Chapter 6

Orbital Mechanics

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Knowledge of orbital motion is essential for a full understanding of space operations. Motion through space can be visualized using the laws described by Johannes Kepler and understood using the laws described by Sir Isaac Newton. Thus, the objectives of this chapter are to provide a conceptual understanding of orbital motion and discuss common terms describing that motion. The chapter is divided into three sections. The first part focuses on the important information regarding satellite orbit types to provide an understanding of the capabilities and limitations of the spaceborne assets supporting the war fighter. The second part covers a brief history of orbital mechanics, providing a detailed description of the Keplerian and Newtonian laws. The third section discusses the application of those laws to determining orbit motion, orbit geometry, and orbital elements. This section has many facts, figures, and equations that may seem overwhelming at times. However, this information is essential to understanding the fundamental concepts of orbital mechanics and provides the necessary foundation to enable war fighters to better appreciate the challenges of operating in the space domain.

Orbit Types

An orbit for a satellite is chosen based on the mission of that particular satellite. For instance, the lower the altitude of a satellite, the better the resolution an onboard camera can have and the shorter the time it takes to travel around the earth (period). On the other hand, the farther out a satellite is, the more of the earth's surface it can observe at one time. Also, the farther the orbit is tilted away from the equator, the more of the earth's surface a satellite will observe over the course of an orbit. These parameters (which will be described in more detail later in the chapter) drive the four basic orbit types: low Earth orbit (LEO), medium Earth orbit (MEO), geosynchronous Earth orbit (GEO), and highly elliptical orbit (HEO). Table 6-1 lists the various orbit types and the missions associated with each one.

Low Earth Orbit Satellites

LEO satellites orbit the earth at an altitude between approximately 100 and 1,000 statute miles (160 to 1,600 km) by the laws of orbits corresponding to periods of about 100 minutes to go around the earth. At these altitudes, onboard sensors have the best resolution, communication systems require the least power to talk to the earth, and rockets require the least energy to get them to orbit. LEO satellites can be divided into three general categories: polar sun-synchronous, polar non–sun-synchronous, and inclined nonpolar.

Table 6-1. Orbit types

Orbit Type	*Mission*	*Altitude*	*Period*	*Tilt*[a]	*Shape*
LEO					
• Polar sun-synchronous	Remote sensing/ weather	~150–900 km	~98–104 min	~98°	circular
• Inclined nonpolar	International Space Station	~340 km	~91 min	~51.6°	circular
• Polar non–sun-synchronous	Earth observing, scientific	~450–600 km	~90–101 min	~80–94°	circular
MEO					
• Semisynchronous	Navigation, communications, space environment	~20,100 km	~12 hours	~55°	circular
GEO					
• Geosynchronous • Geostationary	Communication, early warning, nuclear detection, weather	~35,786 km	~24 hours (23h 56m 04s)	~0°	circular
HEO					
• Molniya	Communications	Varies from ~495 km to ~39,587 km	~12 hours (11h 58m)	63.4°	long ellipse

[a]Orbits roughly stay in the same plane. This indicates the tilt or inclination of this plane relative to the equator. Near zero is along the equator, and near 90° is over the poles. Greater than 90° indicates against the rotation of the earth.

The term *inclined nonpolar orbit* refers to all LEO satellites that are not in near-polar orbits.[1] The inclination of the orbit is equal to the maximum latitude the satellite will pass over. Thus, this type of orbit is used when global coverage of the earth is not needed. The chosen inclination is ordinarily the latitude of the launch site to maximize the amount of energy gained from the rotation of the earth. The International Space Station and space shuttle fall into this orbit category. Figure 6-1 shows an example of an inclined nonpolar orbiting satellite ground track.

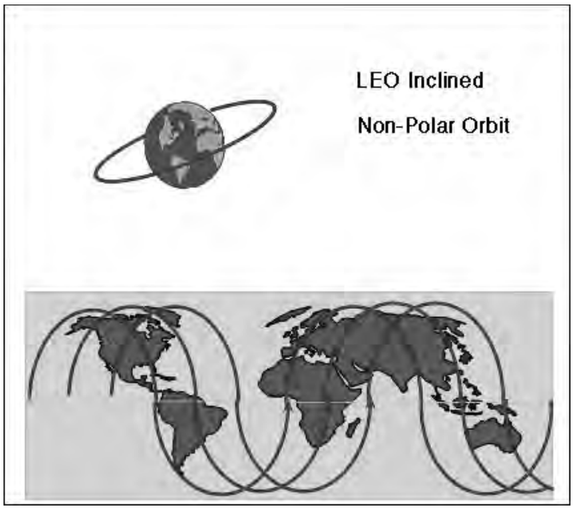

Figure 6-1. Inclined nonpolar orbit. (*Created by* Air Command and Staff College [ACSC])

A polar non–sun-synchronous orbit is like the previous orbit except that the inclination is nearly polar. This type of orbit is used to maximize the coverage of the earth—every latitude will ultimately be passed over, and because of the fast period, a large part of the earth's surface will be seen each day. All the earth's surface will ultimately be overflown. This type of orbit is commonly used for constellations of communication satellites.

One phenomenon affecting a polar, non–sun-synchronous orbit is that, because the earth is not a perfect sphere, the orbit will drift (or precess) over time. If the designers want the orbit to pass over a specific point on the earth at a specific time each day, a polar sun-synchronous orbit is needed. In this type of orbit, a specific altitude and inclination are picked such that the natural orbit precision exactly matches the rate that the earth orbits the sun [(360° per year)/(365.25 days per year) = .986° per day].[2] An example of a polar sun-synchronous satellite orbit and corresponding ground track is shown in figure 6-2.

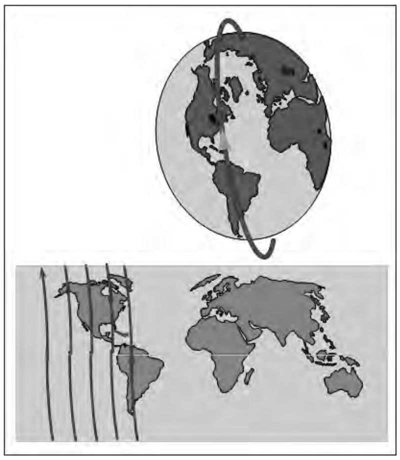

Figure 6-2. Sun-synchronous orbit. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-18.)

Medium Earth Orbit Satellites

MEO satellites orbit the earth at an altitude between approximately 1,000 and 12,000 statute miles (1,600 to 19,300 km), corresponding to periods between 100 minutes and 12 hours. Medium Earth orbits are used to provide longer dwell times over a given region and a larger coverage area of the earth as compared to LEO satellites. In addition, the higher altitude above the earth reduces the effects of atmospheric drag to effectively zero. MEO satellite missions include navigation systems such as GPS.[3]

An example of an MEO satellite, a semisynchronous satellite ground track, can be seen in figure 6-3. This orbit, with an orbital period (the time it takes to make one complete orbit around the earth) of approximately 12 hours, repeats twice a day. Since the earth turns halfway on its axis during each complete orbit, the points where the sinusoidal ground tracks cross the equator coincide pass after pass, and the ground tracks repeat each day as shown. This predictability is very helpful for ground stations monitoring the satellite.

