-in stationkeeping.

V-Bar Approach

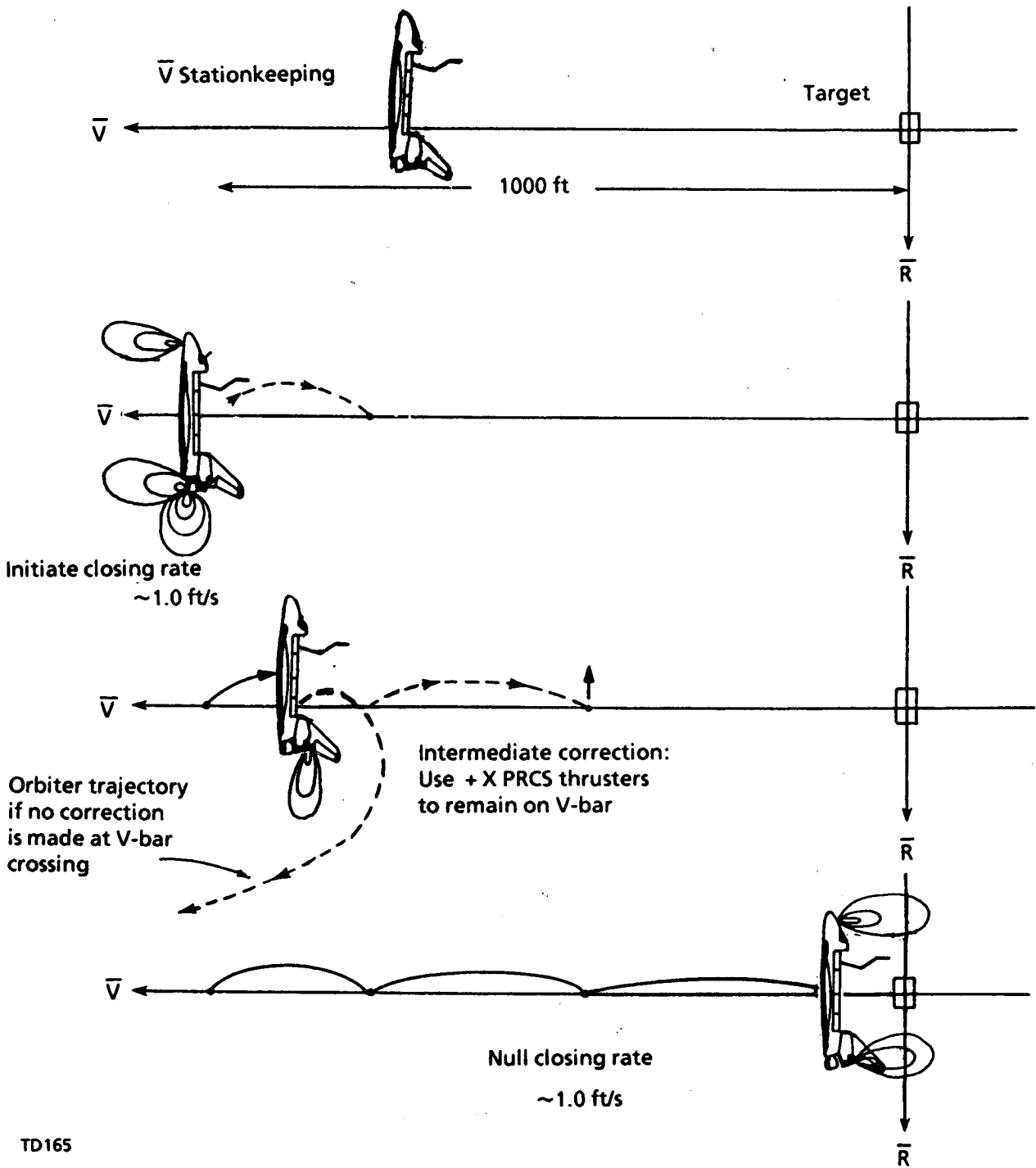
A V-bar approach is made after the Orbiter is transitioned to a position on the target's velocity vector (either ahead of or behind the target) and the relative rates are nulled. It is possible for the Orbiter to stationkeep in this configuration indefinitely, as the Orbiter is now in the same orbit as the target. Typically, the Orbiter will be configured with its $-Z$ axis pointed toward the target to ensure maximum target visibility for the crew and RR. This is done by maneuvering the Orbiter (via the UNIV PTG display or manually) such that its $-Z$ axis is pointing at the target and then placing the Orbiter in LVLH attitude hold (through the panel C3/A6 LVLH PBI or by tracking the center of the earth using the UNIV PTG display).

This type of approach is initiated by establishing an initial closing rate toward the target (approximately 1 ft/sec at 1000 feet). Assume first that the Orbiter is ahead of the target, as shown in figure 4-9. The initial closing rate would be established using the $-Z$ thrusters (a retrograde burn). If no other compensating thrust was made, such a thrust would result in the Orbiter moving initially toward and below the target. Thus, the initial burn should be a combination retrograde and radially outward burn (using the $+X$ and $-Z$ RCS thrusters) so that the Orbiter initially moves toward the target and upward. The $+X$ thrust (upward) will offset the initial drop in the Orbiter's orbit due to the retrograde thrust. This motion can be monitored through the COAS and with the RR. Eventually, orbital mechanics effects, as a result of the retrograde burn, will begin to force the Orbiter to fall back to the V-bar. If no further radial correction ($+X$ thrust) is made as the Orbiter approaches the V-bar, it will continue to fall below and then ahead of the target as it goes into a lower orbit. Thus, each time the Orbiter crosses the V-bar (as detected by the RR or COAS), the $+X$ RCS jets must be fired normal to the target LOS to provide a radially upward thrust. The magnitude of the necessary thrusts would be a function of the relative range and range rate between the target and Orbiter.

Depending on the initial closing rate, it may be necessary to null the final closing rate, using small braking gates. The final braking maneuver will reestablish a position on the V-bar. Thus, the Orbiter will be in a stable stationkeeping position with respect to the

target. If the final maneuver is within the target's C/O sphere, the low $-Z$ mode (\pm XPRCS) will be used to null the relative velocities. If the Orbiter were behind the target, a reverse procedure would be followed, involving a small posigrade thrust with intermediate radially inward ($-X$) thrusting to maintain the V-bar.

A V-bar approach can be done visually or using the PROX OPS targeting software.



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Figure 4-9.- V-bar approach.

+ V-Bar Approach Procedure

Initial conditions:

- Orbiter is 1000 ft from target on the +V-bar.
 - Orbiter is in a stable stationkeeping position with the -Z axis pointing at the target (figure 4-10).
 - -Z sense switch is selected. (This switch determines the control mode of the AFT RHC and THC, e.g., THC in moves the Orbiter in -Z direction and RHC back pitches -Z axis up.)
- A. Configure the DAP as shown in figure 4-11.
- Normal (primary) jets must be used to translate because the vernier jets cannot.

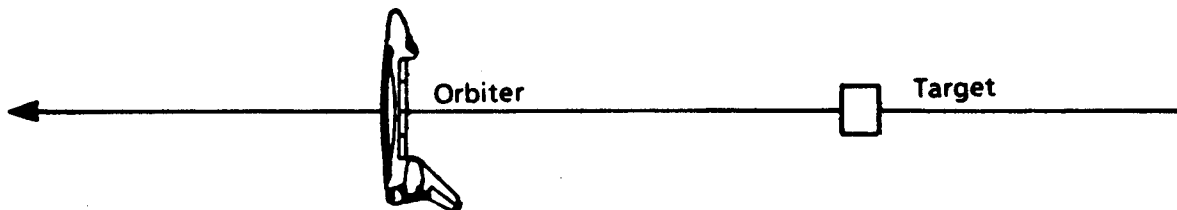
- When not translating, select vernier jets. These jets will be used for attitude control. They provide better fuel efficiency and tighter attitude control.

- DAP A will be used from 1000 to 200 feet and will have a larger pulse size entered for normal jets than DAP B (e.g.; 0.1 ft/sec, 0.05 ft/sec).

B. Call up the relative navigation (REL NAV) display (GNC SPEC 33). Perform the following:

- ITEM 2 EXEC zeros ΔV_x , ΔV_y , ΔV_z velocity component registers
- ITEM 3 EXEC zeros the ΔV_{tot} velocity register.

These registers will be used to determine the rate of the initial thrust maneuver toward the target and the total velocity gained during the V-bar approach.



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Figure 4-10.- V-bar stationkeeping.

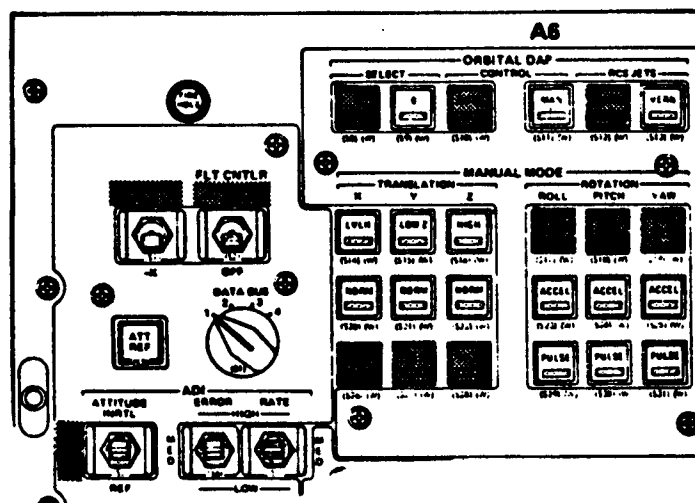


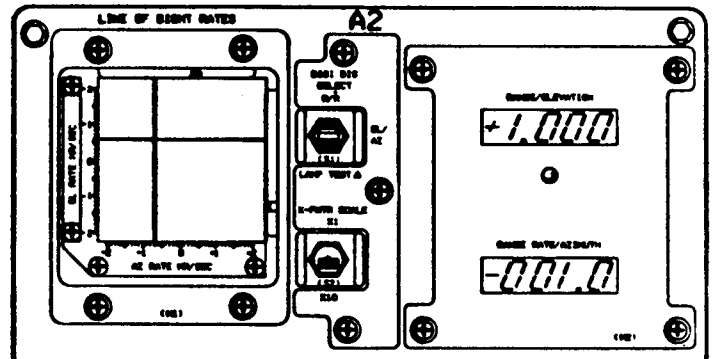
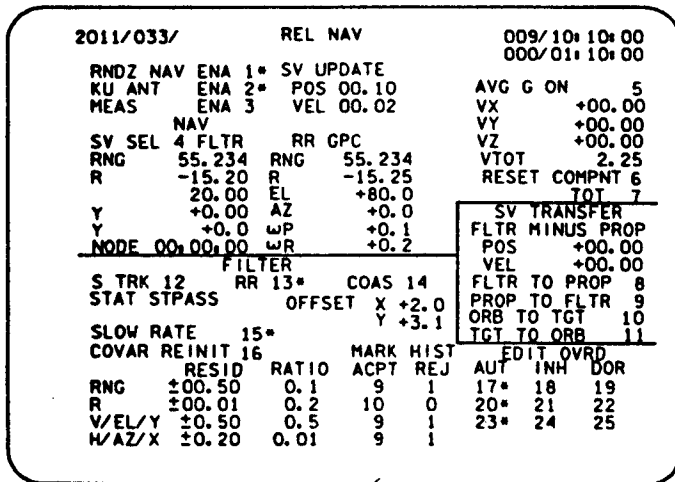
Figure 4-11.- Panel A6, DAP.

C. Begin the approach.

- Command -Z pulses (THC in) to attain a closure rate.
- Read the closure rate or on the rendezvous radar dedicated light emitting diode (LED) display on panel A2 (figure 4-12). The rate will be negative on the LED.
- A good rule of thumb is to use a closure rate equal to 0.1 percent of your distance as displayed by the radar range rate LED.
 $(1,000 \text{ ft} \times 0.001) = 1 \text{ ft/sec}$ initial closure rate.

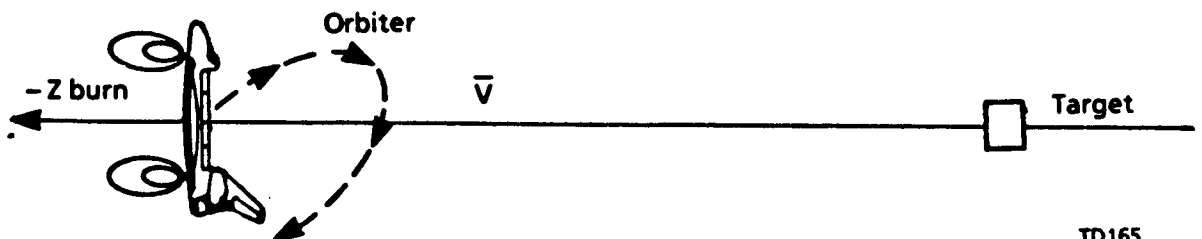
The initial motion of the Orbiter will be towards the target and above the velocity vector (figure 4-13). The reason for the upward motion is that the -Z thrust has a +X small component (cross-coupling).

It is important to note that the initial upward motion is only temporary. Remember that the -Z thrust is a retrograde burn and the long term effect is that the Orbiter falls below the target and then moves ahead of it.



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Figure 4-12.- REL NAV display and panel A2.



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Figure 4-13.- -Z burn to the target.

D. When the Orbiter starts to fall below the V-bar: command a + X thrust to drive the Orbiter center of gravity (CG) back towards the V-bar (figure 4-14). This process is continued until the Orbiter is close to the target.

To prevent the CG from falling any farther, a + X thrust is commanded. Remember to select normal jets when using the THC. After performing this, vernier jets may be reselected for finer attitude control.

To judge when the center of gravity (CG) of the Orbiter falls below the V-bar, the - Z COAS will be used (figure 4-15). When the target reaches the horizontal line on the reticle the Orbiter's CG will be below the V-bar. The reason for this is that the CG is close to 45 feet aft of the - Z COAS.

The target will move toward the bottom of the reticle and may even move out of the bottom. This is not critical because orbital mechanics will bring the Orbiter back toward the V-bar and thus the target back into the reticle.

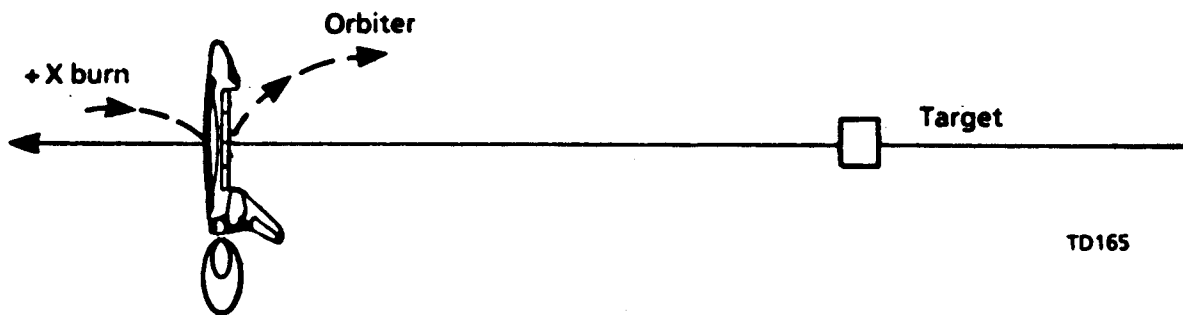


Figure 4-14.- + X burn to the target.

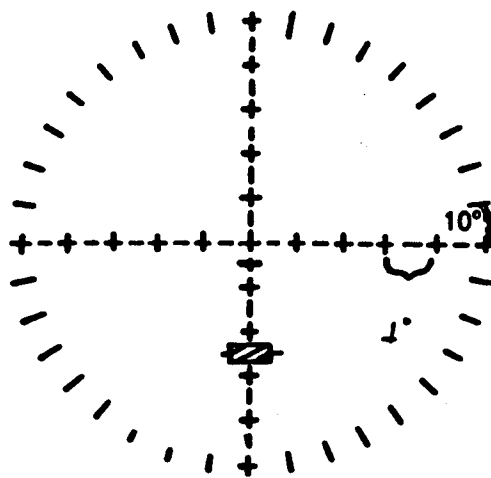


Figure 4-15.- COAS reticle.

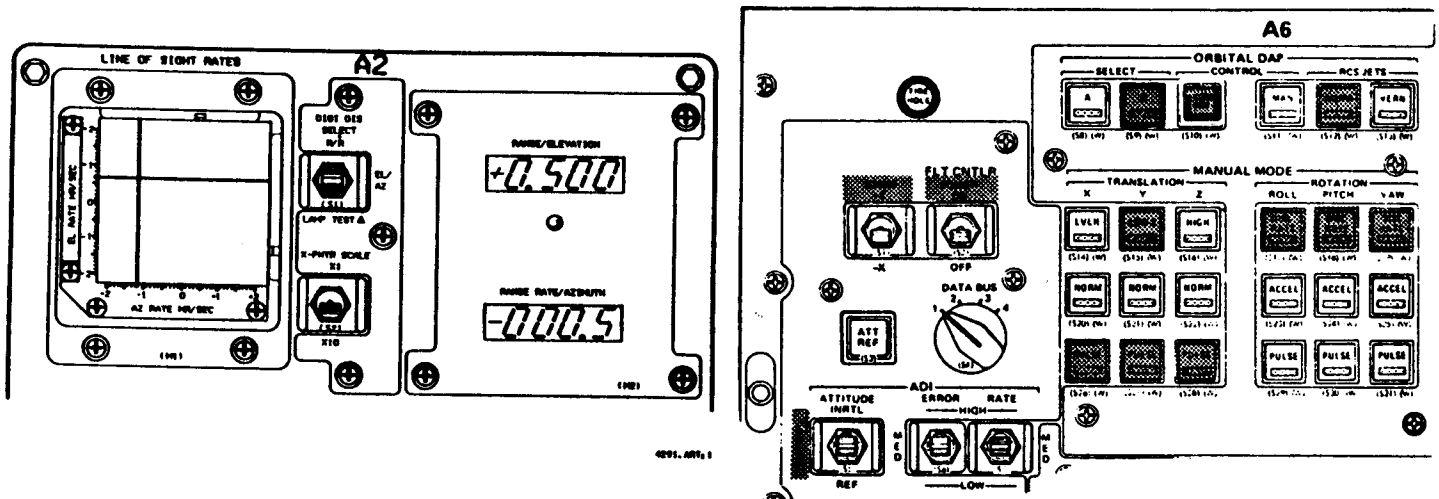
As the range from the target decreases, the closing rate should be decreased accordingly (e.g., 0.7 ft/sec at 700 feet, 0.5 ft/sec at 500 feet). The + Z inputs (THC out) may be required to decrease the closure rate.

- E. At 200 ft select DAP B and low Z mode (figure 4-16).
- Low Z will limit plume impingement on the target because there is no direct thrusting in the direction of the target.
 - Low Z does not aid in fuel economy, however. It uses fuel at a rate of 281 lb/ft/sec versus 30 lb/ft/sec for normal Z in a 180 000-pound Orbiter.
 - Remember that because DAP B and low Z are being used, more THC inputs will be needed to translate. For this reason it is important to keep the closure rate under control. At 200 ft the rate should be 0.2 ft/sec or less.

The slow rate will prevent things from happening too quickly.

- There is a point where RR will no longer supply range data (e.g.; 100 ft). Then CCTV camera angles, ranging rulers, or the laser ranging device can be used for range/range rate.

- F. Command + Z pulses (THC out) to zero the closure rates when within an acceptable range to grapple the target.



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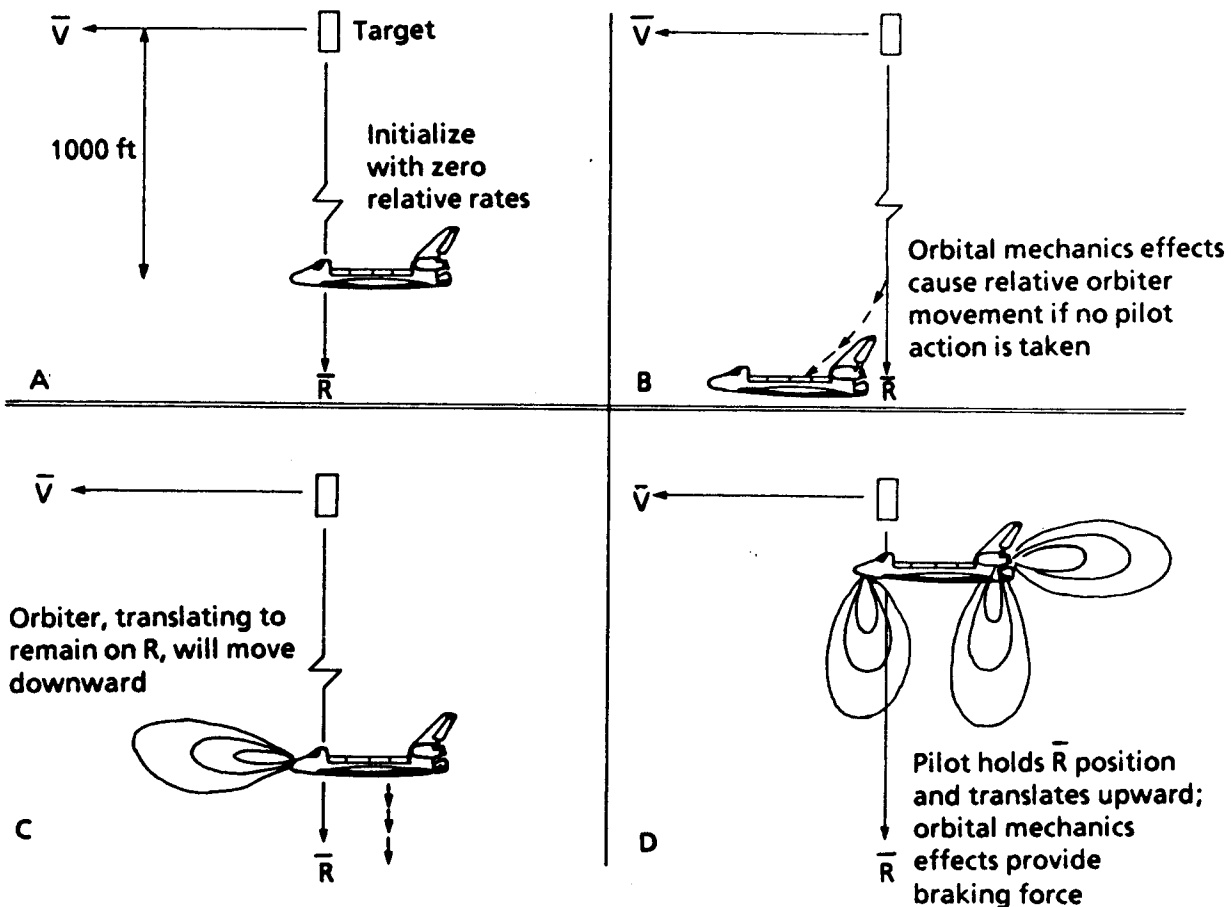
Figure 4-16.- Panels A2 and A6.

R-Bar Approach

An R-bar approach is initiated with the Orbiter on the target's radius vector to the center of the Earth (R-bar) and with the target/Orbiter relative rates nulled. An initial closing rate toward the target is established, and orbital mechanics is relied upon to provide a braking force.

As an example, assume the Orbiter is 1000 feet below the target on R-bar, with its - Z axis pointed at the target, as shown in figure 4-17, and the target is in a near circular orbit. Assume also that the relative rates between them are first nulled. As the relative rates between the target and Orbiter are nulled, orbital mechanics will result in the Orbiter moving below and ahead of the target if no action was taken. Thus, a retrograde burn must be made to keep the Orbiter on R-bar.

Such a burn results in the Orbiter "falling down" the R-bar. An initial closing rate can then be established, in opposition to the orbital mechanics effects, by thrusting upward. In order to keep the Orbiter on R-bar, the + X thrusters must also be fired so that the applied radial burn does not cause the Orbiter to go behind the target. The result of this is that the Orbiter "hops" up the R-bar, with orbital mechanics providing the braking force instead of the RCS jets. This has obvious advantages in terms of plume impingement. The R-bar approach will consist of several of these "hops." In order to avoid collision, the RCS jets may be used in the low Z mode to null any residual ΔV experienced within the target's C/O sphere. Once again, a series of braking gates will be identified which will specify the necessary closing rate at each R-bar crossing (dependent upon the relative range at the crossing). Also, this type of approach can be made either visually or software assisted.



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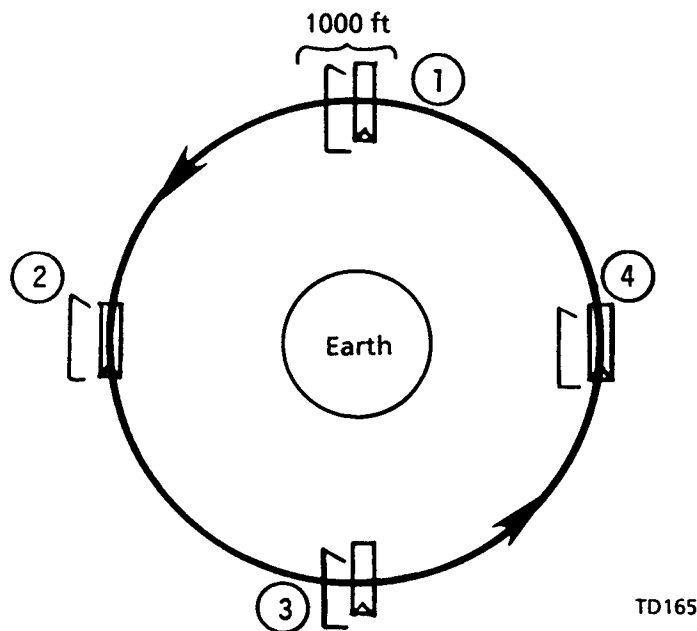
Figure 4-17.- R-bar approach.

Stationkeeping Techniques: Close-In

The stationkeeping techniques used during the close-in PROX OPS mission phase are designed to maintain the Orbiter at desired relative position, attitude, and attitude rate with respect to the target. The relative position, typically, is at a constant range from the target, with the COAS line of sight (LOS) passing through the target's center of mass or some other specified point on the target. The range used would depend on such factors as target plume sensitivity, size, etc. The Orbiter is maintained at this range by a crewmember using RR data and by maintaining the target centered in the COAS. For most operations, the COAS will be mounted in the overhead window and thus will be pointing along the Orbiter's -Z axis. The relative vehicle position can then be adjusted manually with RCS thrusts as required, using the THC. The Orbiter's attitude and attitude rate are maintained automatically with universal pointing software and the digital autopilot (DAP).

The Orbiter's attitude rate may be selected to match the target's attitude rate or to satisfy some other mission objective, such as in flyarounds and transitions (both discussed later). The target is typically in inertial attitude hold or in LVLH attitude hold. If the Orbiter is stationkeeping with a body vector, such as its -X axis, pointed toward the target, and its attitude rate matching that of the target, it will constantly be pointing at the same side of the target.

As an example, suppose the target is in inertial attitude hold (such as solar inertial) and the Orbiter is on its V-bar, 1000 feet from the target. The Orbiter can be maneuvered such that its -Z axis is pointed toward the target, and then the Orbiter can be placed in inertial attitude hold. The crewmember must then thrust with the THC (which does not affect Orbiter attitude) to maintain the 1000 feet relative range and to keep the target centered in the -Z COAS. This is called inertial stationkeeping (figure 4-18).

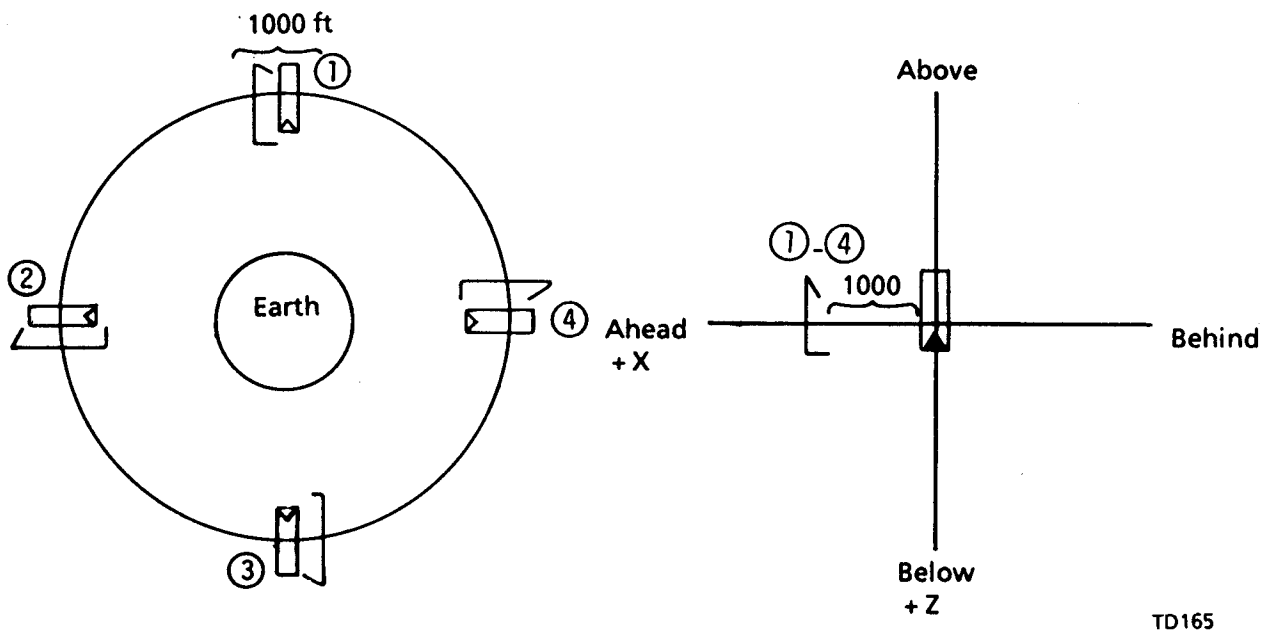


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Figure 4-18.—Inertial stationkeeping.

As another example, suppose the target is rotating at orbital rate; i.e., LVLH attitude hold (such as gravity gradient) and the Orbiter is on its positive V-bar, 1000 feet from the target. The Orbiter can then be maneuvered such that its -Z axis is pointing toward the target (with a specified omicron, such as 0° or 180°) and can then be placed in LVLH attitude hold via an

item entry on the UNIV PTG display or via the orbital DAP pushbutton on panel C3. As before, the crewmember may be required to thrust, using the THC, to maintain the 1000-ft relative range and to keep the target centered in the -Z COAS. This kind of stationkeeping with the Orbiter constantly on the target's V-bar is, logically enough, called V-bar stationkeeping (figure 4-19).



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Figure 4-19.- V-bar stationkeeping.

+ V-Bar Stationkeeping Procedure

Initial conditions:

- Orbiter is 1000 ft from the target on the + V-bar (figure 4-20)
- - Z axis is pointing toward the target
- - Z sense switch is selected

- A. Maintain a - XLV, YPOP attitude (-X axis pointing at the center of the Earth with the Y-axis perpendicular to the orbit plane). This attitude can be maintained automatically using the DAP and universal pointing display. To accomplish this, set up the DAP and universal pointing display as shown in figure 4-21.

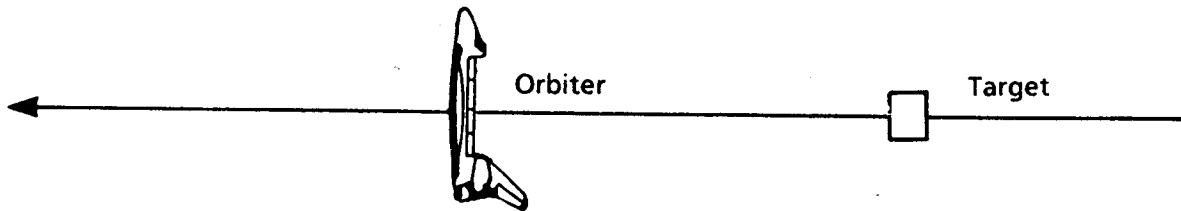
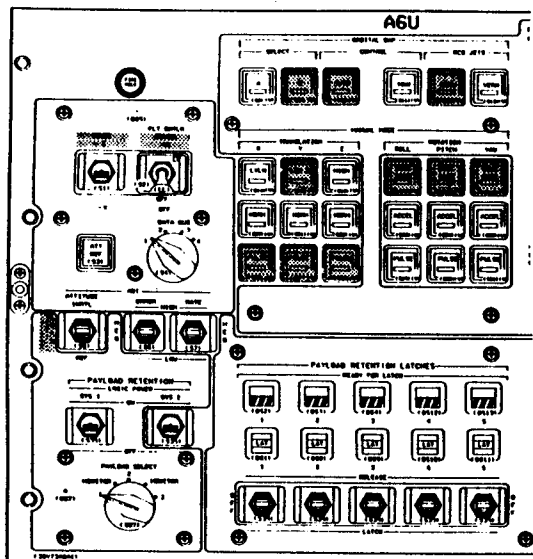