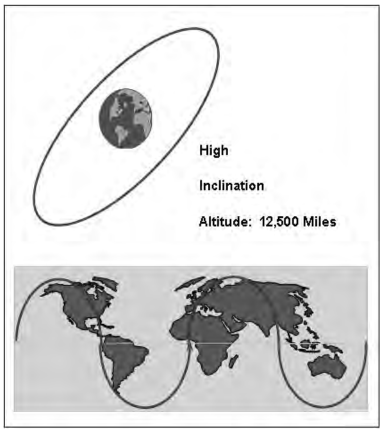

Figure 6-3. Semisynchronous orbit. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-18.)

Geosynchronous Earth Orbit Satellites

GEO satellites orbit the earth at an altitude of 22,236 statute miles (35,786 km). At this altitude, a satellite in a circular orbit and zero inclination will have an orbital period equal to the earth's rotational period (approximately 24 hours). This allows a satellite to remain relatively fixed over a particular point on the earth's surface. At an altitude of 22,236 miles, one geosynchronous satellite has a commanding field of view of almost one-third of the earth's surface from approximately 75° south latitude to approximately 75° north latitude.[4] Therefore, geosynchronous orbits are desirable for communications and early warning systems. However, this altitude and inclination are the most difficult to achieve, especially for nations without an equatorial launch site.

The terms *geosynchronous* and *geostationary* have been used interchangeably, but there is a distinct difference between the two. *Geosynchronous* refers to a satellite with a 24-hour period, regardless of inclination. *Geostationary* refers to a satellite with a 24-hour period, in a near-circular orbit, with an inclination of approximately zero. It appears to hover over a spot on the equator as shown in figure 6-4. All geostationary orbits must be geosynchronous, but not all geosynchronous orbits are necessarily geostationary.[5] An example of a nongeostationary satellite would be the *Syncom 2*, launched in 1963 into a geosynchronous orbit with a 33° inclination.[6]

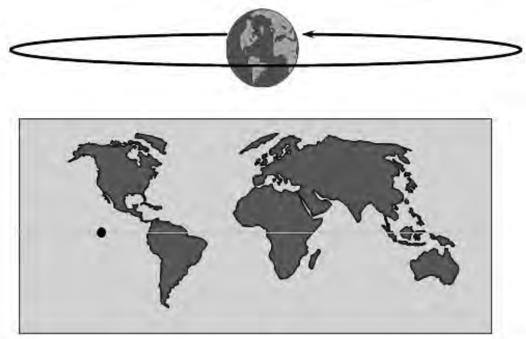

Figure 6-4. Geostationary orbit/ground track. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-17.)

Now take the same orbit and give it an inclination of 30°. The period and orbit shape remain the same. The ground trace will retrace itself with every orbit, in this case in a figure-eight pattern. The ground trace will also vary between 30° north and 30° south latitude due to its 30° inclination. In another example, if the geostationary satellite has an eccentricity near zero and an inclination of 60°, the ground trace would follow a similar, larger figure-eight path between 60° north and 60° south latitude as shown in figure 6-5.

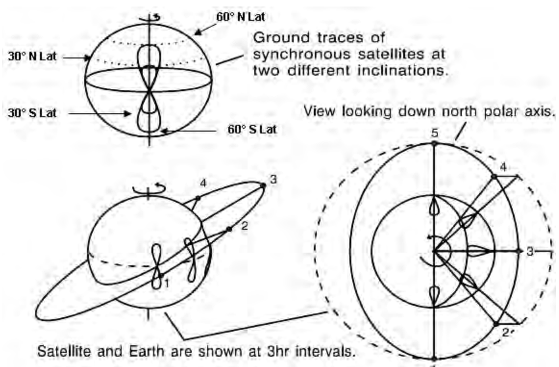

Figure 6-5. Ground traces of inclined, circular, synchronous satellites. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-17.)

Highly Elliptical Orbit Satellites

All the orbits discussed thus far have been circular. However, orbits can also take on an elliptical shape. HEO satellites are the most common noncircular orbits, and they orbit the earth at altitudes which vary between approximately 660 and 24,000 statute miles (1,060 and 38,624 km) in a single period.[7] Satellites travel faster the closer they are to the earth, so HEO orbits enable long dwell times as well as large fields of view when at their farthest points from the earth (apogee). They are primarily used for communications, scientific research, and intelligence, surveillance, and reconnaissance (ISR) missions when GEO orbits are inaccessible.

The most popular highly elliptical orbit is the "Molniya" orbit, named after the Russian word for lightning to describe the speed at which a satellite in this particular orbit travels through its closest point of approach (perigee).[8] Figure 6-6 shows a typical Molniya orbit that might be used for northern hemispheric communications.

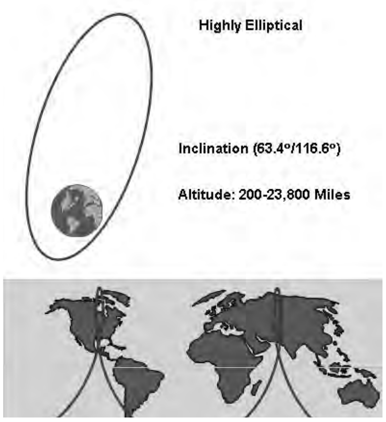

Figure 6-6. Molniya orbit. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-18.)

With an orbital period of 12 hours, the ground track retraces itself every day, just like the medium Earth, semisynchronous orbit of GPS.

A History of the Laws of Motion

The modern orbit types have been developed based on theories dating back centuries. The early Greeks initiated the orbital theories, postulating that the earth was fixed, with the planets and other celestial bodies moving around it—a geocentric universe.[9] About 300 BC, Aristarchus of Samos suggested that the sun was fixed and the planets, including Earth, were in circular orbits around the sun—a heliocentric universe.[10] Although Aristarchus was more correct (at least about a heliocentric solar system), his ideas were too revolutionary for the time. Other prominent astronomers/philosophers were held in higher esteem, and since they favored the geocentric theory, Aristarchus's heliocentric theory was rejected, and the geocentric theory continued to be predominately accepted for many centuries.

In the year 1543, some 1,800 years after Aristarchus proposed a heliocentric system, a Polish monk named Nicolas Koppernias (better known by his Latin name, Copernicus) revived the heliocentric theory when he published *De Revolutionibus Orbium Coelestium* (*On the Revolutions of the Celestial Spheres*). This work represented an advance, but there were still some inaccuracies. For example, Copernicus thought that the orbital paths of all planets were circles around the center of the sun.[11]

Tycho Brahe established an astronomical observatory on the island of Hven in 1576. For 20 years, he and his assistants carried out the most complete and accurate astronomical observations of the period. However, Brahe did not accept Copernicus's heliocentric theory and instead believed in a geo-heliocentric model that had the moon and sun revolving around the earth while the rest of the celestial bodies revolved around the sun.[12]

German astronomer Johannes Kepler, born in 1571, wondered why there were only six planets and what determined their separation. His theories required data from observations of the planets, and he realized that the best way to acquire such data was to become Brahe's assistant.

In 1600, Brahe set Kepler to work on the motion of Mars. This task was particularly difficult because Mars's orbit was the second most eccentric (of the then-known planets) and defied the circular explanation. After Brahe's death in 1601, Kepler finally discovered that Mars's orbit (and that of all planets) was represented by an ellipse with the sun at one of its foci.[13]

Kepler's Laws of Planetary Motion

Kepler's discovery of Mars's elliptical orbit led to another discovery—the first of his three laws of planetary motion, which describe the orbit of the planets around the sun.

Kepler's First Law (Law of Ellipses). *The orbits of the planets are ellipses with the sun at one focus.*[14] Figure 6-7 shows an ellipse where O^1 is one focus and *O* is the other. This depiction illustrates that, by definition, an ellipse is a closed curve such that the sum of the distances (*R1* and *R2*) from any point (*P*) on the curve to the two foci (O^1 and *O*) remains constant.[15]

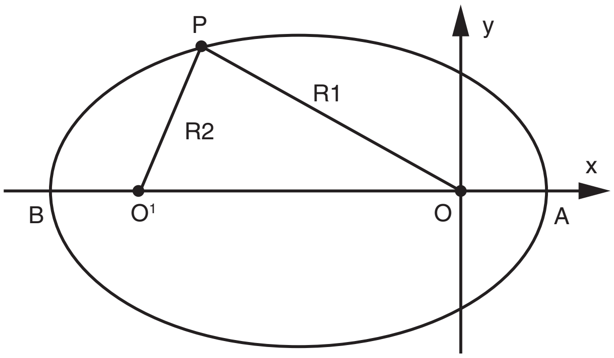

Figure 6-7. **Kepler's first law**. (*Created by* ACSC)

The maximum diameter of an ellipse is called its major axis; the minimum diameter is the minor axis. The size of an ellipse depends in part upon the length of its major axis. The shape of an ellipse is denoted by eccentricity (*e*), which is the ratio of the distance between the foci to the length of the major axis (see the orbit geometry section in this chapter).

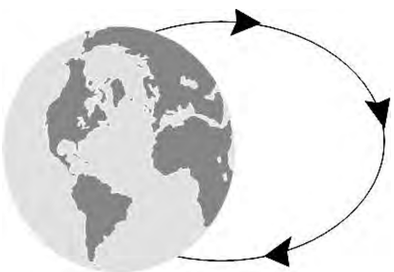

Figure 6-8. Ballistic missile path. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-5.)

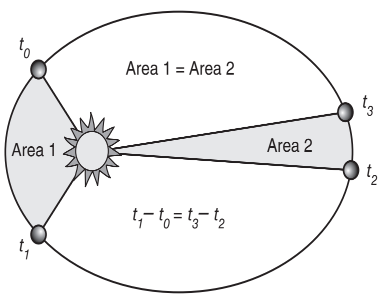

Figure 6-9. Kepler's second law. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-6.)

The paths of ballistic missiles (not including the powered and reentry portion) are also ellipses; however, they happen to intersect the earth's surface (as shown in fig. 6-8).

Kepler's Second Law (Law of Equal Areas). *The line joining the planet to the sun sweeps out equal areas in equal times.*[16] Based on his observation, Kepler reasoned that a planet's speed depended on its distance from the sun.

Kepler's second law is easy to visualize in figure 6-9, where t_0, t_1, and so forth indicate time. If the object in figure 6-9 were in a circular orbit (versus the elliptical orbit shown), its speed and radius would both remain constant, and therefore, over a given interval of time the "shape" of area 1 and area 2 would be identical. It is also apparent from figure 6-9 that the closer a planet is to the sun along the elliptical orbit, the faster it travels. The same principle applies to satellites orbiting the earth, as especially noted in the Molniya orbit discussed earlier.

Kepler's Third Law (Law of Harmonics). *The square of the orbital period of a planet is proportional to the cube of the mean distance from the sun.*[17] Kepler's third law directly relates the square of the period to the cube of the mean distance for orbiting objects. By this law, the altitude of a circular orbit uniquely determines how long it will take to travel around the earth and vice versa.[18] Thus, geostationary orbits, which must have a period of 24 hours, must be at an altitude of 24,000 miles. LEO satellites likewise cannot hover over a spot on the earth.

Newton's Laws of Motion

The laws Kepler developed describe very well the observed motions of the planets, but they made no attempt to describe the forces behind those laws. The laws regarding those forces would be key to ultimately developing artificial satellites. This work was formulated by Sir Isaac Newton.

In 1665, an outbreak of the plague forced the University of Cambridge to close for two years. During those two years, the 23-year-old genius Isaac Newton conceived the law of gravitation, the laws of motion, and the fundamental concepts of differential calculus. Twenty years later the result appeared in *The Mathematical Principles of Natural Philosophy*, or simply the *Principia*,[19] which formulated a grand view that was consistent and capable of describing and unifying the mundane motion of a falling apple and the motion of the planets.

Newton's First Law (Inertia). *Every body continues in a state of rest, or of uniform motion in a straight line, unless it is compelled to change that state by a force imposed upon it.*[20] Newton's first law describes undisturbed motion. Inertia is the resistance of mass to changes in its motion.

Newton's Second Law (Changing Momentum). *When a force is applied to a body, the time rate of change of momentum is proportional to, and in the direction of, the applied force.* Newton's second law describes how motion changes. It is important to define momentum before describing the second law. Momentum is a measure of an object's motion. Momentum ($\boldsymbol{p}$) is a vector quantity (denoted by boldface type) defined as the product of an object's mass (m) and its relative velocity ($\boldsymbol{v}$).

$$\boxed{\boldsymbol{p} = m\boldsymbol{v}}$$

If there is a change in momentum ($\Delta\boldsymbol{p}$), assuming the mass of the object remains the same, then there must be a change in velocity ($\Delta\boldsymbol{v}$) of the object as well. As a result, we have the following equation:

$$\boxed{\Delta\boldsymbol{p} = m\Delta\boldsymbol{v}}$$

Force ($\boldsymbol{F}$) is defined as the time rate of change of an object's momentum.

$$\boxed{\boldsymbol{F} = \frac{\Delta\boldsymbol{p}}{\Delta t} = \frac{m\Delta\boldsymbol{v}}{\Delta t}}$$

Acceleration ($\boldsymbol{a}$) is defined as the change in velocity over time ($\Delta\boldsymbol{v}/\Delta t$). As a result, this second law becomes Newton's famous equation:

$$\boxed{\boldsymbol{F} = m\boldsymbol{a}}$$

Newton's Third Law (Action-Reaction). *For every action there is a reaction that is equal in magnitude but opposite in direction to the action.*[21] This law hints at conservation of momentum. If forces are always balanced, then the objects experiencing the opposed forces will change their momentum in opposite directions and equal amounts.

Newton's Law of Universal Gravitation. *Every particle in the universe attracts every other particle with a force that is proportional to the product of the masses and inversely proportional to the square of the distance between the particles.*[22]

$$\boxed{F_g = G\left(\frac{M_1 m_2}{R^2}\right)}$$

In the above equation, F_g is the force due to gravity, G is the universal gravitational constant with a set value of 6.67259×10^{-11} $m^3kg^{-1}s^{-2}$, M_1 and m_2 are the masses of the central body (the earth, for example) and orbiting bodies, and R is the distance between the centers of the two bodies.[23] This law, in association with the second law, allows scientists and engineers to connect the forces applied (such as gravity) to the acceleration. When the position and velocity are known, the gravity force can be calculated. Knowing the gravity acceleration will change the position and velocity. Plotting the altitude over time for a satellite yields an orbit.