Figure 4-20.- V-bar stationkeeping at 1000 feet.



```

2011/ / UNIV PTG 1 001/10: 15: 00
CUR MNVR COMPL 10: 17: 45 000/ 10: 15: 00
1 START TIME 002/ 10: 14: 00

MNVR OPTION START MNVR CUR FUT
5 R 180.00 TRK 19
6 P 045.00 ROT 20
7 Y 000.00 CNCL 21

TRK/ROT OPTIONS ATT MON
8 TGT ID 002 22 MON AXIS 1 +X
ERR TOT 23*
ERR DAP 24

9 RA 000.000
10 DEC 000.000
11 LAT 000.000
12 LON 000.000
13 ALT 00000.0

14 BODY VECT 1

15 P 000.00
16 Y 000.00
17 OM 000.00

CUR ROLL PITCH YAW
REQD 180.00 000.00 000.00
ERR +000.00 +045.00 000.00
RATE +00.000 +00.200 +00.000

LDG SITE UPDATE
25 PRI 1 KSC15
26 SEC 2 KSC33
27 ALT 3 NOR17
    
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Figure 4-21.- Panel A6 and the UNIV PTG display.

Selecting auto on the DAP and executing items 8, 14, 17, and 19 on the universal pointing display will establish automatic attitude control. The - X axis (item 14: Body Vector 2) will auto track (item 19) the center of the Earth (item 8: TGT ID 2) with an omicron of 0° (item 17). The ADI will read $180^\circ/0^\circ/180^\circ$ in pitch, yaw, and roll.

Remember to select vernier jets. The automatic attitude control will simplify the stationkeeping task while vernier jets will aid fuel economy.

- B. Aim for an "ideal" position for the target in the - Z COAS reticle.

This position will place the Orbiter's CG on the V-bar of the target's with zero relative rates. At 1000 feet the target should be 2.5° below the crosshairs to place the CG on the V-bar, the reason being that the CG lies some distance aft of the - Z COAS (figure 4-22).

- C. Translate using the THC to maintain this position. To conserve fuel, small inputs are essential. Therefore, a good technique is to use the translational pulse mode with as small a pulse size as practical. Remember to use normal jets for this.

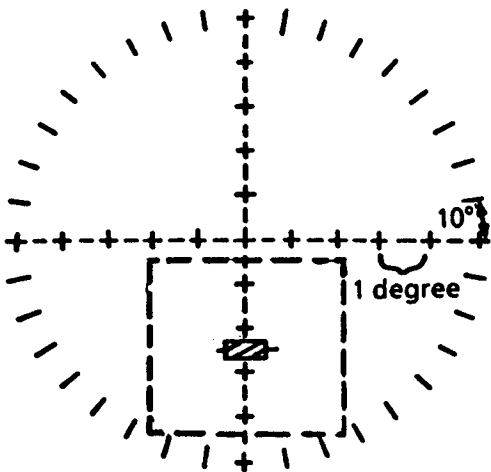


Figure 4-22.- COAS reticle, - Z.

Keeping the Orbiter exactly at the $- 2.5^\circ$ point will require constant THC inputs and greater fuel usage. A good technique is to maintain an imaginary box about the ideal position (e.g.; $- 2.5^\circ \pm 2^\circ$). Translate back toward the "ideal" position only when the target moves out of the box. This will keep THC inputs to a minimum and therefore decrease fuel usage.

Remember, when not translating to keep the target in the box, use vernier jets to maintain attitude.

- D. Maintain the 1000-ft distance from the target using the THC and the rendezvous radar range and range rate data.

There can be a tendency to center all attention on keeping the target in the COAS. Remember to keep an eye on the range rate and command THC inputs when needed to ensure zero rates (figure 4-23).

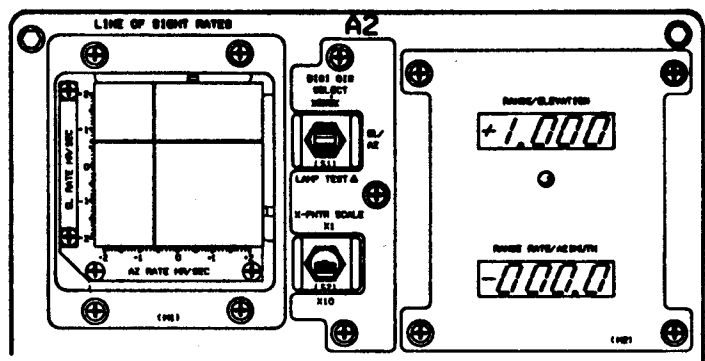


Figure 4-23.- Panel A2 (zero range rate, 1000-ft range).

Flyarounds

A flyaround is a close-in stationkeeping technique in which the Orbiter points a specified body vector at the target and inertially flies around the target within a specified amount of time, while maintaining a constant relative range. This technique is used quite close in to a target, from 35 to several hundred feet, and can be used to inspect a target. The range at which it is performed would depend on such factors as plume impingement, Orbiter control authority, and fuel usage. The technique essentially involves a continuous Orbiter pitch (either up or down, depending on the direction of rotation around the target) as the Orbiter circles the target. The rotational rate which must be induced on the Orbiter is dependent on the desired flyaround time, direction of pitch (tail up or tail down), and fraction of the orbit, angle θ , to be traveled. It also depends on the inertial rate of the target. For this discussion, the target will be assumed to be either inertially fixed or in LVLH attitude hold. Note that knowledge of the orbital rate of the Orbiter/target makes it possible to deduce the desired flyaround time if one knows the required angle of orbit travel and vice versa. For example, in a 150-n. mi. circular orbit, orbital rate is approximately 4 deg/m or 0.067 deg/sec. Thus, to say that the Orbiter flies around the target within 60° of orbit travel is equivalent to saying that the Orbiter will fly around the target in 15 minutes.

Figure 4-24 shows a 180° pitch (tail up) flyaround with an inertially fixed target in a 150-n. mi. circular orbit which takes

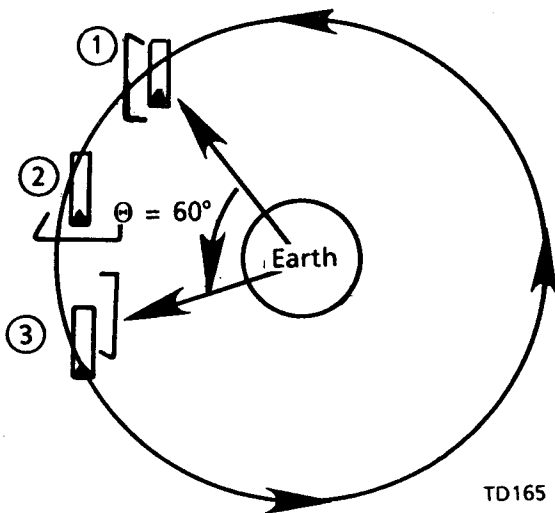
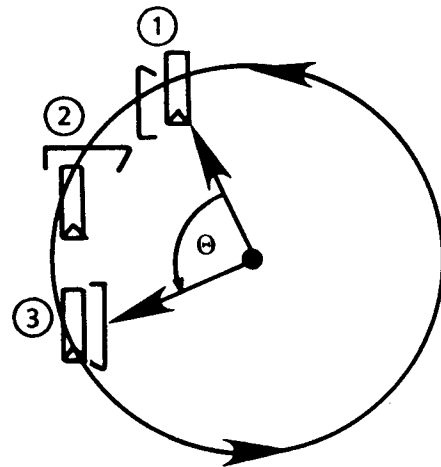


Figure 4-24.- 180° pitch (tail up) flyaround of 15-minute duration.

15 minutes. The 180° refers to the angle the Orbiter rotates inertially about the target.

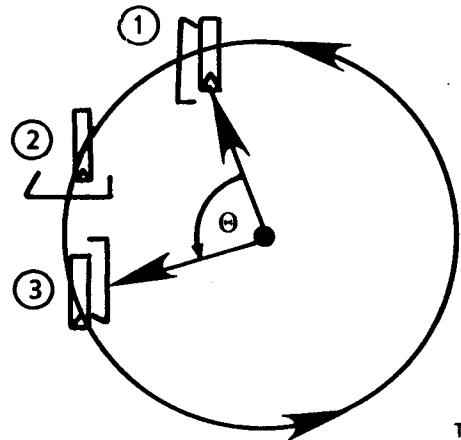
For a flyaround, it is necessary for the crew to make whatever corrective thrusts with the THC that are necessary to maintain a constant range to the target and keep the target centered in the COAS.

Suppose, first, that the target is in inertial attitude hold, the Orbiter's X and Z axes lie in the orbit plane, and the Orbiter's -Z axis is pointing toward the target with an omicron of 0° at a range of 100 feet. Also, suppose that it is necessary to fly 180° around the target. This flyaround could be accomplished one of two ways: by pitching tail up or down about the Orbiter's Y axis at a constant rotational rate, or by simultaneously translating as required (figures 4-25 and 4-26).



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Figure 4-25.- Pitch (tail down) flyaround.



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Figure 4-26.- Pitch (tail up) flyaround.

Given the desired flyaround time, t , to make a 180° flyaround with an inertially fixed target, it is possible to determine the necessary inertial rotational rate for the Orbiter, using a chart such as in figure 4-27, or with the equation

$$\text{rotational rate} = \pm \frac{180^\circ}{t}$$

where the rate is positive for a pitch (tail up) rotation and negative for a pitch (tail down) rotation. In the example given with figure 4-27 where $t = 15$ min, the Orbiter must maintain a

$$\frac{180^\circ}{15 \text{ min}} = \frac{180^\circ}{900 \text{ sec}} = 0.2 \text{ deg/sec pitch (tail up) rate}$$

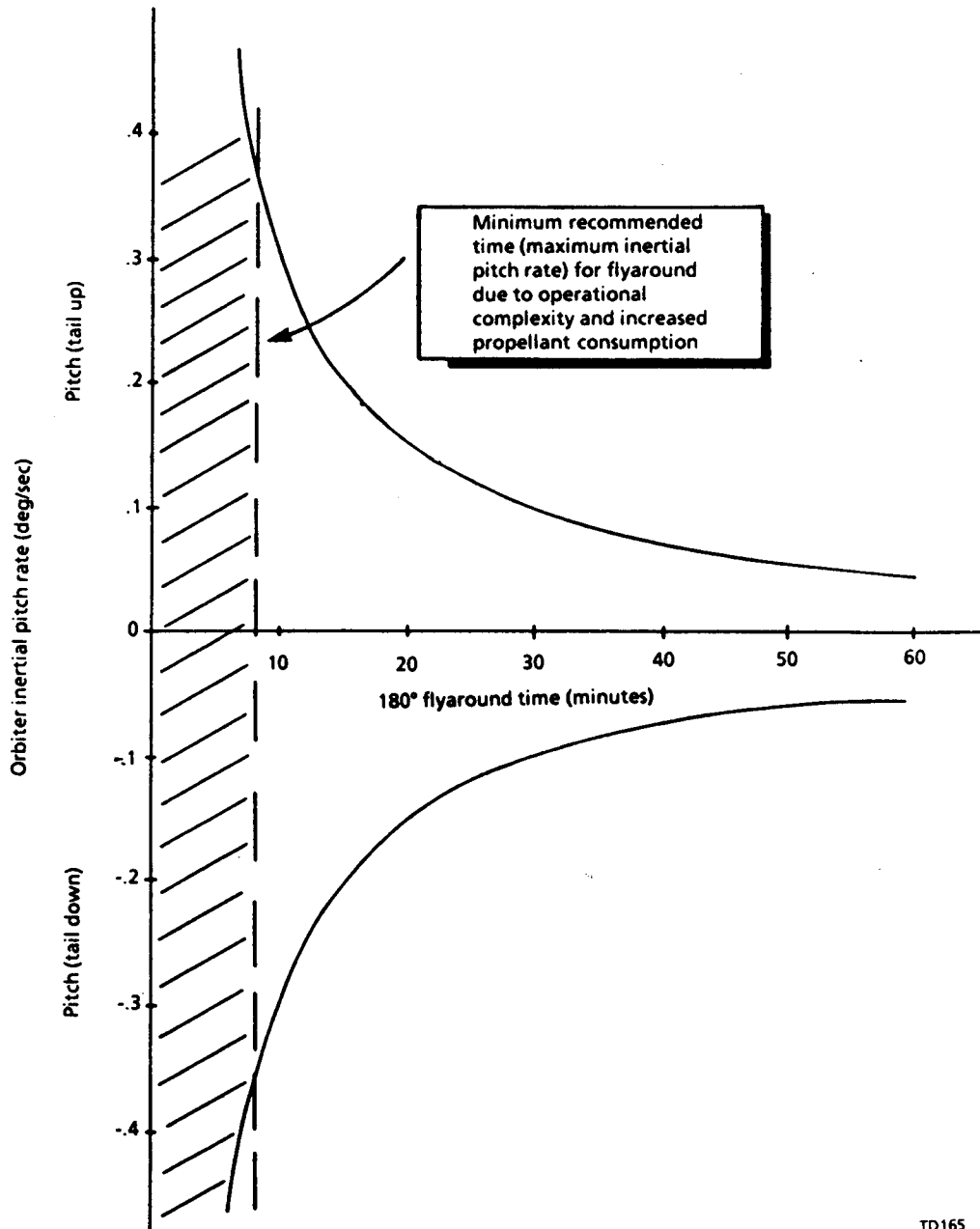


Figure 4-27.— 180° In-plane flyaround time versus Orbiter inertial pitch rate for an inertial attitude hold target.

Next, suppose the target is in LVLH attitude hold (such as being gravity gradient stabilized). Then the target is itself rotating at orbital rate in inertial space. Recall that for a 150 n. mi. orbit, orbital rate is approximately 4 deg/m or 0.067 deg/sec. This means that if the Orbiter maintains inertial attitude hold with the target kept at a fixed range and centered in the COAS using the THC, then the Orbiter will fly around the target after one revolution. In other words, the Orbiter will fly around the target at 0.067 deg/sec (see figure 4-28).

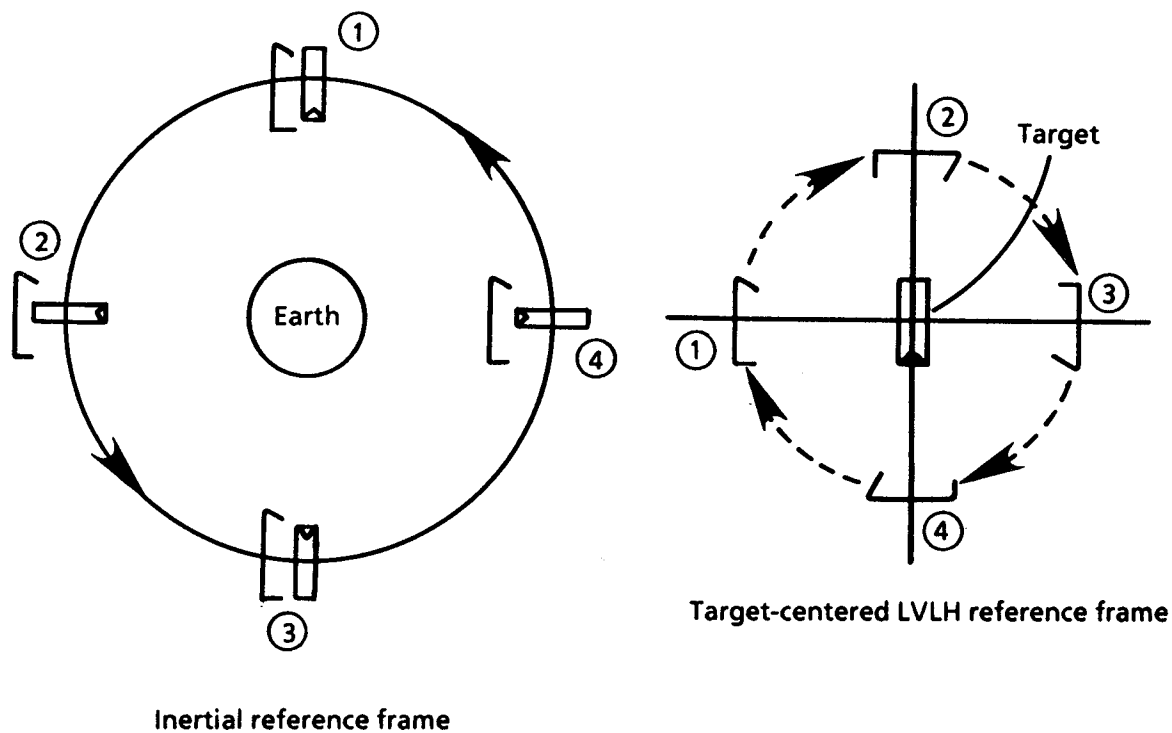
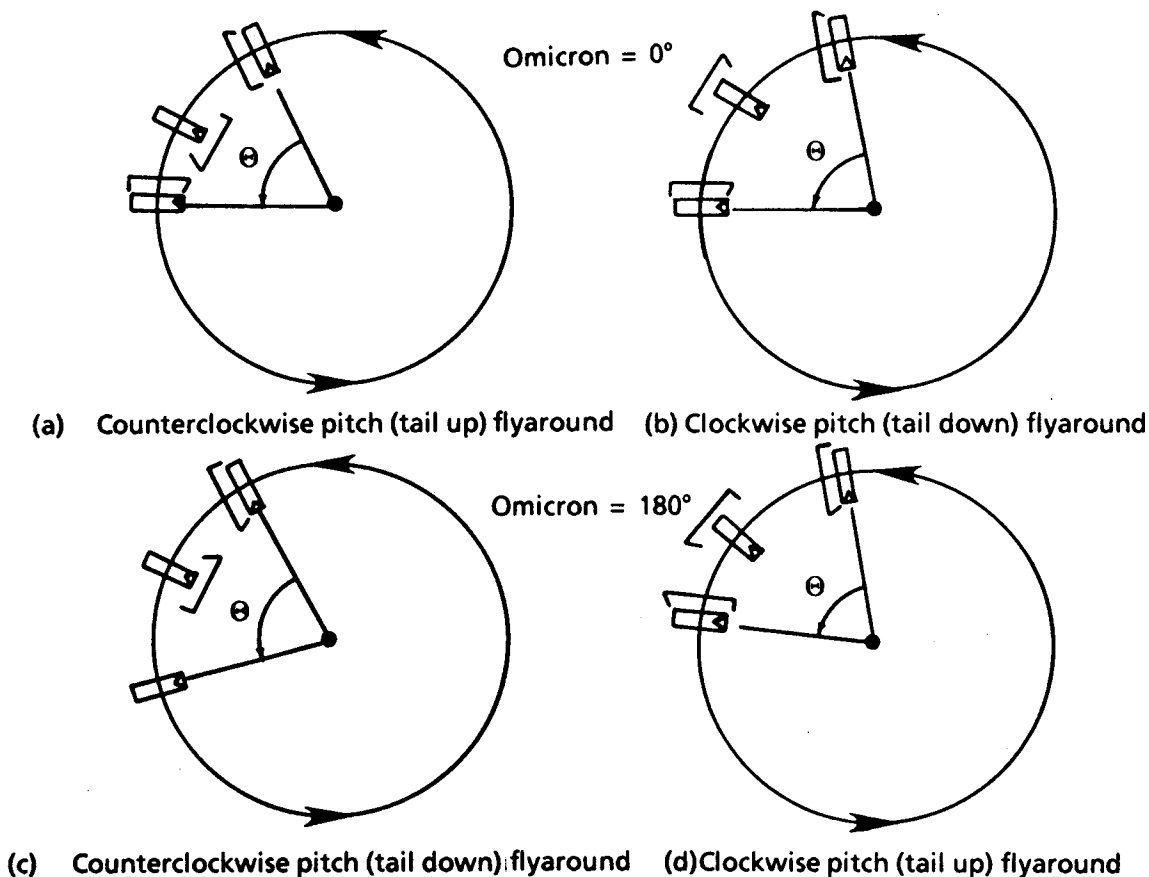


Figure 4-28.- Flyaround of an LVLH attitude hold target.

A target in a circular orbit rotating at orbital rate means that it inertially rotates θ degrees within θ degrees of orbit travel about its negative angular momentum vector. Note that the statement that the target is in LVLH attitude holds means that it is rotating at orbital rate. For our discussion, and for the figures, the direction of orbit travel will be counterclockwise. Thus, the amount of time necessary for a flyaround will depend on the rotation rate of the Orbiter, and the direction (clockwise or counterclockwise) in which this rotation is occurring. Also, the relationship

of the concept of counterclockwise and clockwise rotation to Orbiter rotation depends on the omicron of the Orbiter. If the Orbiter has an omicron of 0° and its X and Z axes are in the orbit plane, a counterclockwise rotation means a pitch (tail up) rotation about its Y axis, whereas a clockwise rotation means a pitch (tail down) rotation. On the other hand, if the Orbiter has a omicron of 180° , a counterclockwise rotation means a pitch (tail down) rotation, and a clockwise rotation means a pitch (tail up) rotation. These ideas are illustrated in figure 4-29.



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Figure 4-29.— Flyaround examples with target rotating at orbital rate.

Refer to figure 4-29 for the following cases.

Suppose, again, that the target and Orbiter are in a 150-n. mi. circular orbit, with the target in LVLH attitude hold and the Orbiter positioned ahead of the target on V-bar with its X and -Z axes lying in the orbit plane, its -Z axis pointed at the target, and an omicron of 0°. Suppose that it is desired to have the Orbiter complete a 180° flyaround "over" the target; i.e. clockwise pitch (tail down), in 10 minutes (600 seconds).

To determine the necessary pitch rate for the Orbiter, it should first be noticed that the target will rotate approximately 40° counterclockwise in 10 minutes ($\theta = 40^\circ$). Thus, the Orbiter needs only to rotate 140° pitch (tail down) in that 10 minutes, such that a total of 180° will be circumnavigated over the target. Its rotation rate is thus

$$\frac{140^\circ}{10 \text{ min}} \text{ or } \frac{140^\circ}{600 \text{ sec}} = -0.233 \text{ deg/sec}$$

The minus sign is for a pitch (tail down) rotation. More generally, if it is desired to have the Orbiter flyaround 180° over an LVLH target with a pitch (tail down) maneuver, in time t, the necessary Orbiter rotation rate is

$$-\left[\frac{180^\circ - (0.067)t}{t} \right] =$$

$$-\left[\frac{180^\circ}{t} - 0.067 \text{ deg/sec} \right] (\text{omicron} = 0^\circ)$$

On the other hand, if it were desired to do a 180° flyaround under the target, it would require a pitch (tail up), counterclockwise rotation by the Orbiter. Once again, suppose this flyaround is to be completed in 10 minutes. Then the target will already have rotated 40° counterclockwise, and thus, the Orbiter must rotate $180^\circ + 40^\circ = 220^\circ$ counterclockwise in order to circumnavigate under and around the target. Thus, the Orbiter's rotation rate is

$$\frac{220^\circ}{10 \text{ min}} = \frac{220^\circ}{600 \text{ sec}} = 0.366 \text{ deg/sec}$$

More generally, if it is desired to have the Orbiter flyaround 180° under an LVLH target using a pitch (tail up) maneuver in time (t) in seconds, and the Orbiter's omicron is 0°, the necessary Orbiter rotation rate is

$$-\left[\frac{180^\circ + (0.067)t}{t} \right] =$$

$$-\left[\frac{180^\circ}{t} + 0.067 \text{ deg/sec} \right] (\text{omicron} = 0^\circ)$$

The opposite situation occurs in the case of an Orbiter omicron of 180°. While the ideas involved are the same, the pitch (tail up) and pitch (tail down) maneuvers are reversed. Thus, the general formulas for 180° flyarounds in t seconds of an LVLH target with an Orbiter omicron of 180° are as follows.

Orbiter rotation to fly counterclockwise about a target pitch (tail down):

$$-\left[\frac{180^\circ + (0.067)t}{t} \right] =$$

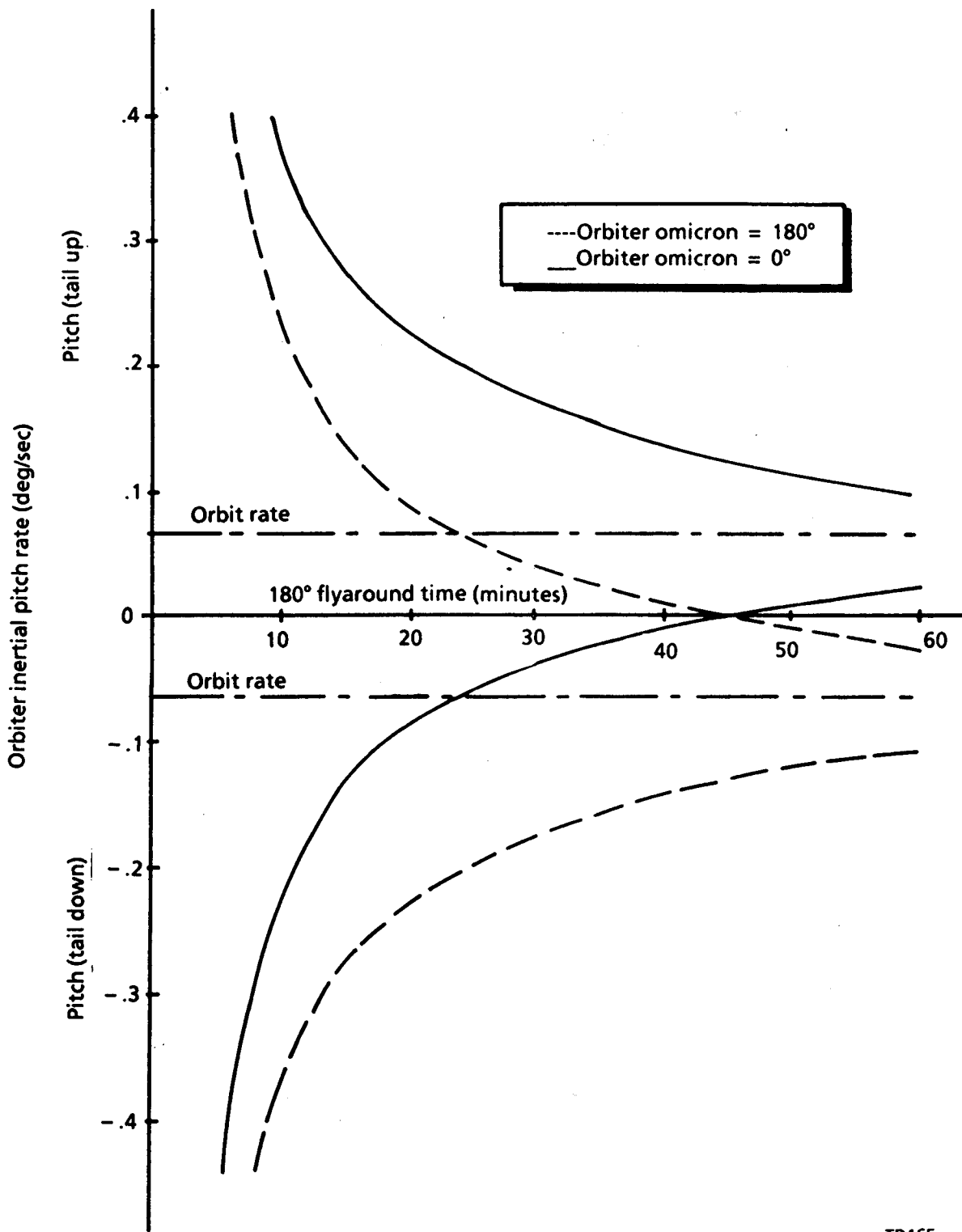
$$-\left[\frac{180^\circ}{t} + 0.067 \text{ deg/sec} \right] (\text{omicron} = 180^\circ)$$

Orbiter rotation rate to fly clockwise about a target pitch (tail up):

$$\left[\frac{180^\circ - (0.067)t}{t} \right] =$$

$$\left[\frac{180^\circ}{t} - 0.067 \text{ deg/sec} \right] (\text{omicron} = 180^\circ)$$

The above general equations are shown graphically in figure 4-30.



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Figure 4-30.- 180° In-plane flyaround time versus Orbiter inertial pitch rate for an LVLH attitude hold target.

Flyaround Procedure

Initial conditions:

- Orbiter is 200 feet from the target on the +V-bar
- Orbiter is in a stable stationkeeping position with the -Z axis pointing at the target and the -X axis pointing at the center of the Earth (figure 4-31).

- A. Before beginning the flyaround, select a rotational rate for the Orbiter. This rate is dependent on the length of time desired to do the flyaround. Suppose it is desired to fly 180° around an inertially stabilized target in 15 minutes. Then:

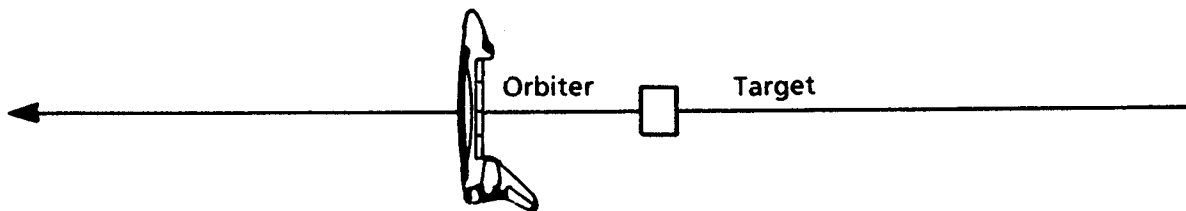
$$\text{Rotational Rate} = \frac{180^\circ}{t} =$$

$$\frac{180^\circ}{900 \text{ sec}} = \frac{0.2^\circ \text{ pitch}}{\text{sec}}$$

(The flyaround can be accomplished one of two ways)

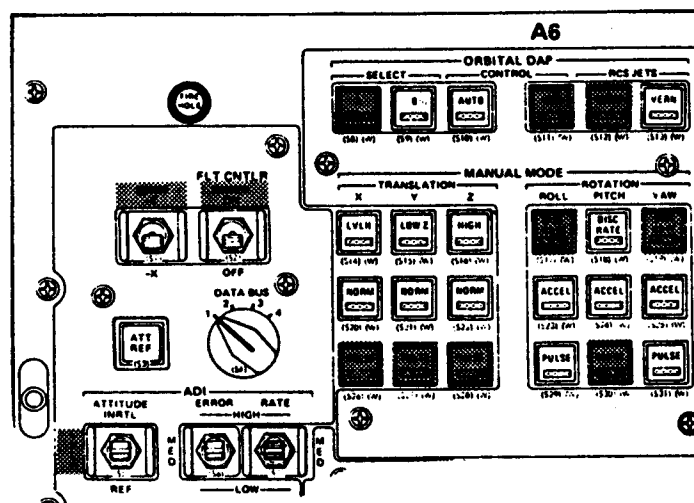
- B. Manual flyaround: Configure the DAP as shown in figure 4-32.

The DAP will now hold attitude in roll and yaw and allow a rotation in pitch.



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Figure 4-31.- V-bar stationkeeping at 200 feet.



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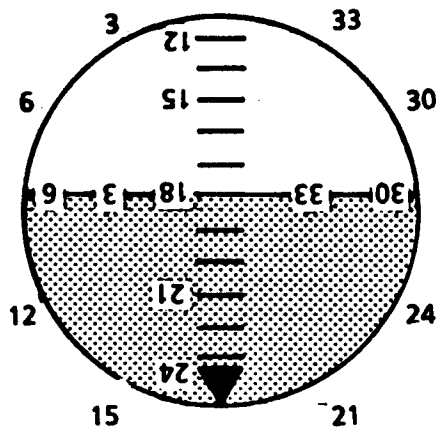
Figure 4-32.- Panel A6 (manual flyaround).

B(1) Begin the flyaround (figure 4-33).

Command A -0.2 deg/sec pitch (tail down) rate using the RHC. The rate is positive for a pitch (tail up) rotation, and negative for a pitch (tail down) rotation (figure 4-34).

The initial motion of the target in the COAS reticle once a pitch (tail down) rotation begins is movement up the center line (figure 4-35).

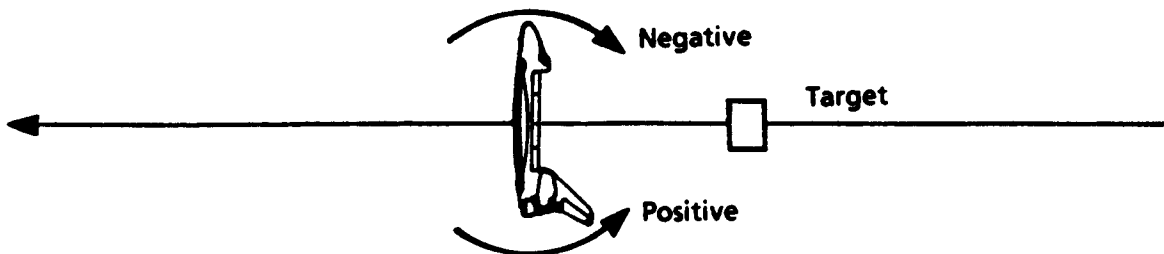
B(2) Command A + X thrust (THC UP) to drive the target back to the center of the COAS reticle. This process of keeping the target centered is continued until the aft ADI ball reflects a 180° pitch change (figure 4-36).



LVLH $\frac{\text{aft ADI } (-Z)}{(180/0/180)}$
P/Y/R

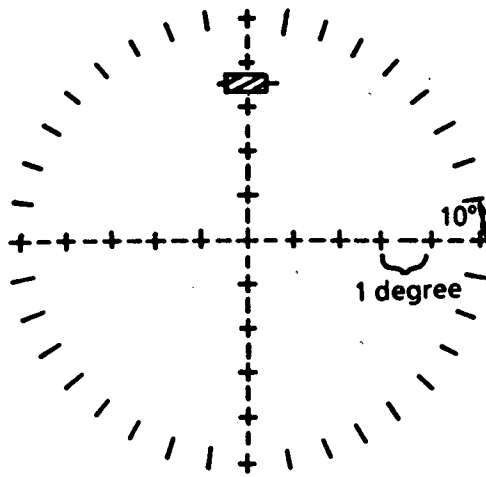
TD165

Figure 4-33.- ADI ball at start of flyaround.



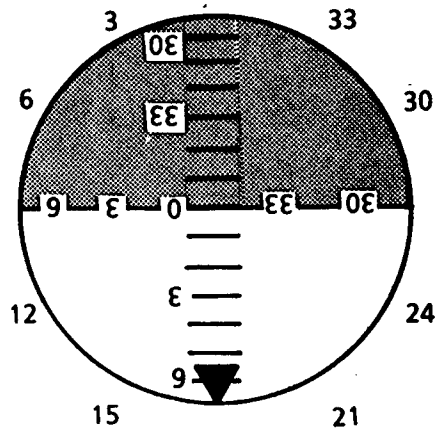
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Figure 4-34.- Orbiter pitch sense in -Z.



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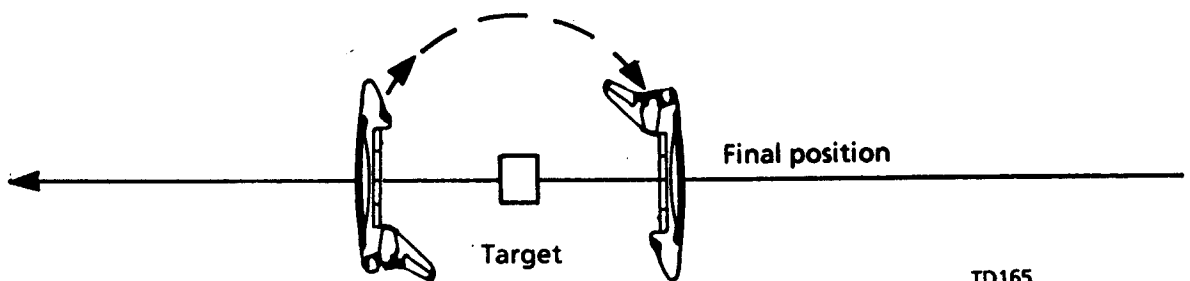
Figure 4-35.- COAS reticle (pitchdown rotation).



Aft ADI (-Z)

LVLH (0/0/180)

P/Y/R



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Figure 4-36.- Flyaround path.

Throughout the maneuver, the RR dedicated light emitting diode (LED) display on panel A2 (figure 4-37) should be monitored and the THC should be used to keep the closure rate on the target at zero.

C. Automatic flyaround: Configure the DAP as shown in figure 4-38.

Call up DAP CONFIG display spec 20 (figure 4-38), and enter ITEM 2 + 0.2 EXEC. This will set up a 0.2 deg/sec discrete rate for the normal jets in DAP A.

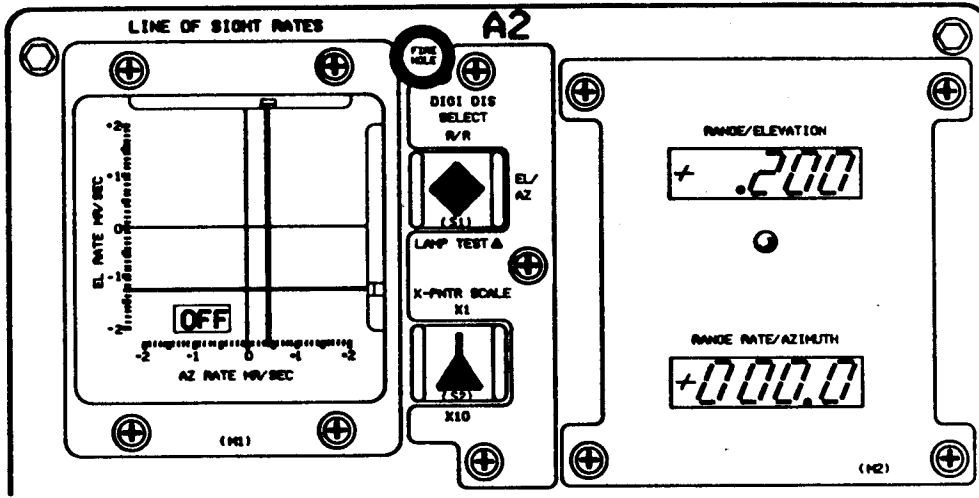


Figure 4-37.- Panel A2.

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XXXX/O20/XXX	DAP CONFIG		XX X	DDD/HH:MM:SS	DDD/HH:MM:SS
TRANSLATION PULSE	A	B			
	1	0.05 15	0.01		
ROTATION DISC RATE	NORM	2 0.20 16	0.15		
	VERN	3 0.200 17	0.200		
PULSE	NORM	4 0.05 18	0.05		
	VERN	5 0.010 19	0.01		
COMP	NORM	6 .00 20	.00		
	VERN	7 .000 21	.000		
DEADBAND ATT	NORM	8 2.00 22	2.00		
	VERN	9 .500 23	.500		
RATE	NORM	10 0.20 24	0.20		
	VERN	11 .010 25	.010		
JET OPT	P	12 1 26	1		
	Y	13 1 27	1		
CNTL ACCEL		14 0 28	0		

INERTIAS

29 IX 0.874

30 IY 6.800

31 IZ 7.070

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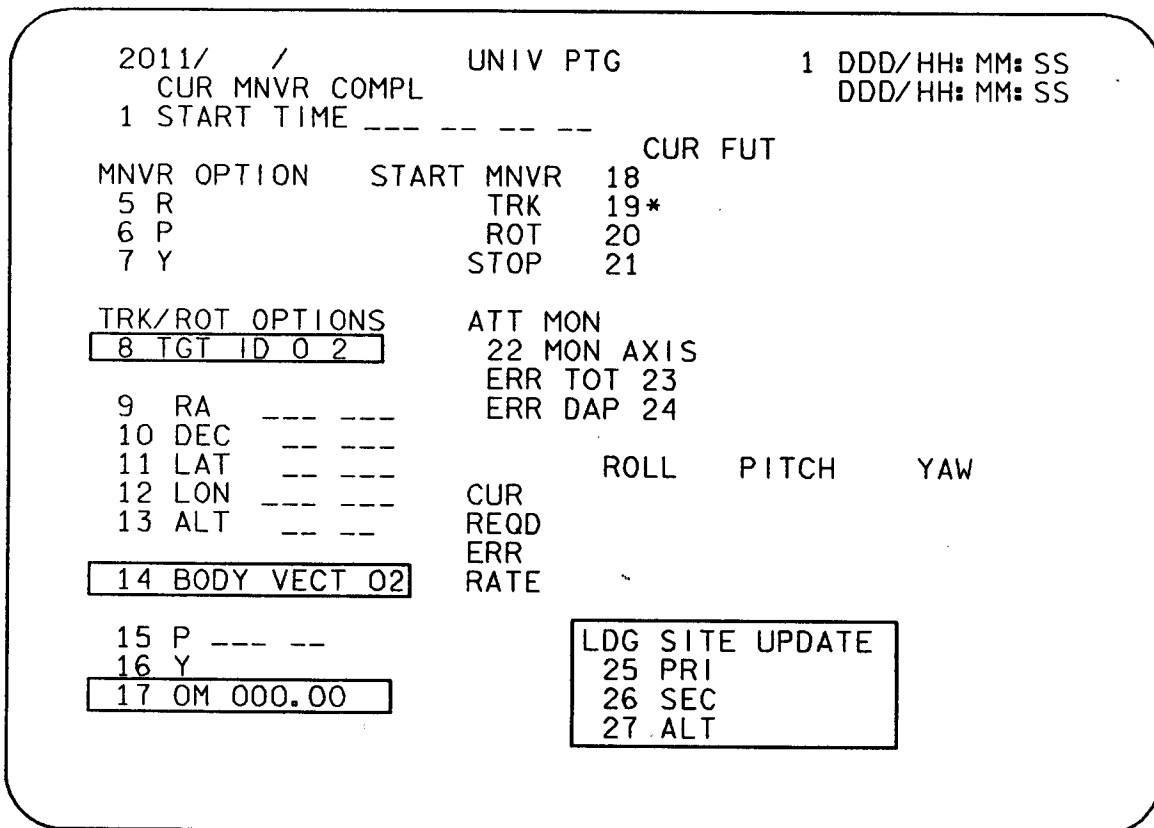
Figure 4-38.- SPEC 20 (automatic flyaround) and DAP configuration.

C(1) Begin the flyaround: Use the UNIV PTG display and start a rotation about the Y-axis as shown in figure 4-39: Items 14 & 5 Exec allows the crew to select the axis they wish to rotate about, item 16 + 90 selects the right wing, which will allow a taildown rotation, and item 20 Exec initiates the rotation.

Once the rotation begins, the THC will have to be used to keep the target centered in the COAS reticle.

C(2) When the flyaround is complete, the rotation should be stopped and V-bar stationkeeping set up (see next page).

This accomplishes the same thing as manually pitching the Orbiter to achieve a 0.2 deg/sec rate. The benefit comes in a steady rotation and reduced crew workload during the flyaround.