In this way, engineers can also calculate the necessary orbit velocities and the subsequent amounts of force necessary to launch a satellite into space.[24] The force (F) required will determine the type of booster (Delta IV, Delta II, space shuttle, etc.) that is selected to launch the satellite.

Once the satellite is at the right spot (position) going a certain speed (velocity), the orbit will be established and predictable using the laws above. The solutions to the equations above also match Kepler's observations of the planets, thus establishing that satellites would move the same way. However, with additional velocity, satellites do not have to travel only in ellipses; they can also travel on parabolas or hyperbolas. This knowledge is key to understanding interplanetary travel.

Orbital Motion

So what is the velocity and position a body needs to get into orbit? According to Newton's second law, for a body to change its motion a force must be imposed upon it. An example is playing catch—when a ball is thrown or caught, its motion is altered. Thus, gravity is compensated for by throwing the ball upward by some angle allowing gravity to pull it down, resulting in an arc. When the ball leaves the hand, it starts accelerating toward the ground according to Newton's laws (at sea level on the earth the acceleration is approximately 9.8 meters per second [m/s] or 32 feet [ft.] per second straight down).[25] If the ball is initially motionless, it will fall straight down. However, if the ball has some horizontal motion, it will continue in that motion while accelerating toward the ground. Figure 6-10 shows a ball released with varying lateral (or horizontal) velocities.

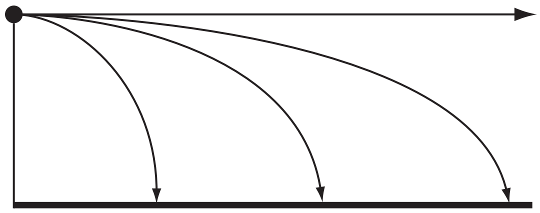

Figure 6-10. Newton's second law. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-9.)

In figure 6-10, if the initial height of the ball is approximately 4.9 meters (16.1 ft.) above the ground, then at sea level, it would take one second for the ball to hit the ground. How far the ball travels along the ground in that one second depends on its horizontal velocity (table 6-2).

Eventually one would come to the point where the earth's surface drops away as fast as the ball drops toward it. As figure 6-11 depicts, the earth's surface curves down about five meters for every eight kilometers.[26]

Table 6-2. Gravitational effects.

Horizontal velocity (m/s)[a]	*Distance travelled in one second (m)*	
	Vertical	*Horizontal*
1	4.9	1
2	4.9	2
4	4.9	4
8	4.9	8
16	4.9	16

[a] All values are in meters and meters per second.

At the earth's surface (without accounting for the atmosphere, mountains, or other structures), a satellite would have to travel at approximately 8 km/second (km/sec) (or about 17,900 mph) to fall around the earth without hitting the surface. In other words, the satellite would have to travel 17,900 mph to remain in orbit at the earth's surface (at a height of approximately zero). This is fundamentally what it means to be in orbit—travelling fast enough forward that by the time the orbiting body would ordinarily hit the ground, the earth will have curved away from the body.

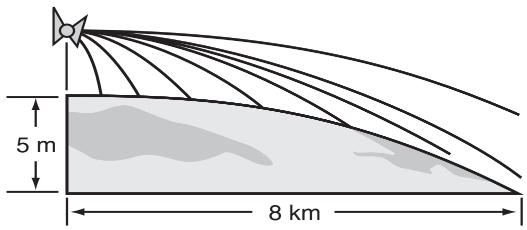

Figure 6-11. Earth's curvature. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-10.)

However, the earth does have an atmosphere, and to stay in a relatively stable orbit, a satellite has to be positioned at an orbital height above the denser parts of the earth's atmosphere. The minimum height is approximately 150 km (about 93 miles) above the earth's surface. To remain in orbit at this height, a satellite must travel at 7.8 km/sec (or 17,500 mph).[27] At this speed, the orbital period of the satellite would be 87.5 minutes. A period of less than 87.5 minutes indicates that the object is “decaying” due to the effects of atmospheric drag and will eventually reenter the earth’s atmosphere and fall back to Earth. This would ordinarily cause the object to burn up. Thus, the job of a rocket is to carry a satellite above the main part of the atmosphere and then get it travelling at the right speed. Once the satellite is released, it is in orbit.

At higher altitudes, the speed needed to maintain an orbit is less, much like the speed needed to keep a ball on the end of a string horizontal. Figure 6-12 shows how differing velocities affect a satellite’s trajectory or orbital path. The figure depicts a satellite at an altitude of one Earth radius (6,378 km above the earth’s surface). At this distance, a satellite would have to travel at 5.59 km/sec (12,500 mph) to maintain a circular orbit, and this speed is known as the satellite’s circular velocity for this altitude. As the satellite’s speed increases, it moves away from the earth, and its trajectory becomes an elongating ellipse until the speed reaches 7.91 km/sec (17,700 mph). At this speed and altitude the satellite has enough energy to leave the earth’s gravity and never return. Its trajectory has now become a parabola, and this velocity is known as its escape velocity for this altitude.[28] The equations for circular velocity (v_c) and escape velocity (v_e) are as follows:

$$v_c = \sqrt{\frac{GM_E}{r}} \qquad v_e = \sqrt{\frac{2GM_E}{r}}$$

In these equations, G is the gravitational constant (6.67259 x 10^{-11} $m^3kg^{-1}s^{-2}$), M_E is the mass of the earth (approximately 5.977 x 10^{24} kilograms [kg]), and r is the distance of the satellite from the center of the earth (i.e., the altitude plus 6,378 km).

As an example, from a low Earth orbit of 161 km (100 miles), the escape velocity becomes 11.2 km/sec (25,050 mph). In figure 6-12, the two specific velocities (5.59 km/sec and 7.91 km/sec) correspond to the circular and escape velocities for the specific altitude of one Earth radius (6,378 km).

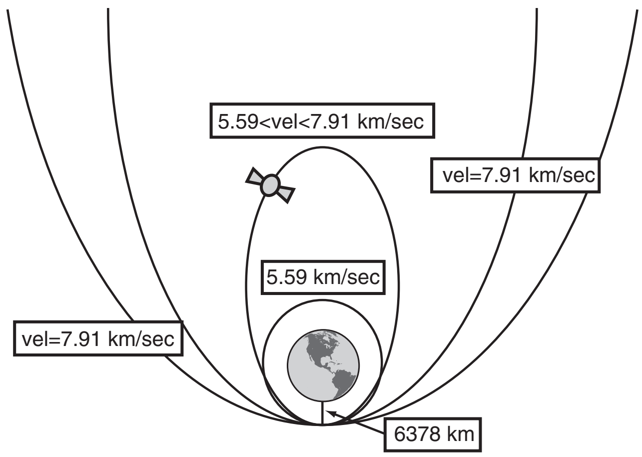

Figure 6-12. Velocity versus trajectory. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-10.)