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Figure 4-39. - UNIV PTG display.

C(2a) Stop the flyaround in manual. Configure the DAP as shown in figure 4-40.

With the DISC RATE selected in all three rotation axes and with the LVLH mode selected, the attitude that existed at the time of selection of LVLH will be held fixed within the LVLH frame.

As soon as the rotation rate stops, the target will immediately begin moving down in the COAS. The translation rate introduced at the start of the maneuver must now be canceled using the THC.

Once all rates are canceled, vernier jets are selected to conserve fuel while holding attitude.

C(2b) Stop flyaround in auto. Use the universal pointing display to stop the rotation.

STOP ITEM 21 EXEC

The Orbiter will now hold current inertial attitude but will not be held fixed within the LVLH frame.

Null out the translation rate with the THC to keep the target centered.

The universal pointing display can be used to track the center of the Earth or the DAP LVLH pushbutton indicator (pbi) can be selected to put the Orbiter in an LVLH frame.

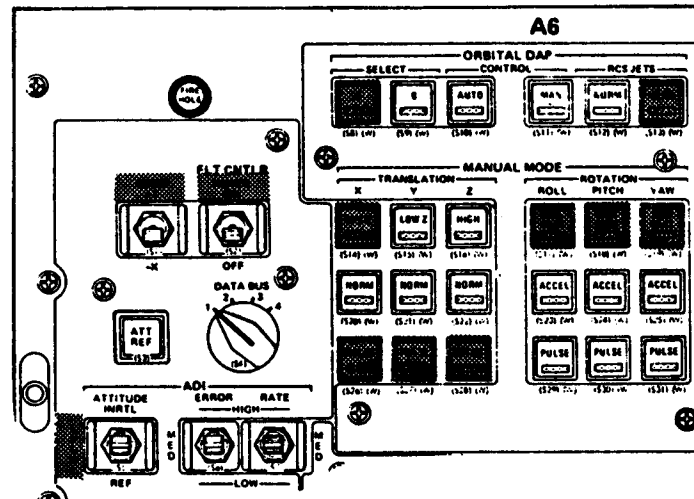


Figure 4-40.- Panel A6 (manual stopping).

Transitions

A transition is a maneuver which maintains the Orbiter at a constant distance from the target as it moves from one position, relative to the target, to another. Also, the Orbiter will continue pointing a specified body vector toward the target throughout the maneuver. Primarily, it is used to move from the final stationkeeping position following TPF to a position from which the final approach may be made. For example, the final braking maneuvers (TPF) of the rendezvous sequence may leave the Orbiter on the target's R-bar at the edge of the target's C/O sphere. If a V-bar

approach is planned, a transition will be made to the target's V-bar, while maintaining a constant range to the target (on the edge of the C/O sphere), and a body vector, such as the $-Z$ axis, will be pointed toward the target. In contrast to a flyaround, which requires that the Orbiter actually fly around the target in an inertial sense, a transition involves moving the Orbiter around the target in a target centered rotating LVLH reference frame without regard to the target's inertial attitude (see figure 4-41). Note also, that a transition can be done one of two ways relative to the target-centered LVLH reference frame: clockwise or counterclockwise.

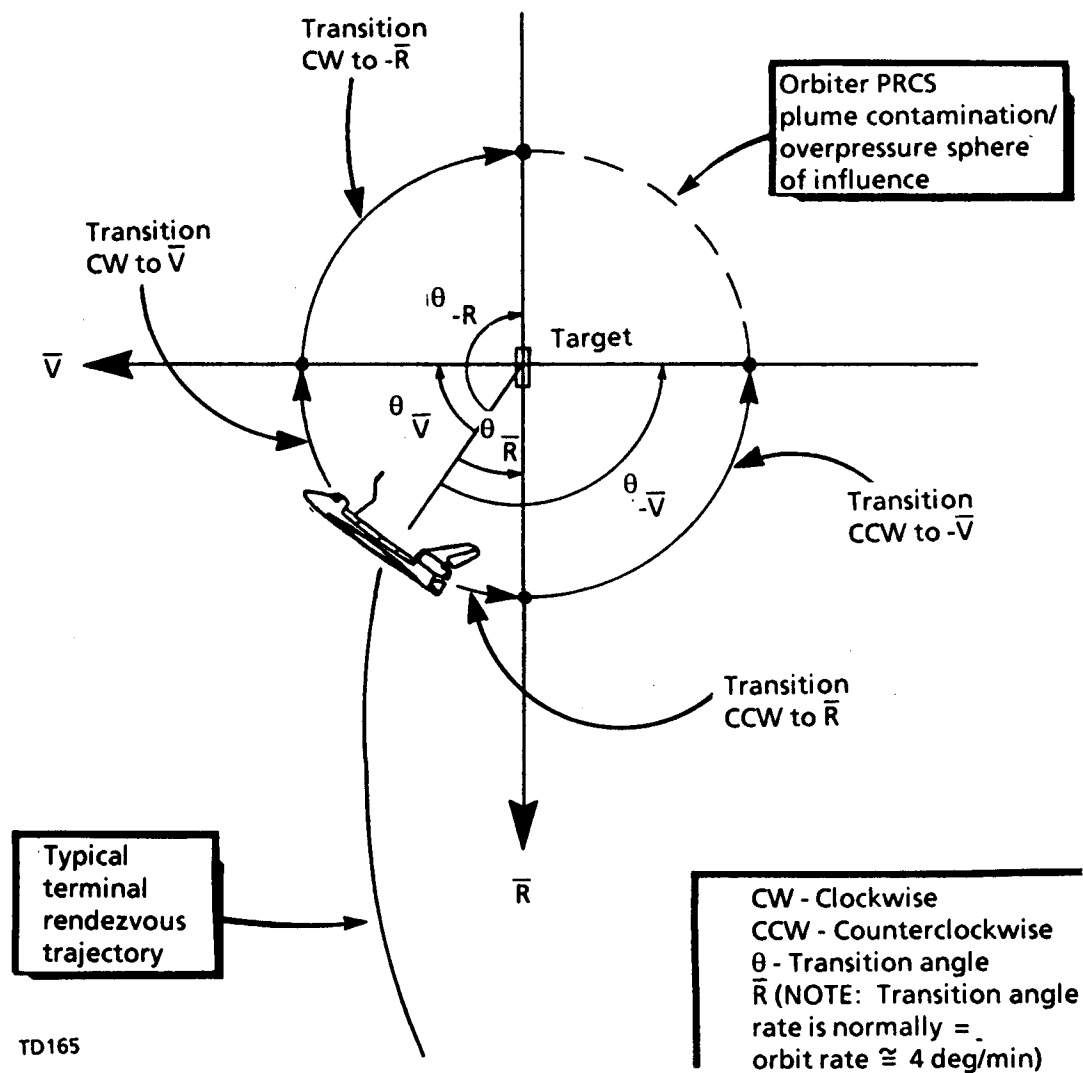
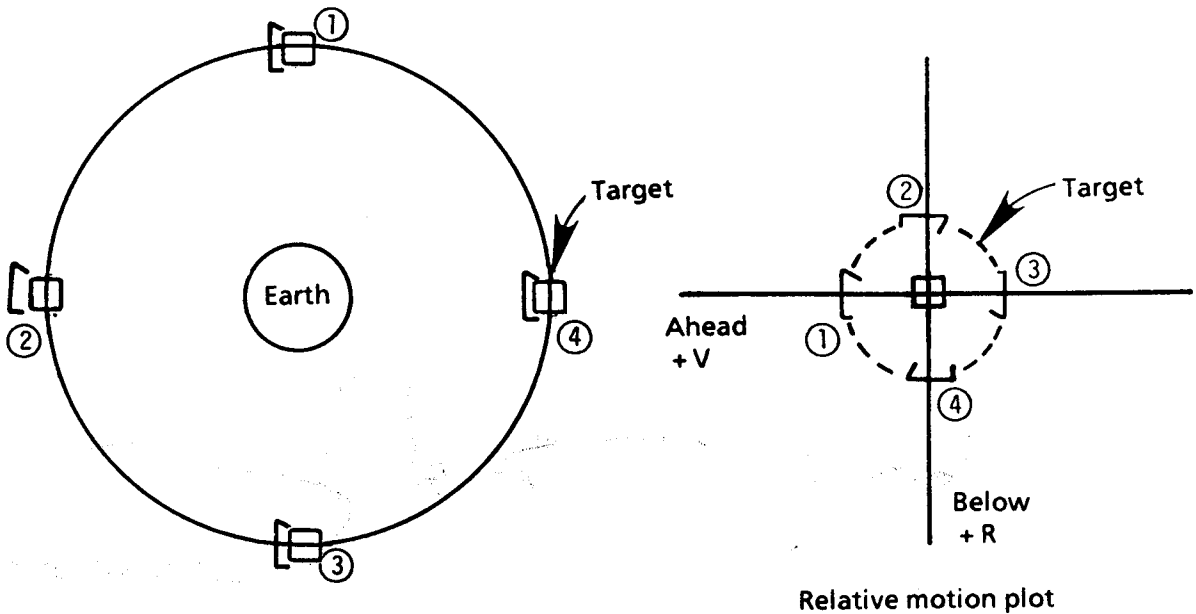


Figure 4-41.- Transition phase provides positioning for final approach.

The most fuel-economical way is to place the Orbiter in inertial attitude hold with its -Z axis pointed toward the target at the required distance (to avoid plume impingement) resulting in a clockwise transition. This requires that thrust corrections with the THC be made to maintain the desired range and to keep the target centered in the COAS. Recall that this is a form of inertial stationkeeping and that the resulting motion is as shown in figure 4-42 if allowed to continue for one full period.

The transition rate is thus orbital rate (~4 deg/min at 150 n. mi.). A transition usually will involve only a subset of the full period, such as from position 4 to position 1 in figure 4-42, and can be described in terms of the initial position and the angle traveled (θ). To find the time required for the transition, it is necessary simply to divide the angle, θ , by orbital rate. Thus, to transition from the +R-bar to +V-bar clockwise requires

$$\frac{90^\circ}{4 \text{ deg/min}} = 22.5 \text{ min}$$



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Figure 4-42.- Clockwise transition.

A transition can also be made by pitching the Orbiter (tail up or tail down). (Refer to figure 4-43 for a pitch tail down transition.) However, this technique requires more fuel since a rotational pitch rate is required. The universal pointing display can be used to establish and maintain the required pitch rotation rate. Also, the target must be kept centered in the COAS and at a constant range using the THC.

Flying a transition from the V-bar to the R-bar or vice versa in 90° of orbit travel is equivalent to a 180° in-plane flyaround about an inertial attitude hold target.

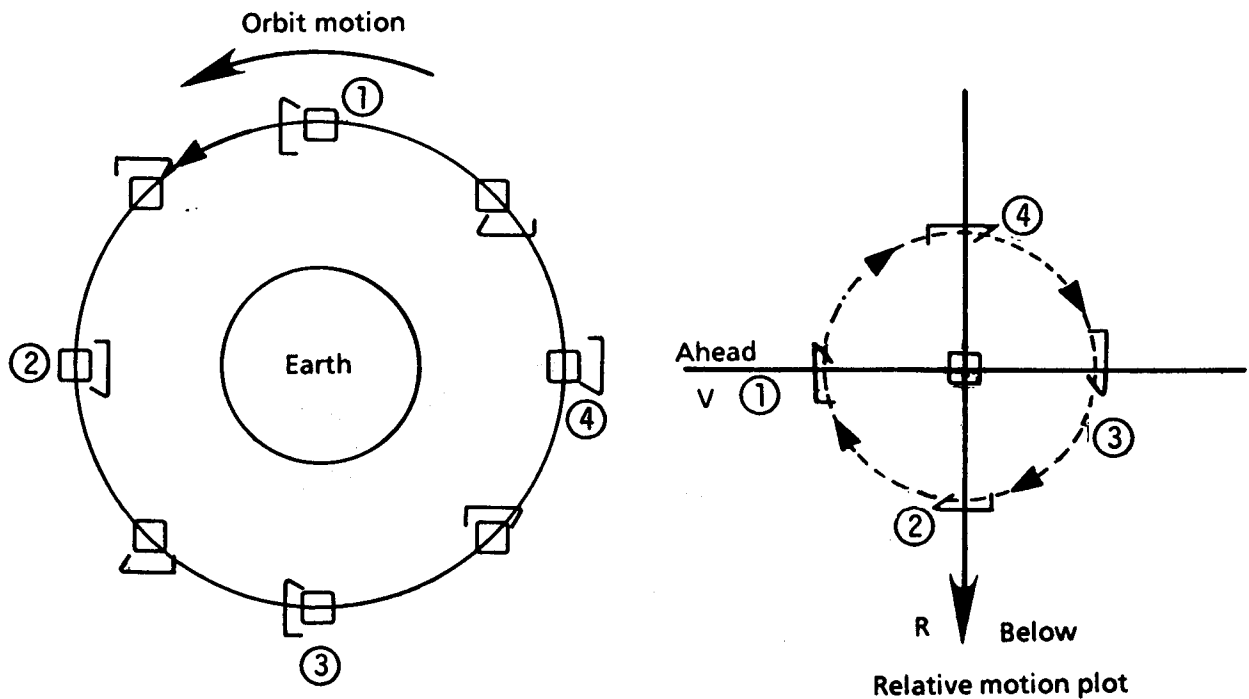
For example, to travel from position 2 in figure 4-43 to position 3, the transition angle

equals 90° or a time of 22.5 minutes. The required rotational rate equals 0.13 deg/sec or twice orbital rate (figure 4-27).

This rate holds true whether you fly clockwise about the target to get to position 3 or fly counterclockwise.

The difference between the two is the increased distance necessary to fly clockwise about the target and, in turn, the increased thrusting as opposed to flying counterclockwise.

Whether an inertial transition or a rotational transition would be done depends on fuel concerns, in which case an inertial transition is preferable; or time concerns in which a rotational pitch transition may be preferable.



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Figure 4-43.- Orbiter clockwise transition.

Transition to the + V-Bar Procedure

Before + V-bar stationkeeping can begin the Orbiter must be positioned on the target's + V-bar. The aft attitude direction indicator (ADI), -Z COAS, RHC, THC, rendezvous radar, and the universal pointing display will be used to maneuver the Orbiter into position.

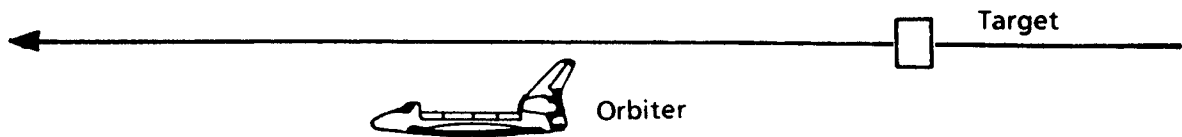
Initial conditions:

- The Orbiter is below and ahead of the target, 1000 feet away.
- The + Z axis is pointing toward the center of the Earth (figure 4-44).

- Select LVLH on the aft ADI and place the sense switch in the -Z position.
- Set up DAP as shown in figure 4-45.

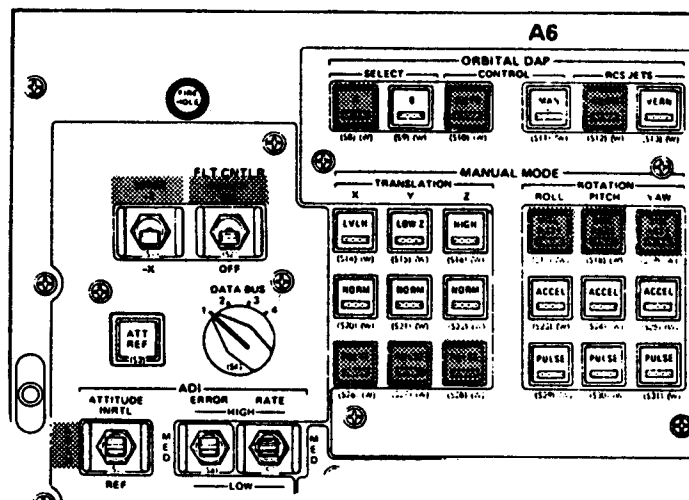
DAP A will be configured with premission defined values.

Normal (primary) RCS jets will be selected only when it is necessary to translate. Vernier jets will be used when not translating.



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Figure 4-44.- Orbiter initial position.



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Figure 4-45.- Panel A6 (DAP).

- C. Without translating, center the target in the COAS using the RHC (pitch the tail down).

Notice that the aft ADI will read something less than 180° in pitch. The ADI will become important in locating the V-bar.

Now that the target is centered in the COAS, the pilot is ready for transition to the V-bar (figure 4-46).

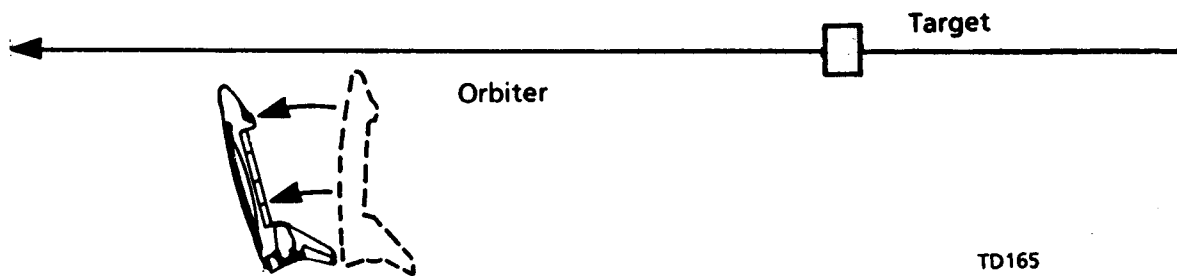
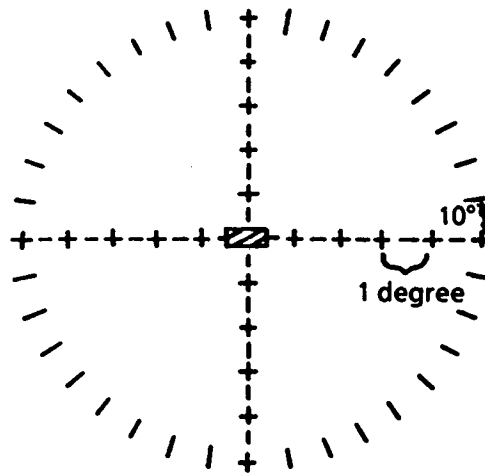
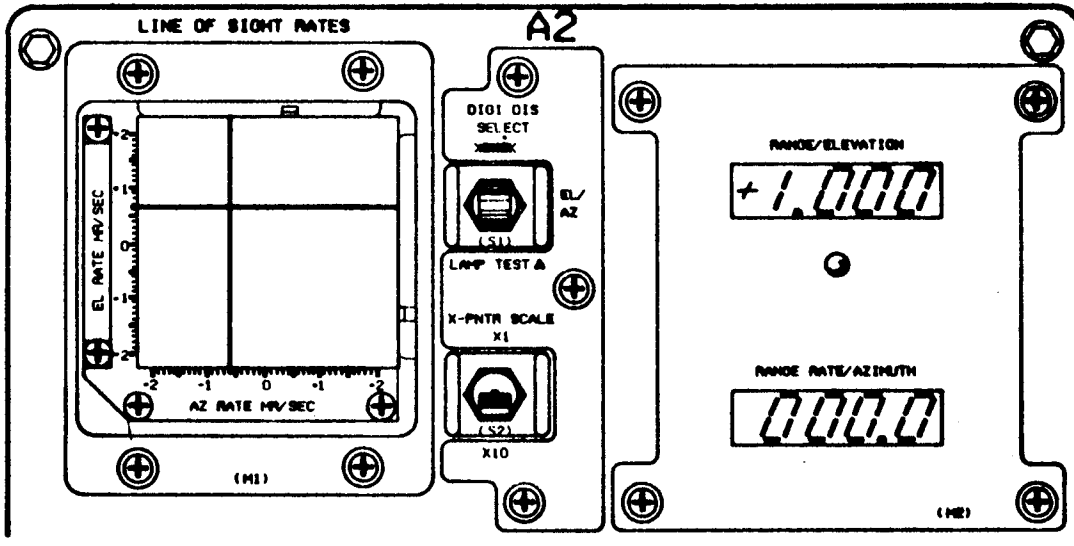


Figure 4-46.—Orbiter rotated to center target prior to transition.

D. Slowly pitch the tail of the Orbiter down, manually, using the RHC or automatically using universal pointing, and translate at the same time to keep the target centered in the COAS. (Refer to the flyaround procedure for more detail.)

It is also important to keep the Orbiter at a constant range (1000 feet in this case). The rendezvous radar information will be used in maintaining the 1000-foot range and zero range rate. The THC (moved in or out) will be used to null any opening or closing rates, respectively (figure 4-47).



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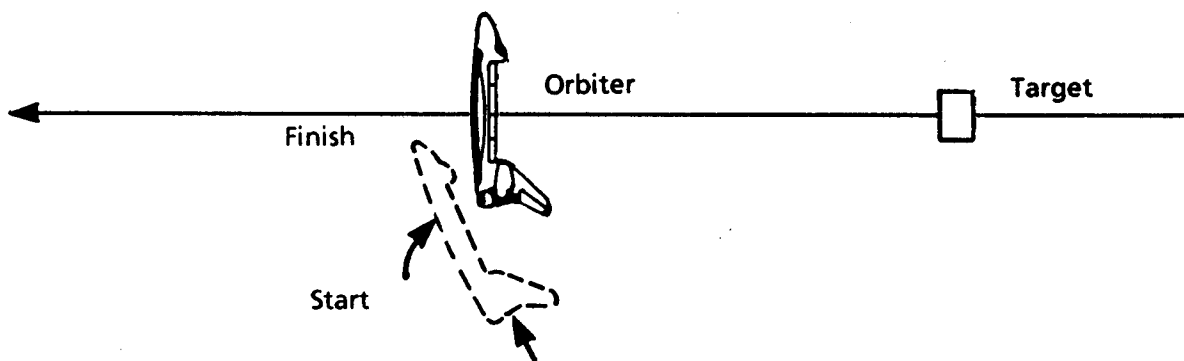
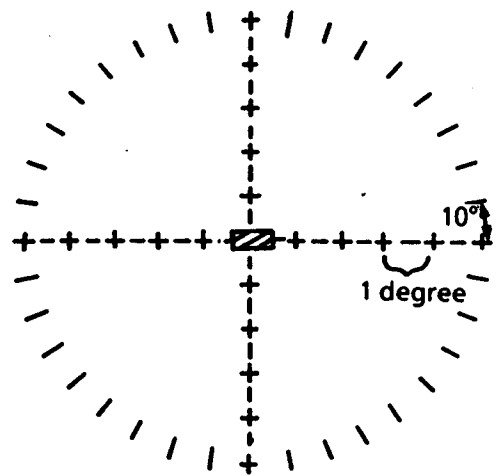
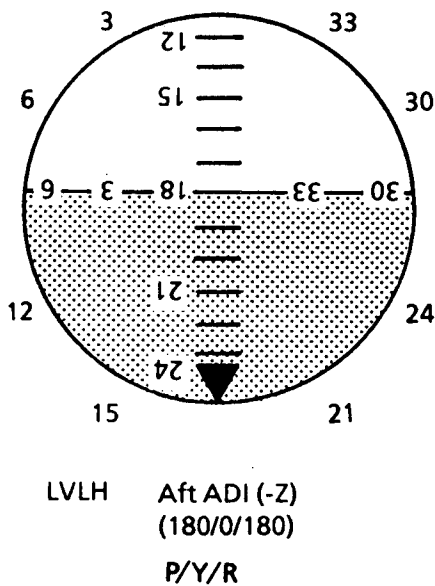
Figure 4-47.- Panel A2 (1000-ft range).

This procedure eventually moves the Orbiter to intercept the target's V-bar. The Orbiter will be on the V-bar when the aft ADI reads 180° in pitch and the target is centered in the COAS reticle (figure 4-48).

The pilot is now ready to set up + V-bar stationkeeping or accomplish the + V-bar approach.

Separation to a Standoff Position

After deploying, retrieving, or servicing a payload, it may be necessary to maneuver the Orbiter away from the payload before continuing. Typically, the Orbiter will be moved along the target's V-bar (the reverse of a V-bar approach). When it is safe distance from the target, their relative rates are nulled and V-bar stationkeeping will commence.



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Figure 4-48.- Transition to V-bar.

Summary (Section 4)

Low Z mode will be used near a target to limit plume impingement.

- Direct approach** – An approach in which the Orbiter is targeted to intercept the target at TPI, using a prescribed sequence of braking gates.
- R-bar approach** – An approach in which the Orbiter moves along the target's positive or negative radius vector to the center of the Earth with a sequence of "hops."
- V-bar approach** – An approach in which the Orbiter moves along the target's positive or negative velocity vector with a sequence of "hops."
- Flyaround** – A maneuver in which the Orbiter points a specified body vector at the target and flies around the target with respect to an inertial reference frame while maintaining a constant range.
- Transition** – A maneuver in which the Orbiter points a specified body vector at the target and flies around the target with respect to a target-centered LVLH reference frame while maintaining a constant range.

Section 5: Rendezvous/Proximity Operations CRT Displays

Introduction

Several CRT displays will provide the crew interface to accomplish the rendezvous and PROX OPS maneuvers. These are as follows:

- A. Universal Pointing (UNIV PTG) – This is the OPS display in GNC major mode 201 (MM 201). It provides the crew with four mutually exclusive options for auto attitude control with respect to the attitude director indicator (ADI) inertial reference frame, and it provides continuously computed Orbiter attitude information. This display also includes the controls for updating the landing sites required for entry.
- B. Orbit maneuver execute (ORBIT MNVR EXEC) – This is the OPS display in GNC MM 202. It provides the crew a means for inputting maneuvers, evaluating their effects on the trajectory, and adjusting parameters as necessary.
- C. Relative navigation (REL NAV) – This is SPEC 33 in GNC OPS 201 or 202. It provides the crew with relative state information and a translational thrust monitor capability for the rendezvous, stationkeeping, and payload deployment/retrieval phase of the mission. It also provides data and controls for Orbit NAV monitoring and rendezvous NAV operations for the star tracker, rendezvous radar, or crew optical alignment sight. NAV filter controls are provided to allow manual control of navigation data from the STRK, RR, or COAS, depending on which is in use.

- D. Orbit targeting (ORBIT TGT) This is SPEC 34 in GNC OPS 201 or 202. It contains the data and controls necessary to set up and target maneuvers required to achieve a given relative position from the target in a given transfer time.

Other CRT displays, of course, will be available during the RNDZ/PROX OPS phases. However, the above displays, or portions thereof, are especially necessary in the RNDZ/PROX OPS phases.

Each of these displays will now be discussed in detail; however, only those portions of the displays related to RNDZ/PROX OPS will be described.

Universal Pointing Display

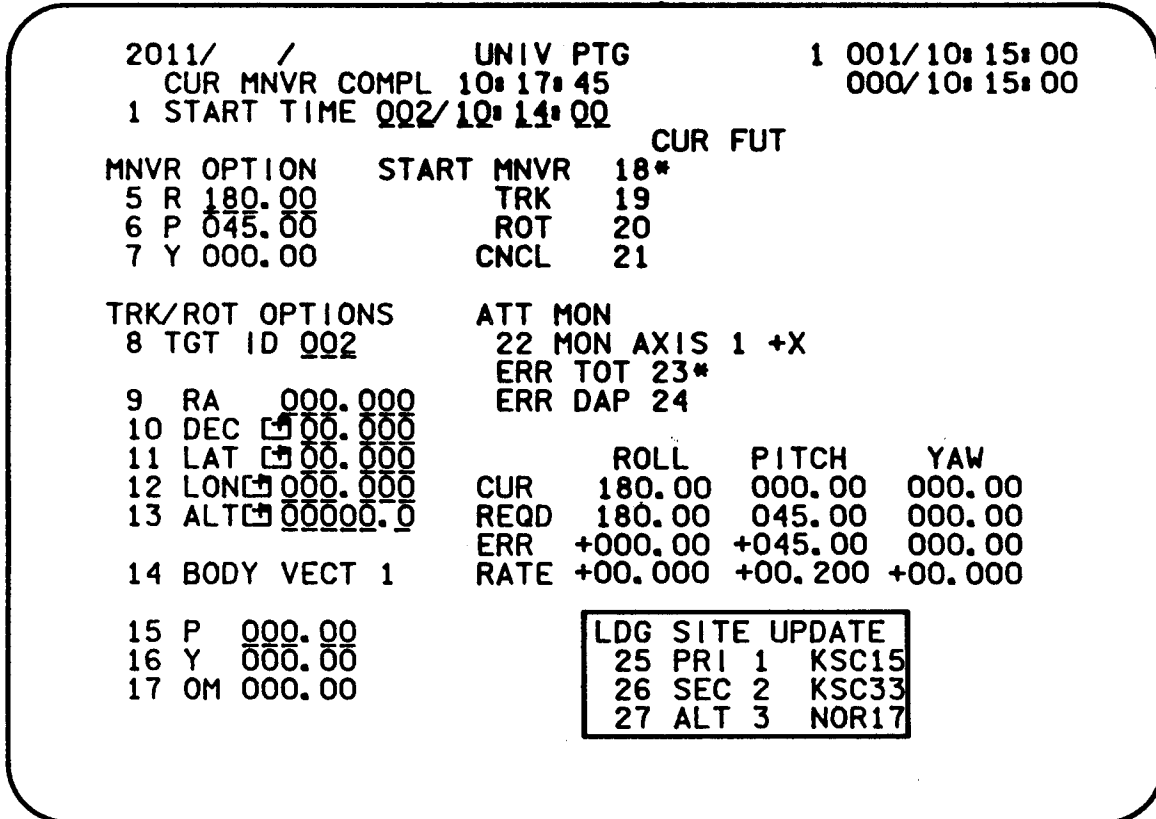
The UNIV PTG display (figure 5-1) is the GNC MM 201 display which provides four mutually exclusive options for auto attitude control with respect to the ADI inertial reference frame, as well as continuously computed Orbiter attitude information. These options are accomplished automatically only if the DAP is in AUTO and are as follows:

- A. Maneuver option (item 18) – This option causes the Orbiter to maneuver automatically to a specified inertial attitude (items 5 to 7) at a specified start time (items 1 to 4).
- B. Tracking option (item 19) – This option causes the Orbiter to point a specified body vector (items 14 to 17) at another orbiting vehicle, an Earth target, the center of the Earth (for LVLH hold), or a celestial target (as selected in item 8) at

a specified start time (items 1 to 4). Among other things, it will be used to point the -Z star tracker at the target prior to taking marks.

- C. Rotation option (item 20) – This option is used to rotate the Orbiter about a specified body vector (items 14 to 16) at the DAP selected rate beginning at the selected start time (items 1 to 4). The DAP selected rate depends on which DAP, A or B, is selected, and the associated rotation rates which appear on the DAP CONFIG SPEC (SPEC 20).
- D. Cancel option item 21) – This option deselects any other option and initiates inertial attitude hold at the attitude existing when the CNCL option was selected.

The controls used in the above four options are included in Items 1 to 21. The following sections give a detailed description of their use.



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Figure 5-1.- UNIV PTG display.

Desired Start Time (Items 1 to 4) – MNVR, TRK, ROT Options

If any one of the first three options (MNVR, TRK, ROT) is selected, the desired starting time, in MET, for the execution of the option must be entered in items 1 to 4. If no start time has ever been loaded, or the currently loaded start time is in the past and an MNVR, TRK, or ROT option is selected, the requested maneuver option will be executed immediately. These items are initialized to all zeroes and maintain the last loaded values after that time.

Roll/Pitch/Yaw Angles (Items 5 to 7) – MNVR Option

These angles specify the desired orientation of the Orbiter relative to the ADI inertial reference frame for the MNVR option. Although they are input in the order of roll, pitch, and yaw, the software uses the pitch, yaw, roll Euler sequence for computation of the required attitude orientation. It also uses a +X axis control sense in interpreting the specified roll, pitch, and yaw angles (as opposed to a -X axis or -Z axis control sense as used in the aft station). These angles are initialized to zero and maintain the last loaded values after that time. The legal range for these values is 0 to 359.99° for roll and pitch, and 0 to 90° or 270 to 359.99° for yaw.

TGT ID (Item 8) - TRK Option

This item allows the input of the type of target used in the TRK attitude option. The six possible target types and their associated TGT ID's are as follows:

TGT ID	TARGET TYPE
1	Other orbiting vehicle
2	Center of the Earth
3	Earth relative target
4	Center of Sun
5	Crew-specified celestial target
11-110	Navigation stars (Star Catalog)

This item is initialized to 2 (for LVLH attitude hold) and it maintains the last loaded value after initialization. Items 9 to 13 may also be necessary in specifying data concerning the target location on the surface of the Earth or specifying the right ascension (RA) and declination (DEC) of a celestial target. The LOS to the Sun is defined by a solar ephemeris maintained by navigation in Earth-centered coordinates.

Right Ascension and Declination (Items 9 and 10)

These items allow input/display of the RA and DEC, with respect to the M50 reference frame, of a crew-selected celestial target when a TGT ID of 5 is entered. Also, if a navigation star is selected, the RA and DEC of that star will be displayed. A user attempt to load a new RA or DEC for a navigation star in the star catalog will result in an ILLEGAL ENTRY message.

Both the RA and DEC are initialed to zero and maintain the last loaded value or the RA/DEC of the last selected navigation star target thereafter. The legal range is 0 to 359.99° for RA, and $\pm 90^\circ$ for DEC.

Entering a TGT ID of 4 causes the data in items 9 and 10 to flash, which indicates that a new value may need to be loaded for the RA and DEC. Each item will cease flashing when it is overwritten by a new value, when the TGT ID is changed, or when an attitude option is executed.

Latitude, Longitude, Altitude (Items 11 to 13)

These items specify the point on or above the Earth's surface, as modeled by the Fischer ellipsoid, at which the Orbiter is to point its specified body vector when a 3 is entered as the TGT ID. When the TGT ID of 3 is entered, the latitude (LAT), longitude (LON), and altitude (ALT) values in items 11 to 13 will flash, which indicates that new data may be input. This flashing will cease when the data is overwritten, when the TGT ID is changed, or when an attitude option is executed. Data for items 11 to 13 can only be input with a TGT ID of 3 in item 8. In any other case, an ILLEGAL ENTRY message will appear. Items 11 to 13 are initialized to zero and maintain the last loaded value after

initialization. The legal ranges are $\pm 90^\circ$ for LAT, $\pm 180^\circ$ for LON, and ± 99.99 n. mi. for ALT.

Body Vector (Item 14) – TRK, ROT Options

The body vector item allows the user to specify a fixed body vector of the Orbiter for use in the TRK or ROT attitude options. In the TRK option, this vector will be the body pointing vector which will point at the selected target. In the ROT option, this will be the body vector about which the rotation occurs.

There are five selectable body vector types:

Body Vector ID	Pointing Vector
1	+ X body axis
2	-X body axis
3	-Z body axis
4	PTC axis
5	Crew-selectable axis

These are the only legal inputs to item 14, and item 14 is initialized to 1.

Pitch and Yaw Body Vector Angles (Items 15 and 16)

These angles specify the pitch (P) and yaw (Y) components of the body vector specified in item 14. If a body vector ID of 5 is input, the values of P and Y begin flashing their currently loaded values. The flashing will cease when the P/Y data is overwritten by new data, when the body vector ID is changed, or when an attitude option is executed. An attempt to load data in these items with a body vector ID different from 5 or outside the allowable range (0 to 90° and 270 to 359.99° for Y; 0 to 359.99° for P) will result in an ILLEGAL ENTRY message. If a body vector ID of 1 to 4 is entered, the associated P and Y values will be displayed. These items are initialized to zero.

Omicron Angle (Item 17) - TRK Option

With this item, the user can specify the attitude orientation of the Shuttle about the selected body pointing vector used in the TRK attitude option. This attitude is specified with the angle omicron.

The omicron item is blanked when a new TGT ID is entered. If the item is still blank when the TRK attitude option (item 19) is selected, two-axis tracking will be implemented. In this mode, the Orbiter-selected body pointing vector is aligned with the target line of sight with an attitude change in which the maneuver eigenangle is minimized. This means that the Orbiter will track the target with the specified tracking body vector but that the Orbiter's orientation about that body pointing vector (omicron) will not be fixed. Instead, it will vary such that the total required rotation is minimized. If an omicron angle is input in item 17 and the TRK option is selected, three-axis tracking will be used. In this mode, both the pointing requirement and the omicron angle restraint are imposed in making computations.

If omicron is not blank, entering a body vector ID in item 14 initiates the flashing of omicron. The flashing is terminated when a value of omicron is entered or when an attitude option is selected.

The allowable range for omicron is 0 to 359.99° . User input outside this range results in an ILLEGAL ENTRY message. The value of omicron is initialized to zero and three-axis tracking is selected.

Selected Attitude Option (Items 18, 19, and 20)

As stated earlier, these items allow the user to select the MNVR (item 18), TRK (item 19), or ROT (item 20) auto attitude option. Assuming the input maneuver start time (items 1 to 4) is not in the future (i.e.; it is in the past or is current), the requested maneuver option will be immediately initiated upon input of one of these three item numbers and an "*" will appear in the CUR column adjacent to the selected option. If the DAP is in AUTO, the vehicle will immediately begin maneuvering to the required orientation. If the tracking or maneuver option is selected with the DAP in

MANUAL, and LVLH is not selected, the final desired attitude and attitude errors will be displayed to the crew. Also, the corresponding error needles will be displayed on the ADI. On the other hand, if a maneuver start time is input that is in the future when the attitude option is selected, an "*" will appear in the FUT column adjacent to the selected option and the maneuver will not be initiated until the previously input start time is reached. In this case, if the DAP is in MANUAL or in the LVLH mode 30 seconds prior to the maneuver start time, a class 3 alarm and fault message, SEL AUTO, will begin flashing on the fault message line to remind the crew to mode the DAP to AUTO. When the start time becomes current, the "*" toggles to the current column and the maneuvering option is initiated as described previously. Whenever the LVLH mode or stop option is selected, the required attitude is set equal to the current attitude, and the attitude errors are zeroed.

The current options are mutually exclusive, and executing another current option will terminate any previously selected current option. It is possible to have one attitude option current and in operation and also input a future start time and attitude option. In this case, the current option remains in effect until the future option becomes current. At that time, it replaces the former active option. The future options are also mutually exclusive. Changing the data necessary for a selected future option or executing a new future option will terminate the previously selected option.

Note that the rotation option (item 20) uses the rotation discrete rate from the DAP CONFIG display (SPEC 20) for the selected DAP (A or B) and the selected jets (NORM or VERN).

Termination of the Auto Attitude Option

The maneuvering resulting from any of the auto attitude options MNVR, TRK, ROT, will be terminated by:

- A. Executing the maneuver item on the MNVR EXEC display, which will relinquish orientation control to the MNVR EXEC display. The MNVR EXEC display is available in major mode 202 (MM 202).

- B. Selecting the LVLH mode (on the ORBITAL DAP crew station panel C3 or A6).
- C. Moving the RHC out of detent.
- D. Selecting MANUAL control on the ORBITAL DAP mode switch.
- E. Terminating the cyclic processing, using the STOP item number (item 21), for Orbiter attitude control and deselecting all previous options. If AUTO control on the orbital DAP mode switch is selected when this control is executed, the Orbiter will go to attitude hold.

Reinitiation criteria – If the STOP option has not been executed, the cyclic processing to orient the Orbiter will be reestablished by:

- A. Exiting the MNVR EXEC major mode.
- B. Selecting auto control on the orbital DAP control panel.

Either of the above will allow the Orbiter to be oriented as specified by the currently selected option.

If the stop option has been executed, the cyclic processing will not be initiated until another option is selected on this display.

Selecting LVLH on the orbital DAP control panel reinitiates the LVLH processing done by the UNIV PTG processor, but does not reinitiate any other auto maneuver mode.

MNVR Completion Time

A maneuver completion time in hours, minutes, and seconds will appear at the top of the UNIV PTG display to indicate the MET of the completion of a current maneuver option or a maneuver to an initial tracking attitude. This number is computed based on the DAP selected rates and is cyclically updated (at 2 hertz) only during a MNVR or TRK attitude option. It will freeze when the current attitude errors are less than 1° or when: (1) the DAP is placed in manual, (2) the LVLH mode is selected via the DAP pushbutton, or (3) when a rotation option is executed.

Cancel Option (Item 21)

Executing the stop item number during an auto attitude option (items 18, 19, or 20) deselects any current or future attitude options and initiates inertial attitude hold about the attitude at the time of selection of the option. Also, an "*" will appear next to the stop option item. This display is initialized with the C NCL option selected.

Attitude Monitoring

It is possible to view the current Orbiter attitude angles, the Orbiter attitude angles required by a selected MNVR or TRK option, the attitude errors between the current and required attitude angles, and the Orbiter body rates during MM 201 using the UNIV PTG display.

Monitoring Axis (Item 22)

This item specifies which axis orientation, + X or -X, will be used as a reference for computing and displaying the current and required attitudes, attitude errors, and Orbiter body rates. If the + X axis is selected by entering an ID of 1, the attitude quantities which are monitored will be displayed relative to the standard Orbiter body axis coordinate system. This corresponds to the attitudes shown on the ADI in the forward crew station. If the -X axis is selected, attitude quantities will be displayed relative to a coordinate system that is rotated 180° about the Z axis with respect to the standard body coordinate system. This is analogous to the attitude information available on the aft ADI when it is in the -X position. A "+ X" or "-X," as appropriate, will appear next to the ID entered in item 22. The display will be initialized with the + X axis (ID 1) selected.

NOTE

This item only affects the displayed attitude parameter below it. It does not pertain to the attitude data entries made in the MNVR option; those values of roll, pitch, and yaw (items 5 to 7) are given in the standard + X attitude control sense.

Attitude Errors (Items 23 and 24)

Two types of attitude errors may be selected for display on both the ADI error needles and the UNIV PTG display. Item 23 (ERR TOT) allows total attitude errors (fly-to-body errors) to be displayed. Item 24 (ERR DAP) allows DAP attitude errors (phase plane errors; i.e., the difference between the DAP desired and the current attitude) to be displayed.

Additional Information on Attitude Monitoring Portion of Display

If the LVLH pushbutton is used or the stop option is executed, the required attitude is set equal to the current attitude, and the attitude errors are zeroed.

If the tracking or maneuver option is selected and the DAP mode switch is in MANUAL at the start time and the LVLH mode has not been selected via the panel pushbutton, the final desired attitude and attitude errors will be displayed on the ADI.

Additional Information Concerning UNIV PTG Display and Software

The processor software associated with the UNIV PTG display remains active after transitioning to MM 202 or OPS 8. Of course, the display itself is not available in MM 202 or OPS 8. However, in MM 202, the powered flight mode, the outputs from the UNIV PTG processor to the orbit DAP are inhibited when a prethrust attitude alignment is made. (Refer to the ORBIT MNVR EXEC display section.) The processing required to support the DAP LVLH attitude mode resumes when LVLH is selected in MM 202. The software required to support the LVLH pushbutton is part of the UNIV PTG software.

When OPS 201 is again called, the outputs from the UNIV PTG processor are again enabled.

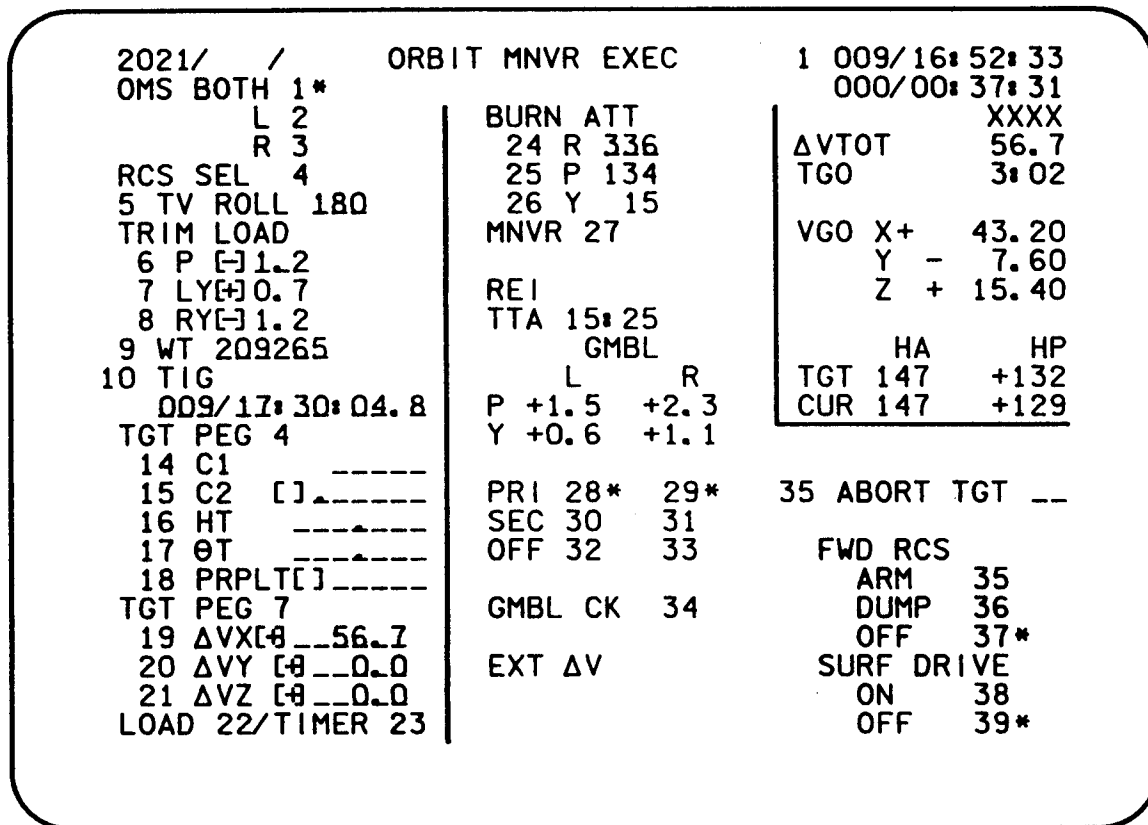
After a transition from MM 201 to MM 202 or OPS 8 and a subsequent transition back to MM 201, the item select status that existed prior to cancellation is shown on the UNIV PTG display.

Selecting the LVLH attitude mode (panel C3) inhibits the cyclic processing of the selected auto attitude option (items 18 to 20) until the LVLH attitude mode is deselected.

The UNIV PTG display is initialized only when OPS 2 is entered from an OPS other than OPS 8.

Orbit MNVR EXEC Display

The ORBIT MNVR EXEC display is the major mode display for GN&C MM 202 (figure 5-2). This display provides the crew interface for monitoring and control of the functions contained in MM 202. These include the controls to maneuver the Orbiter to a burn attitude, monitor, guide, and control the vehicle while executing a translational maneuver or while executing the burns used in the rendezvous/PROX OPS mission phases. The same display is used during ascent, orbit, and entry, with the difference being the title (ASCENT/ORBIT/DEORBIT MNVR EXEC/COAST). Thus, there are some items on this display that are not used in OPS 2 (i.e.; items 14 to 18, REI-range to entry interface status, items 35 to 39). The detailed use of the other items and a description of the displayed parameters follow.



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Figure 5-2.- MNVR EXEC display.

OMS BOTH/L/R, RCS SEL (Items 1 to 4)

These mutually exclusive items allow the user to select whether both OMS engines, or just the left or right one, or the RCS jets will be used in the upcoming maneuver. If an OMS failure is declared by fault detection and identification (FDI), a down arrow will appear in the status column next to item 2 or 3 (L or R OMS engine). The down arrow and FDI are reset if item 1 (BOTH) is subsequently selected. If item 4 is selected (RCS SEL), the burn attitude and burn duration time computations for the targeted maneuver will be made, assuming that the + X RCS thrusters will be used in the high acceleration mode to produce a + X RCS acceleration. A required burn attitude will be computed, based on a minimum attitude maneuver, to align the Orbiter + X axis with the required thrust vector. This data will appear in items 24 to 26.

Thrust Vector Roll (Item 5)

This item allows the crew to select the Orbiter's orientation, in terms of roll, about the thrust vector during an OMS burn. Roll angles from 0 to 359° can be input, with 0° being heads up. It is initialized the first time to an I-loaded value (180°) and retains that value until user intervention. From then on, it comes up with the previously used value which can be changed by the user. If an RCS burn is selected, an attempt to change it will result in an ILLEGAL ENTRY fault message.

Trim Load (Items 6 to 8)

These items relate to the desired pitch and yaw trims for the OMS gimbals that the DAP is to establish for ignition. During a burn, this data will go static.

Weight (Item 9)

This item shows the crew the total Orbiter weight (pounds) as it is computed in guidance. This value may be changed by the crew via this item number or by ground uplink to account for

changes in total Orbiter mass not accounted for by guidance.

Time of Ignition (Items 10 to 13)

These items represent the time of ignition (TIG) in MET (DAYS/HR:MIN:SEC). A check is made by the software to ensure that a time entered in this item is realistic (within some TBD time of the future). If it is not a realistic time, an illegal entry message will be generated.

PEG 4 (Items 14 to 18)

Entries to these items are illegal in OPS 2. These items will be blanked.

PEG 7 (Items 19 to 21)

These items are used to input the three external delta velocities (ΔV 's), ft/s, in the local vertical system at ignition. Also, the ΔV 's computed with the ORBIT TGT software processor (and display) will appear here. In that case, a flag in the software may be set to use Lambert targeting for the burn. (The three ΔV 's and the required offset position are passed to guidance in that case.)

LOAD (Item 22)