Constants of Orbital Motion: Momentum and Energy

For a satellite, if you know the position and velocity when the satellite is released from the rocket, you can use Newton's laws to plot out the long-term trajectory. However, to compare orbits, it is helpful to have some common parameters to describe them (like altitude and eccentricity, as described above). This section explores how to find some of those constants to help visualize an orbit.

For a system such as a satellite affected only by gravity (i.e., no drag or thrust), some basic properties remain constant or fixed; that is, they are conserved. Energy and momentum are two such properties which are conservative in such a closed system.

Momentum. Linear momentum is the product of mass times velocity, as discussed previously in Newton's second law. For rotating or spinning bodies, such as a satellite orbiting the earth, a second form of this law is formulated to describe motion in angular terms. Angular momentum ($\vec{H}$) is the product of the linear momentum of an object (i.e., satellite) times the object's position from the center of rotation (the center of the earth): $\vec{H} = m(\vec{r} \times \vec{v})$.[29] This property remains constant for orbiting objects which are not torqued.

In an elliptical orbit, the radius (R) is constantly varying. Thus, for angular momentum to be conserved, the orbital speed must change. Hence, there is greater velocity at perigee than at apogee. Also, since the direction of the angular momentum is also conserved, the plane formed by the rotating object is fixed. Thus, unless an orbit is torque, the orbit plane will not drift through space.

Energy. A system's mechanical energy can also be conserved. Total mechanical energy (E) is derived from an object's position and motion and is usually depicted as the sum of kinetic energy (KE) and gravitational potential energy (PE):[30]

$$\boxed{E = KE + PE}$$

Kinetic energy is the energy associated with an object's motion, and gravitational potential energy is the energy associated with an object's position. Potential energy is measured relative to the center of the earth (hence, it is not the "mgh" you may have learned in high school). Potential energy is the mass of an object (m_1) times the earth's gravitational acceleration (M_2G) over the height above the earth's center. Kinetic energy (KE) is expressed as one-half an object's mass times the square of the object's velocity.[31] These equations are expressed as follows:

$$\boxed{KE = \frac{1}{2}mv^2 \qquad PE = m_1M_2G/R}$$

The Law of Conservation of Energy in its simplest form states that, under the premise that energy cannot be created or destroyed, the sum of all energies (in this case total mechanical energy [E]) in a particular system remains constant unless energy is added (such as by thrust) or taken away (such as by drag).[32] Therefore, any increase in kinetic energy will result in a proportional decrease in gravitational potential energy since the value of total mechanical energy (E) will not change.

Hence, in a circular orbit where the radius remains constant, so will the velocity, as both gravitational potential and kinetic energy remain constant. In all other orbits (elliptical, parabolic, and hyperbolic), the “radius” and speed both change, and therefore, so do both the gravitational potential and kinetic energies in such a way that the total mechanical energy of the system remains constant. Again, for an elliptical orbit, this results in greater velocity at perigee than apogee.

Orbit Geometry

When Newton’s second law is combined with his gravitational law, the solutions are all conic sections, which are shapes that can be made by slicing off sections of a cone at various angles. The conic section an object will follow depends on its kinetic and potential energy as described above. Conic sections consist of four types: circular, elliptical, parabolic, and hyperbolic. If an object lacks the velocity (insufficient kinetic energy, $KE < PE$) to overcome the earth’s gravitational attraction, then it will follow a closed-path orbit in the form of a circle or ellipse. However, if the object has enough velocity (kinetic energy equal in magnitude to the gravitational potential energy in the absence of friction resistance, $KE = PE$) to overcome the earth’s gravitational attraction, then the object will follow an open path in the shape of a parabolic orbit. Finally, if the object has excess velocity (more than sufficient kinetic energy, $KE > PE$) to overcome the earth’s gravitational attraction, then the object will follow an open path in the shape of a hyperbolic orbit.[33] Figure 6-13 shows a three-dimensional representation of the various possible conic sections (orbit geometries).

Figure 6-14 shows a two-dimensional representation of the conic section geometry. The parameters that describe the size and shape of the conic are its semimajor axis (a) and eccentricity (e). The semimajor axis, a measure of the orbit’s size, is half the distance between perigee and apogee; it is also the average distance from the attracting body’s center. Eccentricity, which describes the orbit’s shape, is the ratio of the linear eccentricity (c) to the semimajor axis. The linear eccentricity is half the distance between the two foci.

These parameters apply to all trajectories. A circular orbit is a special case of the elliptical orbit where the foci coincide ($c = 0$). Figure 6-15 depicts a satellite orbit with additional parameters whose conic section is an ellipse.

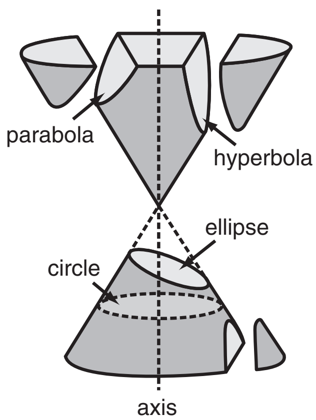

Figure 6-13. Conic sections. (*Reprinted from* David P. Stern, “Kepler’s Three Laws of Planetary Motion: An Overview for Science Teachers,” http://www.phy6.org/stargaze/Kep3laws.htm [accessed 18 April 2008].)

Coordinate Reference Systems and Orbital Elements

All positions and velocities have to be measured with respect to a fixed frame of reference. Many such frames exist—which is used depends on the situation and the nature of the knowledge to be retrieved. Table 6-3 lists several common coordinate reference systems that are used for space applications.[34] For describing the orbit itself, the Earth-centered inertial (ECI) system is used, while the other two describe how the satellite is oriented within that frame.

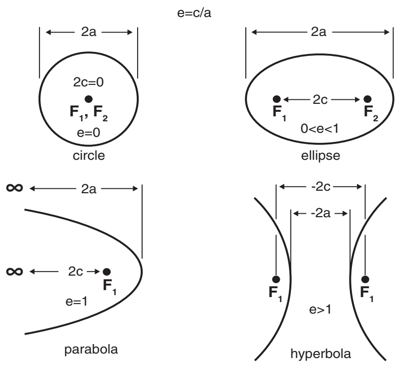

Figure 6-14. Conic section geometry. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-11.)

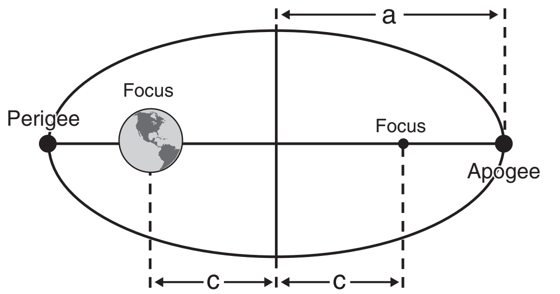

Figure 6-15. Elliptical geometry. (*Adapted from* Air University, *Space Primer***,** unpublished book, 2003, 8-11.)

Table 6-3. Coordinate reference systems.

Coordinate Name	*Fixed with Respect to*	*Center*	*Z-axis or Pole*	*X-axis or Reference Direction*	*Applications*
Earth-centered inertial (ECI)	Inertial space	Earth	Celestial pole	Vernal equinox (J2000.0 reference frame)	Orbit analysis, astronomy, inertial motion
Spacecraft-fixed	Spacecraft	Defined by engineering drawings	Spacecraft axis toward nadir	Spacecraft axis in direction of velocity vector	Position and orientation of spacecraft instruments
Roll, pitch, yaw	Orbit	Spacecraft	Nadir	Perpendicular to nadir toward velocity vector	Earth observation attitude maneuvers

Adapted from Wiley J. Larson and James R. Wertz, ed., *Spacecraft Mission Analysis and Design*, 3rd ed. (El Segundo, CA: Microcosm Press, 1999), 96.

In three-dimensional space, the position and velocity each have three components in each dimension. Therefore, any element set defining a satellite's orbital motion contains at least six parameters to fully describe that motion. The Keplerian, or classical, element set is useful for space operations and tells us four attributes of orbits: orbit size, orbit shape, orientation (to include orbital plane in space and orbit within plane), and location of the satellite at any point in time during its orbit. The most popular Keplerian element set format is the two-line element (TLE) set, which will be discussed later in this chapter.

Orbit Size. The orbit size is described by the semimajor axis (a)—half the distance between apogee and perigee on the ellipse.

Orbit Shape. Eccentricity (e) measures the shape of an orbit. Recall from the discussion of orbit geometry above that eccentricity is a ratio of the foci separation (linear eccentricity [c]) to the size (semimajor axis [a]) of the orbit.

$$\boxed{e = c/a}$$

Size and shape relate to orbit geometry and tell what the orbit looks like. The other orbital elements deal with orientation of the orbit relative to a fixed point in space. With energy being conserved, both e and a are constant.

Orientation. The first angle used to orient the orbital plane is inclination (i)—a measurement of the orbital plane's tilt relative to the equatorial plane. It is measured counterclockwise at the point at which an object crosses the equatorial plane traveling north in its orbit (the ascending node) while looking toward Earth as shown in figure 6-16.[35]

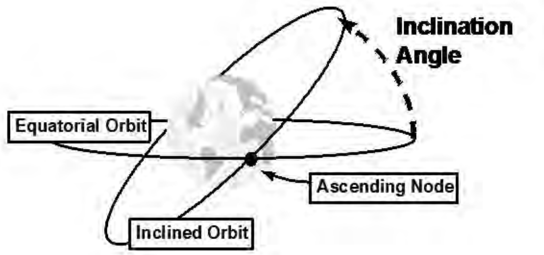

Figure 6-16. Inclination tilt. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-13.)

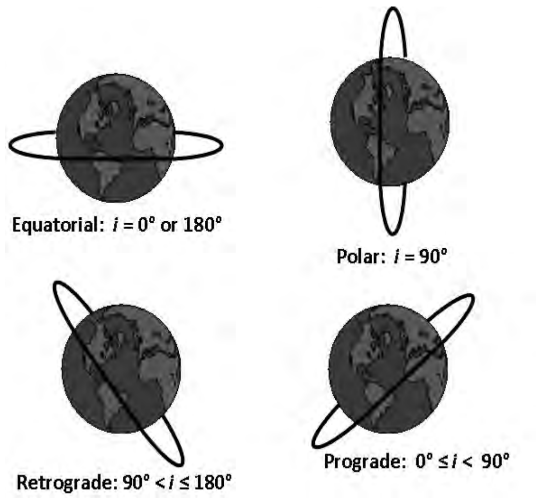

Figure 6-17. **Orbital inclination types**. (*Created by* ACSC)

Inclination is utilized to define several general classes of orbits as shown in figure 6-17. Orbits with inclinations equal to 0° or 180° are equatorial orbits, because the orbital plane is contained within the equatorial plane. If an orbit has an inclination of 90°, it is a polar orbit, because it travels over the poles. If $0° \leq i < 90°$, the satellite orbits in the same direction as the earth's rotation (orbiting eastward around the earth) and is called a prograde orbit. If $90° < i \leq 180°$, the satellite orbits in the opposite direction of the earth's rotation (orbiting westward about the earth) and is in a retrograde orbit.

The second measure used to orient the orbital plane is the right ascension of the ascending node (Ω—uppercase Greek letter omega). It measures where the ascending node is relative to a reference line within the ECI coordinate system eastward to the ascending node ($0° \leq \Omega \leq 360°$) as shown in figure 6-18.[36] It is mostly used to space out constellations of similar satellites.

The reference line is established by drawing a line from the center of the sun through the center of the earth and extending out into space as the earth crosses the sun's equatorial (ecliptic) plane.[37] These crossings occur twice a year and are called the vernal or autumnal equinox (the first day of spring or fall). For astronomical purposes we use the spring or vernal equinox to establish our reference point. When first established as the reference point, this line pointed to the constellation Aries, hence the name "first point of Aries" (fig. 6-19).[38]

Argument of Perigee. Inclination and right ascension fix the orbital plane in space. The orbit must also be fixed within the orbital plane. For elliptical orbits, the perigee is the reference point in the orbit. The argument of perigee (ω—lowercase Greek letter omega) is used, and it is the angle within the orbital plane from the ascending node to perigee in the direction of satellite motion ($0° \leq \omega \leq 360°$) (fig. 6-20).[39]

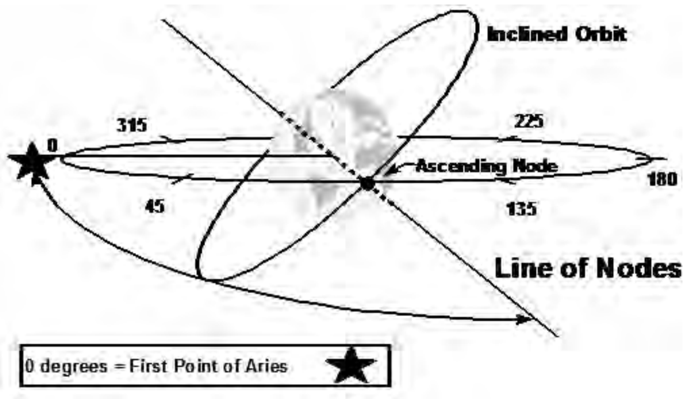

Figure 6-18. Right ascension of the ascending node. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-11.)

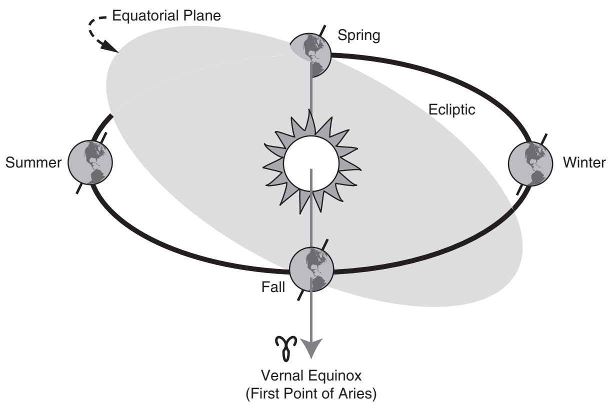

Figure 6-19. Vernal equinox. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-14.)