The word LOAD will flash any time the display is called up on OPS 2. Once the LOAD item is executed, the word LOAD will cease flashing and the engine selection, trim, vehicle weight, and target data items will be sent to guidance. Also, certain display parameters will be computed including burn attitude, total ΔV , targeted apogee/perigee, burn time, and required ΔV in body components. Later, if the engine selection, trim, vehicle weight, or target data items are altered by crew input or by ground uplink prior to burn enable or after burn completion, the word LOAD will flash. Such an entry will terminate all burn guidance computations and the load dependent parameters will go blank. Any change of these targeting or setup parameters while the burn is enabled, or prior to burn completion, will be considered as an illegal entry.

TIMER (Item 23)

The GNC CRT timer may be started to count down to TIG via this item.

Prethrust Burn Attitude (BURN ATT, Items 24 to 26)

The prethrust burn attitude at ignition in degrees, with respect to the ADI inertial reference frame, will appear by these items after crew execution of the load item. (Recall that the roll orientation about the thrust vector was specified in item 5.) The BURN ATT items may not be used by the user for data entry in OPS 2. Such an attempt would result in an ILLEGAL ENTRY message. The burn attitude generates ADI attitude error such that the crew can maneuver to the desired attitude. In OPS 2, this maneuver may be done automatically with the next item.

MNVR Execute (Item 27)

This item enables an automatic DAP maneuver to the desired ignition attitude given in items 24, 25, and 26. While the maneuver is in progress, an asterisk will appear next to item 27.

Range-to-Entry Interface

This is not applicable in OPS 2 and remains blank.

Time to Next Apsis (TXX XX:XX)

A time is displayed (MIN:SEC) which is the time to next apsis. An alphanumeric title next to the time will read TTA if apogee is the next apsis, TTP if perigee is the next apsis, or if apogee and perigee differ by less than 5 n. mi., the time will be blank and the title will read TTC (for circularization).

Gimbal Status (Items 28 to 34)

In the GMBL section of the display, the position (in degrees) and status ("Δ" for actuator failure, "M" for data missing) of the pitch and yaw engine gimbal actuators for the left and right OMS are given. Also, items 28 to 33 provide the crew controls for selecting the primary or secondary gimbal drives as the active system for the left or right sides. Also, the capability exists to deselect and power down either the left or right gimbal drive systems. The items relating to the left side (items 28, 30, 32) are mutually exclusive, as are the items controlling the right side (items 29, 31, and 33). Either or both of the left and right side systems may be used.

Also, item 34 provides an automatic TVC check of the active gimbal drive system. An "*" will appear while the test is in progress. After the check is completed, the gimbals are returned to their initial positions and the asterisk is blanked.

Burn-Related Parameters

Several parameters used during a burn will be displayed in the upper right-hand corner of this display.

- A. EXEC: AT TIG – 15 seconds, the word EXEC will appear and will flash until a keyboard EXEC is performed, at which time the characters will disappear. Ignition is inhibited until this entry is made. If the EXEC entry is made after the specified TIG, the maneuver starts immediately following the EXEC entry. Releasing the display prior to enabling the maneuver (e.g.; transitioning to MM 201, OPS 3, or OPS 8) cancels that maneuver.
- B. ΔV TOT: This parameter identifies the predicted total ΔV of the targeted maneuver.
- C. TGO: This parameter reflects the predicted burn length (MIN:SEC) before TIG occurs. After ignition, it reflects the remaining burn time.
- D. VGO X/Y/Z: These parameters show the velocity to be gained

component-wise in ft/s with respect to the vehicle body axis system. Note that for RCS selected burns, the computed initial burn attitude is such that the velocity to be gained is only in the Orbiter's X-axis direction. Cross-coupling effects may cause Y and Z components to appear during the burn, however. These values can be used by the crewmember to execute the required RCS burn manually using the THC.

- E. HA, HP TGT/CUR: These parameters reflect apogee (HA) and perigee (HP) information in nautical miles (n. mi.) for the current orbit trajectory (CUR) and for the predicted postburn orbit (TGT).

Abort Modes (Item 35)

This item is not active in MM 202. In MM 104 and 105 this item overrides the rotary switch selection for abort modes.

Forward RCS Propellant Burn-off Controls (Items 36 to 38)

These items are available only in OPS 3. Execution of them in OPS 2 results in an ILLEGAL ENTRY message.

Surface Drive (Items 39 and 40)

These items allow the initiation and termination of the surface drive function which provides simultaneous motion of all aerosurfaces for hydraulic fluid thermal conditioning in OPS 3. Execution of these items in OPS 2 results in an ILLEGAL ENTRY message. Initially, OFF (item 39) will be selected.

Relative Navigation Display

The relative navigation (REL NAV) display (figure 5-3) is SPEC 33 and is available in GNC OPS 2. It provides the crew with relative state information and a translational thrust monitor capability for the rendezvous, PROX OPS, and payload deployment/retrieval phases of the mission. It also provides data and controls for rendezvous NAV operations using the star tracker (ST), rendezvous radar (RR), or crew optical alignment sight (COAS) (as specified by the user). NAV filter controls are provided to allow manual control of navigation data from the ST, RR, or COAS, depending on which is being used.

Rendezvous Navigation Enable (Item 1)

This item is used to transition between the basic orbit navigation and rendezvous navigation. It is a flip-flop item and is inhibited on initial display call-up. An asterisk will appear when rendezvous navigation is enabled. During rendezvous navigation the target state vector and the Orbiter/target covariance matrix are propagated using preloaded acceleration models and external sensor data (STRK, RR, COAS), if available. In orbit navigation, the target state vector is not propagated and no external sensor data is used to update the target/Orbiter state vector. TGT SV must be loaded in order to enable RNDZ NAV.

Ku-band Antenna Enable Function (Item 2)

Execution of item 2 results in the GNC GPCs passing the GNC parameters (such as the current navigated Orbiter and target state vectors) which are necessary for Ku-band antenna management to the SM GPC over the ICC data buses. Recall that the rendezvous radar is part of the Ku-band antenna system, and that this function is controlled by the SM OPS 2 GPC. Prior to rendezvous radar acquisition of a target by the GPCs, this item must be enabled.

State Vector Update

Two parameters, SV UPDATE POS and VEL, display the magnitude of the change to the filtered state vector position (POS) and velocity (VEL) made by the rendezvous navigation filter in the last computation cycle.

Relative Motion

The relative motion between the Orbiter and the target vehicle, including the current navigated relative state information and the relative state information from the RR system will be displayed in the upper left portion of the display.

The following current navigated relative state information will be blank until rendezvous navigation is enabled (item 1). After that time, it will be updated once every two seconds until the display is terminated via crew command.

- RNG: Line of sight range (kft) between vehicles.
- \dot{R} : Range rate in ft/s; sign convention is negative for a closing rate and positive for an opening rate.
- θ : The angle between the projection of the Orbiter tracking body vector (selected in UNIV PTG) into the orbit plane and the positive local horizontal vector (0° to 360°): This could be used in detecting the TPI point where the elevation angle would be 28.4° , assuming the + X axis is used as the Orbiter tracking body vector.
- Y: Out of plane position: The current perpendicular distance between the Orbiter and the plane of the target vehicle (kft); positive along the negative angular momentum vector of the target.
- \dot{Y} : Out of plane velocity: The current rate at which the Orbiter is moving relative to the target orbital plane in ft/s; positive along the negative angular momentum vector of the target.

- NODE: Time of the next Orbiter/target orbital nodal crossing (MET HR:MIN:SEC):

The current RR determined relative state information shall be initialized upon display callup and subsequently updated at 1 Hz until the display is terminated via crew command. A status word will be displayed next to RR to inform the crew of the rendezvous mode currently being used. This status word will be one of the following: COMM (communications mode), GPC (GPC acquisition radar mode), GDSG (GPC designation radar mode), ATRK (automatic tracking radar mode), MSLW (manual slew radar mode), STST (self-test in progress), or blank (initialization). This is a reflection of an RR rotary switch on panel A1.

A status symbol shall be provided to the right of each parameter either as an indication of loss of data good (down arrow), an indication that the radar is in the COMM mode and a commfault exists (M), or left blank. The appropriate parameters will be blanked if the radar system is not operative or if the specific parameter has a commfault associated with the data.

The RR determined relative state information includes the following:

- RNG: Line of sight range (kft) between the vehicles.
- \dot{R} : Range rate (ft/s); sign convention is the same as for the current navigated state above.
- \dot{E} L: Pitch position (ELEVATION) of the radar antenna gimbal relative to its null position ($\pm 90^\circ$); positive sense – antenna motion toward the + X Orbiter axis.
- AZ: Roll position (AZIMUTH) of the radar antenna gimbal relative to its null position ($\pm 180^\circ$); positive sense – antenna motion toward + Y Orbiter axis.
- ω P: Pitch inertial angle rate (\pm mr/sec); positive sense – antenna motion toward the + X Orbiter axis. (mr = milliradians)

- ω_R : Roll inertial angle rate (\pm mr/sec); positive sense – antenna motion toward the $-Y$ Orbiter axis. (mr = milliradians)

This radar information can also be seen on a crew station panel (panel A2).

Powered Flight Navigation Controls and Displays (Items 5, 6, and 7)

The upper right-hand portion of the display includes displays and controls useful during powered flight. In particular, there is a flip-flop item number (item 5) which is used to turn on or off the powered flight propagation integrator in navigation. This is called AVG G or super G. The powered flight propagation integrator, AVG G, uses IMU-sensed accelerations (if their magnitude exceeds a specified threshold) to propagate the Orbiter state vector. If the user does not intervene via item 5, AVG G is on during major mode 201 (the orbit/rendezvous coast mode). When AVG G is off, IMU-sensed velocities are not used in updating the Orbiter state vector. Instead, models of the Earth's gravitational acceleration, aerodynamic acceleration, venting acceleration, and uncoupled RCS thrusting acceleration will be used to update the Orbiter state vector. With this type of modeling, RCS translations or rotations made manually by a crewmember with the THC or RHC will not be accounted for in the navigation model.

In summary, in MM 201 AVG G will be enabled until the user executes item 5. Following the transition to MM 202, AVG G will automatically be enabled and the user will receive an ILLEGAL ENTRY message if an attempt is made to use this item in MM 202.

In the upper right-hand portion of the display, a translation thrust monitor capability is possible when AVG G is enabled. The velocity gained along each of the X, Y, and Z body component axes, as well as the total velocity to be gained, will be displayed. The individual velocity component registers will increase or decrease according to the sense of the velocity change, whereas the total velocity will increment due to any velocity change. An item number is provided (item 6) to reset all of the component registers to zero, and another item number (item 7) is used to reset the total velocity register to zero. The first time in OPS 2

that this display is called, these values will be zeroed. Thereafter, they will reflect the values which were available the last time the display was terminated.

State Vector Transfer (Items 8, 9, 10, and 11)

Four item controls are provided for the transfer of state vectors. Whenever any of the four transfer items is executed, an asterisk (*) will be displayed next to the appropriate item number while the selected transfer is in progress.

Filtered to Propagated or Propagated to Filtered (Items 8 and 9)

Item 8 transfers and overwrites the filtered state vector into the propagated state vector slots while item 9 performs the reverse. The filtered state vector is one that incorporates rendezvous navigation data derived from rendezvous radar, star tracker, or COAS measurements. The propagated state vector is a separate state vector which is not updated by RNDZ NAV filter inputs. The difference between the PROP and FLTR state vector estimates of the vehicle's distance from the center of the Earth (OS) and orbital velocity (VEL) are displayed above the state vector transfer items. These data will be used as a criterion for updating the PROP state vector with the FLTR state vector.

Target to Orbiter or Orbiter to Target (Items 10, 11)

During rendezvous navigation, the GNC GPC's propagate both the Orbiter and target state vectors. Item 10 (ORB to TGT) can be used to overwrite the target state vector with the Orbiter state vector, while item 11 (TGT to ORB) can be used to overwrite the Orbiter state vector with the target state vector. An asterisk will be driven next to the appropriate items while the transfer is taking place, and will continue to be displayed for at least one full NAV cycle (4 to 60 sec). Depending upon the length of the NAV cycle and where in the NAV cycle the item (10 or 11) is executed, a delay of

up to 16 sec may be experienced before the transfer takes place.

Rendezvous Angle Sensor Data (Items 12, 13, and 14)

When rendezvous navigation is enabled with item 1, the following external sensor data will be processed: RR shaft angle (elevation), trunnion angle (azimuth), range and range rate, ST horizontal and vertical angles, and COAS horizontal and vertical angles. In MM 202, these measurements will be used for display purposes only, unless the MEAS ENA item is selected (item 3). Assuming RNDZ NAV is enabled and, if in MM 202, the MEAS ENA item is selected, the following occurs. The RR range and range rate are always incorporated in navigation as long as the RR DATA GOOD flag indicates valid data and the data must pass a navigation filter test. However, only one of RR, ST, or COAS will be used as a source of angle data as selected by the crew via items 12, 13, or 14, respectively. These are mutually exclusive items. Once again, a DATA GOOD flag for the selected external sensor must indicate valid data from the sensor, and the data must pass a navigation filter test in order to be used in navigation. Thus, the sensor must be in the appropriate mode. In particular, the RR must not be in a self-test mode, the ST must be in the target track mode, and the COAS data must not have been previously processed or over 2 minutes old. The display is initialized with ST angle data selected (item 12). When a new angle source is selected, a delay of up to 60 sec may be experienced before the asterisk switches to the new source.

Also included in the RNDZ ANGLE SENSOR block is information used to monitor ST operation. The STRK/COAS CNTL display is used to select which ST (-Y or -Z) is to be in the target track mode. The status of the selected ST will be displayed below item 12 and can be one of the following: (blank), ST FAIL, ST PASS, FALSE TRK, NO TARGET, OUT FOV, and HI RATE. Below this status, a BITE indication can be driven for the selected ST. Also, if the -Z ST is being used, the offset angles (X and Y) that represent the displacement of the target being tracked (by the -Z ST) in the -Z COAS field of view will appear. These offset angles allow the crewmember to locate the target that the -Z ST is tracking in the COAS.

Rendezvous NAV Filter (Items 15, 17 to 25)

As stated in the preceding section, data from the RR, ST or COAS must pass a navigation filter residual edit test before it can be used to update navigation (the target/Orbiter relative state vector and covariance matrix). In the bottom third of this display, the results of this filter test and some crew controls are located. This allows the crew to monitor the condition of the filter and the external sensor measurements. In particular, the RR range and range rate residual edit test results are displayed, as well as the results for the selected sensor's angle data (ST-V/H; RR-EL/AZ; COAS-Y/X).

NAV Rates (SLOW RATE, Item 15) Currently No Slow Rate

The state vector is updated and data (marks) from the ST and RR are automatically requested by NAV for processing at one of two rates during MM 201. Some tentative numbers for the NAV rates in MM 201 are once every 16 sec for the slow rate and one every 8 sec for the high rate. In MM 202, the rate is once every 3.88 sec. In MM 201, it is desirable to process marks at the faster rate. Item 15 on this display is used to cause the other rate to be used. It is a flip-flop item. An asterisk is displayed next to item 15 when NAV is processing marks at the slower rate. Thus, the first time the display is called in OPS 2, the asterisk will be present.

Residuals

A residual is calculated by taking the difference between the NAV computed estimate of the relative position of the Orbiter and target, and the observed measurement using sensor data (ST, RR, COAS). If rendezvous navigation is enabled, the residuals are computed and displayed for RR range and range rate, and for the selected sensor's angle data (V/EL/Y, H/AZ/X). These data are blanked when rendezvous NAV is inhibited or if data are missing or invalid.

Residual Edit Ratio

The residual edit ratio is the absolute magnitude of the actual measurement residual divided by the maximum magnitude that the residual may attain before automatic data editing by the filter. The maximum magnitude the residual may attain before data editing is, in part, a function of the covariance matrix. This ratio will be blank if valid data were not presented to the filter.

Mark History, Accept/Reject

The number of NAV marks of each type (range, range rate, or angle) that have been accepted or rejected by the RNDZ NAV filter will be displayed. When four sequential marks are rejected, a down arrow will be driven in the status column. This down arrow is removed when the data are forced and the ratio test is passed, or when three acceptable marks occur.

These ACC/REJ counters are reset to zero under the following conditions:

- When the input data source (STRK, RR, COAS) is changed
- When rendezvous navigation using external sensor data is enabled (execution of item 1 in MM 201 or execution of both items 1 and 3 in MM 202)
- After covariance matrix reinitialization (execution of item 16 or when rendezvous NAV is enabled – item 1).

Also, if a state vector update is forced, the reject counters are zeroed.

NAV Edit Test (EDIT OVRD – AUT/INH/FOR, Items 17 to 25)

The displayed residuals, ratios, accept, and reject counters serve as the basis for the crew decision as to how external measurement data are to be processed by the NAV filter. Three mutually exclusive options are available to the crew for each of the range (items 17, 18, and 10), range rate (items 20, 21, and 22), and selected sensor angle data (items 23, 24, and 25). These are as follows:

- **AUTO:** The normal NAV filter edit criterion will determine whether the data are to be used to update the state vector and covariance matrix.
- **INHIBIT:** Valid data are to be used for computing display parameters, but are not to be used to update the state vector and covariance matrix. Thus, when in INHIBIT, the ACCEPT/REJECT counters will not be incremented.
- **FORCE:** The maximum allowable residual used in the NAV filter edit test is increased to a higher value. Valid data will be utilized to update the state vector and covariance matrix only if the new edit criterion is met on the residual edit test. This new test criteria is in effect for only one NAV cycle. The control then reverts to the previously selected option (AUT or INH). An "*" appears beside the FORCE item only while the control is in effect.

When the display is first called, the INHIBIT items will be selected. These items are applicable only when rendezvous navigation is enabled, and if in MM 202, the MEAS ENA item is selected. Also, in MM 202, external sensor data are used only when the IMU-sensed delta V are below a certain threshold. When the threshold is exceeded, external measurements are not incorporated in the state vector or Orbiter/target covariance matrix, and thus, the ACC/REJ counters will go static.

Depending upon the NAV cycle length, a delay of up to 16 sec may be experienced prior to the "*" appearing when one of items 17 to 25 is selected.

Covariance Matrix Reinitialization (Item 16)

This item is used to reinitialize the 13 by 13 dimensional Orbiter/target covariance matrix. An asterisk will appear next to this item while the reinitialization takes place and will remain on for a minimum of one full NAV cycle (4 to 16 sec). As in the state vector transfer, a delay of up to 16 sec may be experienced prior to the start of the reinitialization. This covariance matrix is propagated only when rendezvous navigation is enabled (item 1), and thus, this item is only useful at that time. It is

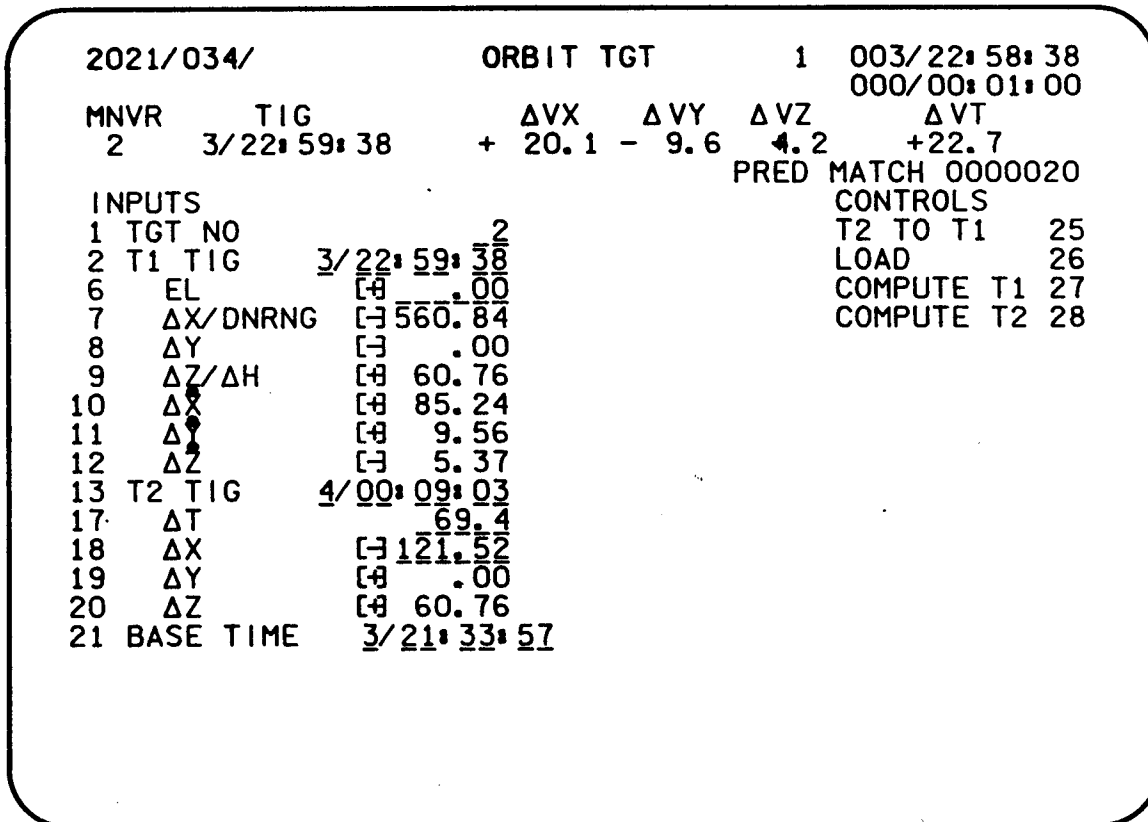
also the case that the covariance matrix is reinitialized when RNDZ NAV is first enabled. Item 16 allows the crew to do this manually. When the covariance matrix is reinitialized, the MARK HIST ACC/REJ columns at the bottom of the display will be zeroed. Note that the covariance matrix measures the "cloud of uncertainty" that navigation sees about the state vector. It is used to determine the validity of external data by the NAV filter prior to incorporation of that data into the state vector.

Orbit Targeting Display

The Orbiter targeting (ORBIT TGT) display is SPEC 34 and is available in GN&C OPS 201 or 202. (See figure 5-4.) This display and the associated orbit targeting software give the crew the capability to generate a software targeted two maneuver set. The first maneuver

is targeted to move the Orbiter from a given position to a desired target relative position in a given amount of time (ΔT). The second maneuver is targeted to null the Orbiter's velocity relative to the target centered reference frame. (This reference frame is the same one as used in the relative motion plots.) Some examples of when this software could be used are: at NCC, TPI, MCC, TPF braking, stationkeeping, transition, approach, and separation. The targeted maneuver outputs of this software are displayed to the crew on the ORBIT TGT SPEC and are also available as inputs to the maneuver execution software and CRT display, ORBIT MNVR EXEC in MM 202. The ORBIT TGT display is divided into three sections:

- Inputs
- Controls
- Maneuvers



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Figure 5-4.- ORBIT TGT display.

General Description

In general, a sequence of rendezvous, braking, stationkeeping, or transition maneuvers can be performed by selecting the appropriate target set and executing the computation with the ORBIT TGT display, then executing the maneuver with the aid of the ORBIT MNVR EXEC display in MM 202 and repeating that procedure for each remaining maneuver in the sequence. One of 40 targeting sets can be chosen by the crew via item 1 on the ORBIT TGT display. The possible target sets are part of the I-load and may be changed by ground uplink. In most cases, each target set has a nominal time (T1) associated with it, relative to the "Base Time", at which the first maneuver should occur and the desired position, with respect to the target (ΔX , ΔY , ΔZ) where the Orbiter should be a given amount of time (ΔT) later. In this case, the target's elevation with respect to the Orbiter's local horizontal is not used to determine T1. (Item 6 is zero.)

Upon entering the target set's ID in item 1, the following items will appear with data: 2 to 5, 17, 18, 19, and 20. These parameters are used by the targeting software to compute the first burn of the specified two-impulse maneuver set. This computation is not made until a COMPUTE T1, item 27, is executed. When the COMPUTE T1 item is executed, the other items (7 to 12, 3 to 16) in the input list will appear. Items 7 to 12 ($\Delta X/DNRNG$, ΔY , $\Delta Z/\Delta H$, ΔX , ΔY , ΔZ) show the Orbiter's relative position (kft) and velocity (ft/s), with respect to the target-centered rotating curvilinear reference frame, at the input time T1 based on current onboard navigation information. Items 13 to 16 (T2 TIG) represent the time of the second maneuver of the two-impulse set and are computed based on the T1 TIG time and the given ΔT . As will be described later, the computed ΔV 's (ΔV_X , ΔV_Y , ΔV_Z , total ΔV_T) necessary for the first burn will appear in the MNVR section of the display and will also be passed to the on-orbit guidance software. The burn is not executed, however, until it is verified. The burn is then executed automatically following a crew entry to the ORBIT MNVR EXEC display (OMS) or manually by the crewmember using the THC (RCS).

Other options are available using this display. For example, with the elevation option, the T1 time is determined onboard and is the time when the target line of sight is the

specified number of degrees (item 6) from the Orbiter's local horizontal axis. As an example of another user option, if the input ΔT time (item 17) is negative, a flag is set in the S/W such that a new T1 and T2 time will be computed with an algorithm which ensures that the second burn is accomplished in a direction orthogonal to the target line of sight. This could be used in orthogonal braking.

The following sections contain a detailed description of the items on this display.

Inputs

Target Number (Item 1)

As stated earlier, the capability exists to load 40 predefined maneuvers in the I-load preflight. These maneuvers may be changed through ground uplink or by the crew through items 2 to 20 and execution of a LOAD item (item 26) on this display. There is a pointer internal to the software which is set to identify which maneuvers are targeted via Lambert targeting and which are solved using the closed form Clohessy-Wiltshire (C-W) equations. All target sets with TGT ID numbers less than or equal to the pointer will be solved using the Lambert equations. All target sets with TGT ID greater than the pointer value will be solved using the C-W equations. This pointer is part of the I-load and can be changed by the ground via uplink or by the crew via GPC main memory modification using the GPC MEMORY SPEC. The number entered with item 1 will retrieve the necessary targeting inputs for that maneuver set from main memory and display them in the appropriate locations on the display. An example of the projected target sets is given below.

TARGET SET ID	MNVR TO BE COMPUTED
1	NCC
2	T1
3	MC1
4	MC2
5	MC3
6	MC4

NOTE

In the following sections, assume that a TGT SET has been entered in item 1.

T1 TIG (Items 2 to 5)

These items specify the time (days, hours, minutes, seconds) in MET of the first burn in the set. If the T1 TIG is in the past (as may happen if a burn is missed or if it is recomputed due to high velocity residuals following the first execution of the burn) and the COMPUTE T1 item (item 27) is executed, a new T1 TIG will automatically be computed as the current time plus one minute. Also this will cause software to ignore any inputs in items 7 to 12, and instead, the software will use the onboard navigated state at that new T1 TIG time. It is also possible to change the T1 TIG through a keyboard entry to items 2 to 5. In this case, the word LOAD will flash next to item 26, and after the input values are verified, they will be loaded into the I-loaded slots for the T1 TIG for that target set after execution of item 26.

The I-loaded T1 TIG times are actually loaded in the software as a specified ΔT which is then added to a BASE TIME (items 21 to 24).

Elevation Angle (EL-Item 6)

This item refers to the elevation angle, positive up, to the target. If the EL = 0, the input T1 TIG and associated values will be used. However, for certain maneuvers, such as TPI, an elevation angle $\neq 0$ (such as 27°) will be loaded. When this is the case, the T1 TIG time is determined by the onboard software as the next time at which the elevation angle is equal to value given in item 6.

Relative Position Items (T1 $\Delta X/\Delta Y/\Delta Z$, $\Delta X/\Delta Y/\Delta Z$ - Items 7 to 12)

These items show the Orbiter relative position (kft) and velocity (ft/s) with respect to the target at the T1 TIG time. If they are all zeroes, if any item of the six is blank, or if the T1 TIG time is in the past, all of their values will be computed and displayed based on the onboard navigation information at the time a COMPUTE T1 (item 27) is executed. Otherwise, the values

which appear beside items 7 to 2 will be used as the correct relative position and velocity of the Orbiter at the T1 TIG time regardless of the onboard NAV software in the targeting algorithms by inputting new values in items 7 to 12. When one of items 7 to 12 is entered, the LOAD item (item 26) will flash until it is executed. Execution of this item actually loads the values in the INPUTS section of the display and in the I-loaded table for that target set, which is then used in the computations.

T2 TIG (Items 13 to 16)

This item represents the time in MET of the second maneuver of the two-impulse set. If ΔT (item 17) is not blank, a COMPUTE T1 (item 27) will cause the T2 TIG to be calculated and displayed. If the T2 TIG is in the past when a COMPUTE T2 is done, it will be reset to current time plus one minute, and the maneuver is computed using the navigated state vector at the new T2 TIG time.

Delta-T (Item 17)

Delta-T (ΔT) represents the transfer time in minutes between the first maneuver time (T1 TIG) and the second maneuver time (T2 TIG). If ΔT is blank, it will be computed based upon the T2 and T1 TIG times when a COMPUTE T1 is executed. If ΔT is negative upon execution of a COMPUTE T1, both ΔT and T2 TIG will be computed based on an algorithm used in orthogonal braking and will be displayed. The algorithm computes the ΔT (and thus the T2 TIG) necessary to ensure that the second burn of the two-burn maneuver set occurs normal to the line of sight from the Orbiter to the target.

T2 Relative Position ($\Delta X, \Delta Y, \Delta Z$ - Items 18 to 20)

These parameters represent the desired Orbiter relative position (kft) at the T2 time (ΔT after T1 TIG - items 13 to 16). This is the target position for the first COMPUTE T1. If a COMPUTE T2 is done, a burn will be computed that will null the relative velocity between the Orbiter and target at this position (which is at the T2 TIG time).