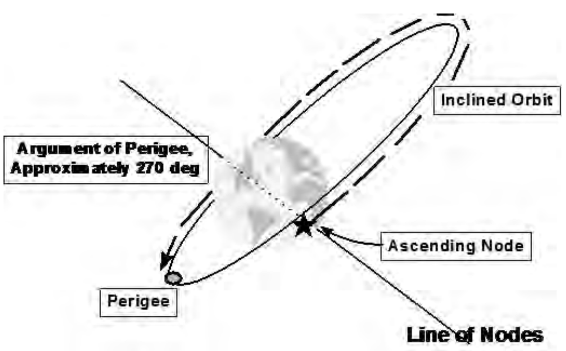

Figure 6-20. Argument of perigee. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-13.)

True Anomaly. At this point all the orbital parameters needed to visualize the orbit in space have been specified. In fact, due to conservation of momentum and energy, the parameters are all constant unless the orbit is perturbed. The final step is to locate the satellite within its orbit. True anomaly (ν—lowercase Greek letter nu) is an angular measurement that describes where the satellite is in its orbit at a specified time. It is measured within the orbital plane from perigee to the satellite's position in the direction of motion ($0° \leq \nu \leq 360°$).[40]

True anomaly locates the satellite with respect to time and is the only orbital element that changes with time.[41] The true anomaly cannot be defined in cases where the eccentricity is exactly zero (perfectly circular orbit) since there would be no perigee from which to measure. Likewise, the argument of perigee is undefined for a circular orbit (which has no perigee), and the right ascension of the ascending node is undefined for an equatorial orbit (which never crosses the equator).

Table 6-4 summarizes the Keplerian orbital element set and orbit geometry and its relationship to the earth.[42]

Table 6-4. Classical orbital elements.

Element	*Name*	*Description*	*Definition*	*Remarks*
a	semimajor axis	orbit *size*	half the long axis of the ellipse	orbital period and energy depend on orbit size
e	eccentricity	orbit *shape*	ratio of half the foci separation (*c*) to the semimajor axis (*a*)	closed orbits: $0 \le e < 1$ open orbits: $e \ge 1$
i	inclination	orbital plane's *tilt*	angle between the orbital plane and equatorial plane, measured counterclockwise at the ascending node	equatorial: $i = 0°$ or $180°$ prograde: $0° \le i < 90°$ polar: $i = 90°$ retrograde: $90° < i \le 180°$
Ω	right ascension of the ascending node	orbital plane's *rotation* about the earth	angle, measured eastward, from the vernal equinox to the ascending node	$0° \le \Omega < 360°$ undefined when $i = 0°$ or $180°$ (equatorial orbit)
ω	argument of perigee	orbit's *orientation* in the orbital plane	angle, measured in the direction of satellite motion, from the ascending node to perigee	$0° \le \omega < 360°$ undefined when $i = 0°$ or $180°$, or $e = 0$ (circular orbit)
ν	true anomaly	satellite's *location* in its orbit	angle, measured in the direction of satellite motion, from perigee to the satellite's location	$0° \le \nu < 360°$ undefined when $e = 0$ (circular orbit)

Two-Line Element Sets

The way the orbital elements are usually presented to space personnel is through the TLE set. It is used by agencies such as NASA and USSTRATCOM to describe the location of satellites orbiting the earth. The two-line element set actually has three lines. The first line is reserved for the satellite's name.[43] The next two lines in essence describe the "address" of the satellite (fig. 6-21). The components of the two-line element set are defined by NASA as follows:[44]

Name of Satellite (*NOAA 6*). This is simply the name associated with the satellite. *NOAA 6* is a weather satellite operated by the National Oceanic and Atmospheric Administration.

International Designator (84 123A). The *84* indicates that the launch year was 1984. The *123* indicates that this launch was the 123rd of the year and *A* shows it was the first object resulting from this launch.

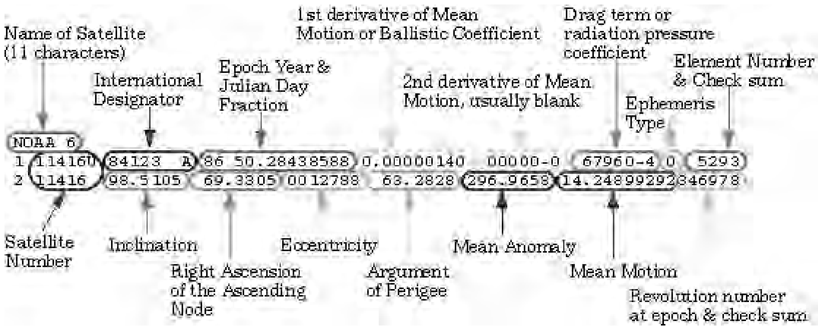

Figure 6-21. TLE set format. (*Reprinted from* NASA, "Definition of Two-Line Element Set Coordinate System," Human Space Flight Web site, http://spaceflight.nasa.gov/realdata/sightings/SSapplications/Post/JavaSSOP/SSOP_Help/tle_def.html [accessed 18 April 2008]).

Epoch Date and Julian Date Fraction (86 50.28438588). The Julian date fraction is just the number of days passed in the particular year. For example, the date above shows *86* as the epoch year (1986), and the Julian date fraction of 50.28438588 means a little over 50 days after 1 January 1986. The resulting time of the vector would be 1986/050:06:49:30.94, computed as follows:

Start with 50.28438588 days (days = 50)

50.28438588 days - 50 = 0.28438588 days

0.28438588 days x 24 hours/day = 6.8253 hours (hours = 6)

6.8253 hours - 6 = 0.8253 hours

0.8253 hours x 60 minutes/hour = 49.5157 minutes (minutes = 49)

49.5157 - 49 = 0.5157 minutes

0.5157 minutes x 60 seconds/minute = 30.94 seconds (seconds = 30.94)

Ballistic Coefficient (0.00000140). Also called the first derivative of mean motion, the ballistic coefficient is the daily rate of change in the number of revolutions (revs) the object completes each day, divided by two. Units are revs/day. This is a "catch all" term used in the Simplified General Perturbations (SGP4) USSTRATCOM predictor to represent the atmospheric drag slowing down a satellite. Mean motion is the average angular rate of a satellite, reflecting that any satellite with a distinct apogee and perigee would change speeds over the course of an orbit. For a circular orbit, the ballistic coefficient would be a constant.

Second Derivative of Mean Motion (00000-0 = 0.00000). The second derivative of mean motion is a second-order drag term in the SGP4 predictor used to model terminal orbit decay. It measures the second time derivative in daily mean motion, divided by six. Units are revs/day^3. A leading decimal must be applied to this value. The last two characters define an applicable power of 10 (12345-5 = 0.0000012345).

Drag Term (67960-4 = 0.000067960). Also called the radiation pressure coefficient (or BSTAR), the parameter is another drag term in the SGP4 predictor. Units are Earth radii^-1. The last two characters define an applicable power of 10. Do not confuse this parameter with "B-Term," the USSTRATCOM special perturbations factor of drag coefficient, multiplied by reference area, divided by weight.