Base Time (Items 21 to 24)

This is a time in MET from which subsequent maneuvers are calculated. As was stated earlier, each maneuver time in each maneuver set is computed as a ΔT added to a base time. Thus, in general, a sequence of maneuvers such as NCC, NSR2; or TPI, MCC, MCC2, TPF can be executed without adjusting T1 TIG times. If a maneuver has to slip in time, it is preferable to change the base time instead of that maneuver's individual T1 TIG time. Otherwise, all of the following maneuver's T1 TIG times would also have to be altered. The base time is originally an I-loaded value. It automatically changes when the elevation angle search option is enabled for a Lambert targeted maneuver, as this will cause the T1 TIG time to change. It is reset to be that maneuver's computed T1 TIG time. Thus, the base time is updated to the TPI time when the TPI maneuver is computed, based on an elevation search.

Controls

T2 to T1 (Item 25)

This item forces the T2 TIG time of the previous maneuver set to be transferred to the T1 TIG slot of the currently called maneuver set. This is useful, for example, in orthogonal braking where the ΔT and T2 TIG times for each maneuver are recomputed in order to ensure that the burn at the T2 time is normal to the target LOS. In this case, it is desirable to target the next two-impulse maneuver set with the T1 maneuver being at the previously defined T2 TIG.

LOAD (Item 26)

Execution of this item loads item entry data values into the data slot for use by the targeting software. They actually overwrite the I-loaded values so that if that maneuver set is again requested via item 1, the loaded values will appear. If any item entry (except TGT NO) in the INPUTS section is made, the word LOAD flashes until item 26 is executed. After it is executed, it will remain static until a new item is entered in the INPUT section.

COMPUTE T1 (Item 27)

Execution of this item results in the software calculating the first maneuver of the specified two-maneuver set. The calculated results are displayed in the MNVR section of the display and are also output to guidance for display on the ORBIT MNVR EXEC display. Also, all of the items in the INPUTS section of the display will fill in. An "*" will be driven beside ITEM 27 until the calculations are complete (15 to 90 seconds for Lambert targeting; only a few seconds for C-W targeting). As soon as the T1 computation is complete, the GNC CRT timer will start counting down on the next unexecuted maneuver TIG.

COMPUTE T2 (Item 28)

This item is used to initiate the calculation of the second maneuver of the two-impulse set. This second maneuver is calculated to null the relative velocity at the T2 TIG. As stated earlier, if the T2 TIG is in the past, it will be updated to the current time plus one minute. An "*" is driven by item 28 until the maneuver calculations are complete. When the calculations are complete, the required ΔV 's are displayed in the MNVR section of the display and are passed to guidance awaiting execution. Also, the GNC CRT timer will start counting down to the next unexecuted maneuver TIG.

Note that execution of an item 28 (COMPUTE T2) while the computation for item 27 is still in progress, and vice versa, is illegal and an ILLEGAL ENTRY message will be generated if that is attempted.

Pred Match

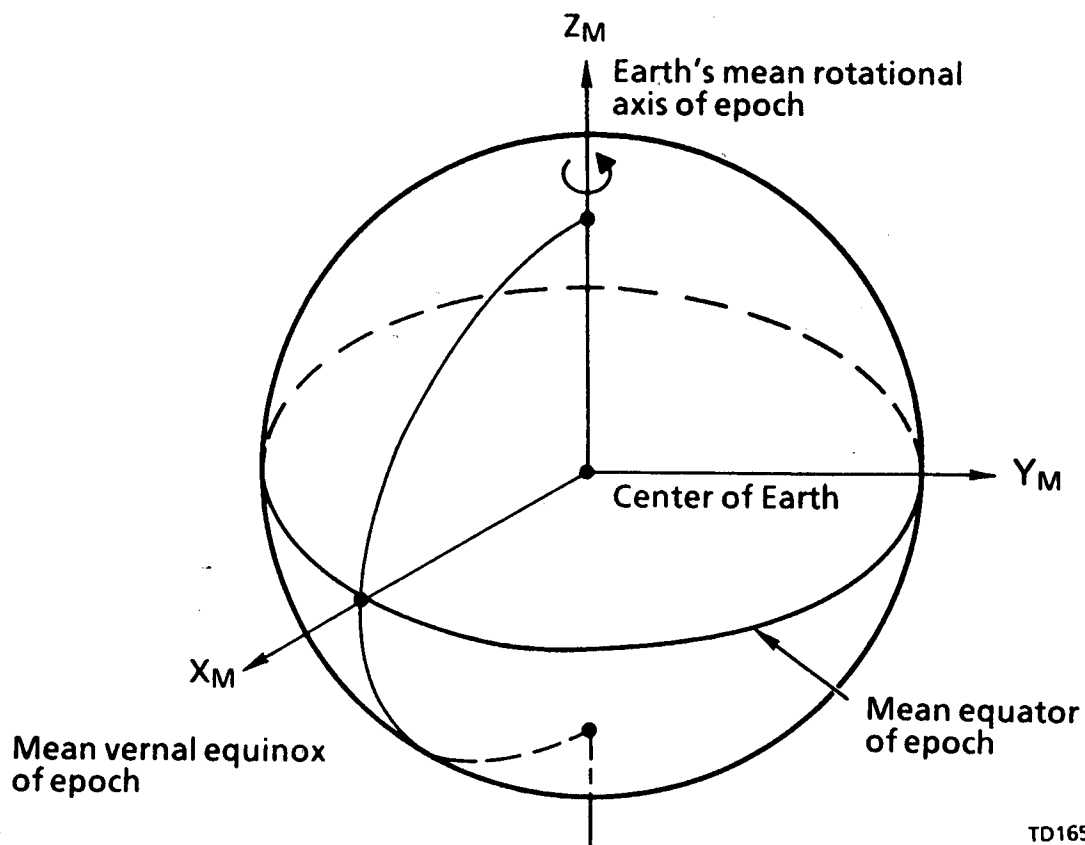
This is the accuracy of the targeted burn in feet. It tells how well the on-orbit predictor was matched in the targeting.

Maneuvers

This section (top portion of display) identifies parameters for one of either of the two-impulse maneuvers which has been computed. A maneuver using COMPUTE T1 is that required to intercept the desired position in a desired time, and a maneuver using COMPUTE T2 is that required at the intercept time to null relative motion (as observed in a target-centered rotating curvilinear reference frame). Quantities displayed in this section for each maneuver include:

- XX – The target number on which the maneuver is based
- X – An indicator (*) that the TIG is in the past
- TIG – The impulsive ignition time (days, hours, minutes, seconds) in MET
- ΔV_X – The LVLH X-component of the targeted ΔV (ft/s)
- ΔV_Y – The LVLH Y-component of the targeted ΔV (ft/s)
- ΔV_Z – The LVLH Z-component of the targeted ΔV (ft/s)
- ΔV_T – The magnitude of the targeted ΔV (ft/s).

Appendix A: Coordinate Systems



Name: Aries-mean-of-1950 Cartesian coordinate system.

Origin: The center of the Earth.

Orientation: The epoch is the beginning of Besselian year 1950 or Julian ephemeris date 2433282.423357.

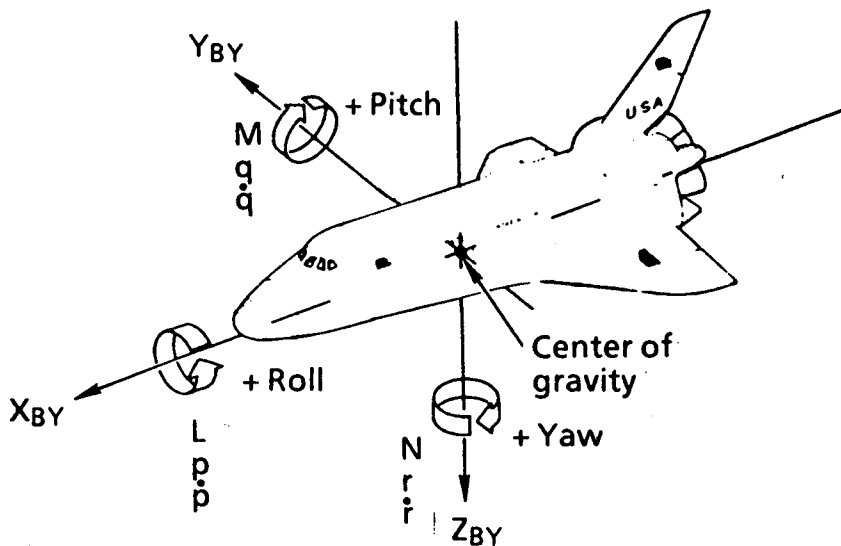
The X_M - Y_M plane is the mean Earth's equator of epoch.

The X_M -axis is directed toward the mean vernal equinox of epoch.

The Z_M axis is directed along the Earth's mean rotational axis of epoch and is positive north.

The Y_M -axis completes a right-handed system.

Characteristics: Inertial, right-handed, Cartesian system.



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- Name:** Body axis coordinate system.
- Origin:** Center of mass.
- Orientation:** X_{BY} -axis is parallel to the Orbiter structural body X_0 -axis; positive toward the nose.
- Z_{BY} -axis is parallel to the Orbiter plane of symmetry and is perpendicular to X_{BY} ; positive down with respect to the Orbiter fuselage.
- Y_{BY} -axis completes the right-handed orthogonal system.
- Characteristics:** Rotating, right-handed, Cartesian system.
- L, M, N : Moments about $X_{BY}, Y_{BY},$ and Z_{BY} axes, respectively.
- p, q, r : Body rates about $X_{BY}, Y_{BY},$ and Z_{BY} axes, respectively.
- $\dot{p}, \dot{q}, \dot{r}$: Angular body acceleration about $X_{BY}, Y_{BY},$ and Z_{BY} axes, respectively.
- The Euler sequence that is commonly associated with this system is a yaw, pitch, and roll sequence where $\Psi =$ yaw, $\theta =$ pitch, and $\phi =$ roll or bank. This attitude sequence is yaw, pitch, and roll around the $Z_{BY}, Y_{BY},$ and X_{BY} axes, respectively.

Appendix B: Lambert/Clohessy – Wiltshire Targeting

The Lambert routine assumes a conic world; i.e., the chaser and target vehicles are orbiting in a central gravitational field with no outside forces acting upon them. Since the world is actually more complex, the transfer orbit computed by Lambert will not be that which is actually desired. Therefore, an iterative method is used to find a transfer that will get the Orbiter to the desired target. Lambert targeting is accurate at larger ranges (> 30 n. mi.) and is time consuming. The onboard computers may require up to 2 minutes to compute the necessary ΔV .

The targeting sequence begins when the crew selects a predefined set of rendezvous maneuvers. The on-orbit predictor coasts the inertial nav states of the chaser, $R_c(t_0)$, and target, $R_t(t_0)$, to the T1 time. T1 is the time of the first maneuver of the set. Once the software has the nav states at T1, $R_c(t_1)$ and $R_t(t_1)$, it will coast the target's nav state to T2 and use target set I-loaded offsets to generate the chaser's desired inertial position at T2, $R_{cd}(t_2)$. T2 is the second maneuver of the set.

The Lambert routine, by conic assumption, now computes the velocity required (V REQ) to get the chaser vehicle from T1 to its desired inertial position at T2. Now we are ready to execute the maneuver at T1. Right? Wrong! The precision coast routing now coasts the chaser's nav state and V REQ to the inertial position R_{arrive} which misses the desired targeted offset. This first computation missed the target because it was conic. The software then measures the missed distance and internally changes the chaser's desired inertial position at T2 to an inertial offset point, R_{offset} , equal to the miss distance. This offset point is an imaginary point where the chaser would go if the world were conic. Since the world is not conic, by aiming for the offset point, the chaser hopefully will go to the desired chaser initial position. This process is then iterated to define the most precise offset point that Lambert should aim for to hit the desired target.

Every 4 seconds the Lambert guidance routine will be called to update the velocity

required and inertial offset. This is important if for some reason the engines do not burn at T1. The Lambert routine will have automatically retargeted a new offset and ΔV . Even after the burn is complete the routine will continue to update its parameters. During the burn the thrust direction will also be updated.

The Lambert routine can also compute the ΔV required at T1 when T1 is desired to be defined in terms of the elevation angle. Usually only the terminal phase initiation (TPI) maneuver uses the elevation angle. Therefore, this technique will probably be used only for TPI.

As Lambert targeting is time consuming, it probably will not be used for close-in PROX OPS maneuver computations. For these maneuvers, a faster method will be used: Clohessy-Wiltshire (C-W) targeting.

The C-W equations involve a simplification of the Lambert equations. They are derived in terms of the relative positions and state of the chaser with respect to a target centered curvilinear coordinate system rather than Lambert's inertial positions. C-W results in a simpler and faster solution (10 to 15 seconds), accurate at small ranges (< 30 min). The accuracy of this method is enhanced by incorporation of a constant orbital drag model.

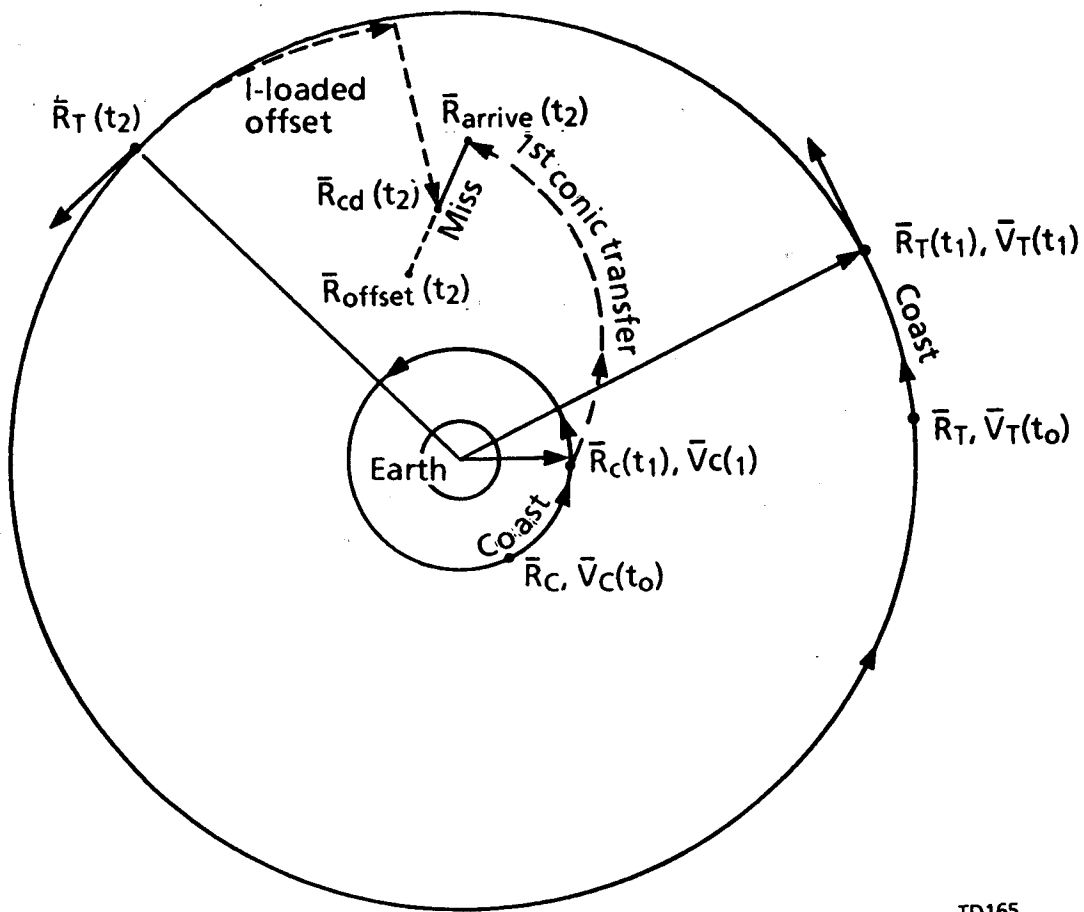
The routine again begins when the crew selects a predefined set of maneuvers. This set will be identified in the software to be targeted by C-W equations. The on-orbit predictor coasts the inertial nav states of both the chaser and target to T1.

From this information, the relative state (X) between the two at T1 is computed. The C-W logic then combines this with a desired relative position at T2, X_2 , and a transfer time to compute the relative velocity change required to move the chaser from X to X_2 , (XD REQ), and hence the T1 maneuver.

A recent addition to the C-W routine is that of an iterator. After computing the T1

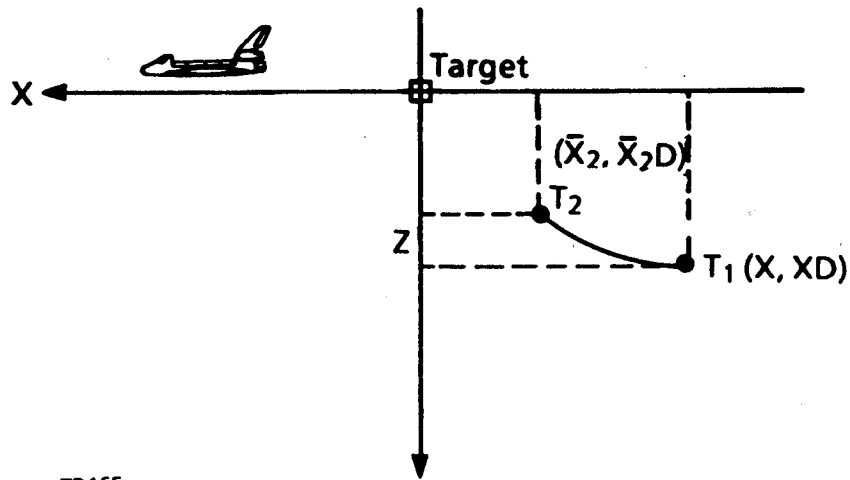
maneuver, the chaser's relative state, X , XD REQ, is converted to the chaser's inertial state, R_c and V REQ. This state is propagated to T_2 and a relative arrival position, X arrive, is computed. The software then measures the miss distance between the C-W targeted X arrive position and X_2 , the desired position. This error is used along with the C-W targeting matrix equation to compute the Δ to the T_1 maneuver necessary to reach the desired T_2 relative position. The routine is then iterated to fine tune the Δ . This iterative process has slowed the C-W technique slightly but has increased its accuracy. The routine still converges very rapidly.

Several other points about C-W targeting are (1) it supports multi-rev targeting, (2) it can not use an elevation angle, (3) the maneuver is executed as an external ΔV , and (4) there is not cyclic retargeting involved.



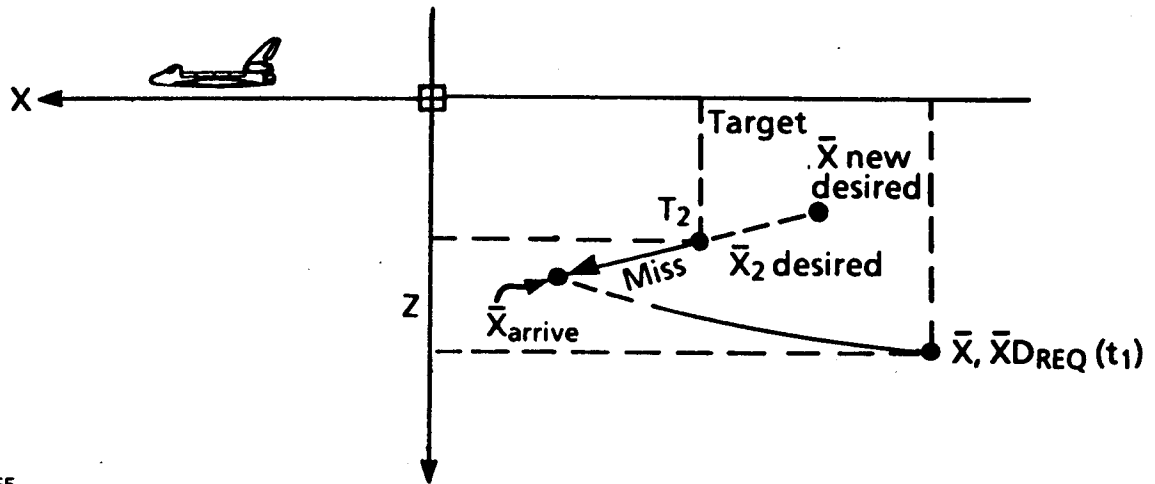
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- | | |
|--|---|
| $R_T(t_1)$ - Inertial position of target at first maneuver time | R_{offset} - Imaginary position Lambert will use as the new aim point. By computing a conic transfer to reach R_{offset} the chaser will actually reach $R_{cd}(t_2)$ |
| $V_T(t_1)$ - Velocity of target at first maneuver time | |
| $R_C(t_1)$ - Inertial position of chaser at first maneuver time | |
| $V_C(t_1)$ - Velocity of chaser at first maneuver time | |
| $R_{cd}(t_2)$ - Desired chaser inertial position at second maneuver time | |
| $R_T(t_2)$ - Inertial position of target at second maneuver time | |
| R_{arrive} - Inertial position chaser will reach at t_2 if initial Lambert conic transfer computation selected. OFF target by a measured miss distance | |



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1. Coast NAV states to T1. R_c, V_c, R_t, V_t
2. Compute relative state at T1: X, XD
3. Compute ΔV required XD_{REQ} , T1 maneuver
4. Convert X, XD_{REQ} to R_c, V_{REQ}
5. Propagate X, XD_{REQ} through transfer time to X_2 at T_2



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6. Get error: $X_{arrive} - X_2$ desired
7. C-W targeting matrix uses miss distance to compute a Δ to the computed T1 maneuver to each X_2 desired
8. Iterate to fine-tune

Appendix C: Abbreviations/Acronyms

ALT	altitude
ACS	attitude control system
ADI	attitude direction indicator
ARCS	aft RCS
AVG G	average G (acceleration)
AZ	azimuth to target
AZ DOT ($\dot{A}Z$)	azimuth rate
bp	barberpole (on a talkback)
BPV	body point vector
CCTV	closed circuit television
CCW	counter clockwise
CDR	commander (crewmember)
CG	center of gravity
COAS	crew optical alignment sight
CRT	cathode ray tube
CUR	current orbit trajectory
CW	clockwise
DAP	digital autopilot
DEC	declination
ΔH	delta height between target orbit and chaser
ΔS	arc distance
ΔV	delta velocity
ΔT	delta time
EL	elevation of target
EL DOT ($\dot{E}L$)	elevation rate
EXEC	execute
FDf	flight data file
FDI	fault detection and identification
FLTR	filtered dffff
FOD	Flight Operations Directorate
FOV	field of view
FRCS	forward RCS
ft	feet
ft/sec	feet per second
GG	gravity gradient stabilization
GN&C	guidance, navigation, and control system
GPC	general purpose computer
H	orbital height (above surface)
H-bar (\bar{H})	angular momentum vector
I-CNECT	OMS-to-RCS interconnect
ICOS	improved COAS
IMU	inertial measurement unit
IUS	inertial upper stage
kft	kilofeet (1000 feet)

LAT	latitude
LDEF	long duration exposure facility
LED	light emitting diode
LON	longitude
LOS	line of sight
LOSA	line-of-sight angle
LVLH	local vertical/local horizontal
MC	midcourse correction maneuvers
MECO	main engine cutoff
MET	mission elapsed time
MM	major mode, software
MMU	manned maneuvering unit
MNVR	maneuver
MPAD	Mission Planning and Analysis Division
MPS	main propulsion system
MS	mission specialist (crewmember)
n. mi.	nautical mile (6076 ft)
nav	navigation
NCC	corrective combination
NH	height adjustment
NLOS	normal to line-of-sight
NSR	normal slow return
OM	omricon
OMS	orbital maneuvering system
OPS	operational sequence, software
ORB	orbital
PAM	payload assist module
pb	pushbutton
pbi	pushbutton indicator
pc	plane change
PL	payload
PLBD	payload bay doors
PLT	pilot (crewmember)
PNL	panel
PRCS	primary RCS
PREL	preliminary
PROP	propagated
PROX OPS	proximity operations phase
R V (R)	radius vector axis
\dot{R}	range to target
R Δ (R)	range rate to target, (+) opening, (-) closing
RCS	radar cross section, reaction control system
RED	radar enhancement device
REL	relative
REL	release (flight software)
REV	revolution about Earth, usually about 90 minutes in low orbit
RHC	rotational hand controller
RMS	remote manipulator system
RNDZ	rendezvous
RR	rendezvous radar
sec	seconds
SEP	separation

SES	Shuttle Engineering Simulator
SMM	Solar Maximum Mission Satellite
SOR	stable orbit rendezvous
SPARTAN	Shuttle pointed autonomous research tool for astronomy (a retrievable free-flier)
SPAS	Shuttle Pallet Satellite (STS-7)
SPEC	special display, software
SRM	solid rocket motor
SSUS	spin-stablized upper stage
STRK	star tracker
STDN	spaceflight tracking and data network
STS	space transportation system
SUM	summary
SYS	system
t	time
tb	talkback
TBD	to be determined
TBS	to be supplied
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TGT	target
THC	translation hand contrbller
TI	terminal initiation
TIG	time of ignition
TPF	terminal phase finalization
TPS	thermal protection system
TRK	track
V-bar (\vec{V})	velocity vector axis
VEL	velocity
VGO	velocity to go
VRCS	vernier reaction control system

14 5 END