Element Set Number and Check Sum (5293). The element set number is a running count of all TLE sets generated by USSTRATCOM for this object (in this example, 529). Since multiple agencies perform this function, numbers are skipped on occasion to avoid ambiguities. The counter should always increase with time until it exceeds 999, when it reverts to one. The last number of the line is the check sum of line one. A check sum (or checksum) is simply a value used in computer programming to verify the validity of the information contained within that particular line of information or line of computer code. It is used to check whether errors occurred during the transmission or storage of data.[45]

Satellite Number (11416U). This is the catalog number that USSTRATCOM has designated for this object. A *U* indicates an unclassified object.

Inclination (98.5105). The angle, in degrees, is the mean inclination.

Right Ascension of the Ascending Node (69.3305). The angle, in degrees, is the mean right ascension of the ascending node.

Eccentricity (0012788). The value is the mean eccentricity over the orbit. A leading decimal must be applied to this value.

Argument of Perigee (63.2828). This value is the mean argument of perigee over the orbit.

Mean Anomaly (296.9658). The mean anomaly is the angle, in degrees, measured from perigee of the satellite location in the orbit referenced to a circular orbit with the radius equal to the semimajor axis.

Mean Motion (14.24899292). The value is the mean number of orbits per day the object completes. There are eight digits after the decimal, leaving no trailing space(s) when the following element exceeds 9999. The period of the satellite's orbit can be determined by taking the total number of minutes in a sidereal day (1,436 minutes) and dividing it by the mean motion. For this particular satellite, the period would be 1,436 ÷ 14.24899292 = 101.06 minutes.

Revolution Number and Check Sum (346978). This is the orbit number at epoch time. This time is chosen very near the time of true ascending node passage as a matter of routine. At the time of this element set, the *NOAA 6* had completed 34,697 revolutions around the earth. The last digit is the check sum for line two.

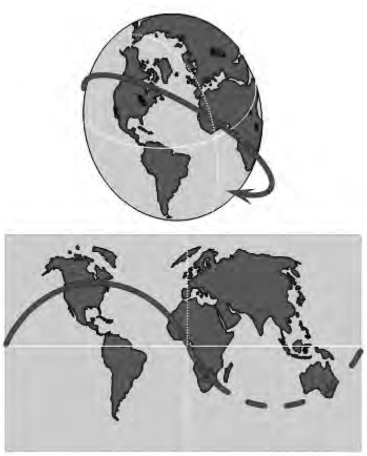

Figure 6-22. Ground track. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-16.)

Ground Tracks

The orbit parameters determine which points on the earth a satellite flies over and when. The fly-over points will be key for controlling or communicating with satellites from fixed ground stations and also knowing where on the earth a satellite sensor can see. To determine the fly-over points, a line is drawn between the earth's center and the satellite. The point on the line at the surface of the earth is called the satellite subpoint, or nadir.[46] The path the satellite subpoint traces on the earth's surface over time is referred to as the satellite ground track, or ground trace, as shown in figure 6-22.

Since the earth is rotating under the satellite, the intersection of the orbital plane and the earth's surface is continually changing. Because of this relative motion, ground tracks come in various forms and shapes based on the orbit parameters discussed above.

Inclination. Inclination defines the tilt of the orbital plane and therefore defines the maximum latitude, both north and south of the ground track. A satellite with a 50° inclination will have a ground track that moved between 50° north and 50° south latitude. In fact, due to symmetry, if a satellite passes over 50° north, it *must* pass as far south as 50°. Any orbit passes over the pole if, and only if, it has an inclination of 90°.

Period. With a nonrotating Earth, the ground track would be a circle passing over the same terrestrial points every orbit. Because the earth does rotate 15° per hour, by the time the satellite returns to the same place in its orbit after one revolution, the earth has rotated eastward by some amount. The ground track therefore looks like it has moved westward on the earth's surface (westward regression). The amount of regression is proportional to the time it takes for one orbit (i.e., the period). The orientation of the satellite's orbital plane does not change in space; the earth has just rotated beneath it.

The example in figure 6-23 shows a satellite in a circular orbit with a period of 90 minutes and an inclination of approximately 50°. With a 90-minute period, the satellite's ground trace regresses 22.5° westward per revolution (15°/hour × 1.5 hours = 22.5°) around the earth. This figure shows three successive orbits around the earth.

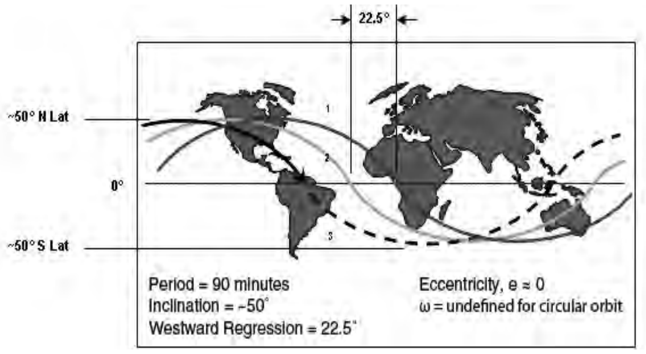

Figure 6-23. Earth's rotation effects. (*Adapted from* Air University, *Space Primer*, unpublished book, 2003, 8-16.)

Eccentricity. The above example shows a circular orbit ($e \approx 0$), which produces sinusoidal ground tracks. Eccentricity affects the ground track because the satellite spends different amounts of time in different parts of its orbit (it is moving faster or slower). This means it will spend more time over certain parts of the earth than others. This has the effect of creating an unsymmetrical ground track.

Argument of Perigee. The argument of perigee skews the ground track. For a prograde orbit, at perigee the satellite will be moving faster eastward than at apogee, in effect tilting the ground track. A great example of this type of effect on a ground track can be seen in figure 6-6, which shows the track of a Molniya orbit.

Launch Considerations

When a satellite is launched, it is targeted for a specific orbit. Several factors must be taken into consideration such as launch window, launch azimuth, desired orbital inclination, desired orbital altitude, and launch booster type. These factors are addressed in two general categories: launch location and launch velocity. The final and probably most important consideration is the launch cost.

Launch Location. The location of the launch site is extremely important because it usually determines the range of possible orbital inclinations in which to insert a satellite. Most satellites launched into orbit are considered direct launch satellites. Note that a direct launch from a latitude of 28° will by definition have an inclination of at least 28° since the orbital plane must pass through the launch site and the center of the earth. Lower inclinations will require an on-orbit plane change or maneuver, which has significant fuel penalties.

A launch window is defined as the period of time during which a satellite can be launched directly into a specific orbital plane from a specific launch site.[47] If the orbital plane inclination is greater than the launch site latitude, the launch site will pass through the orbital plane twice a day, producing two launch windows per day. The direction to point is known as the launch azimuth, measured from the north clockwise.

If the inclination of the orbital plane is equal to the launch site latitude, the launch site will be coincident with the orbital plane once a day, producing one launch window per day at a launch azimuth of 90° (due east). If the inclination is less than the launch site latitude, the launch site will not pass through, or be coincident with, the orbital plane at any time, so there will not be any launch windows for a direct launch.[48]

A simplified model for determining inclination (*i*) from launch site latitude (*L*) and launch azimuth (*Az*) is:

$$\boxed{\cos(i) = \cos(L) \bullet \sin(Az)}$$

The launch azimuths allowed (in most countries) are limited due to the safety considerations that prohibit launching over populated areas or foreign airspace. This restriction further limits the possible inclinations from any launch site.[49]

Launch Velocity. When a satellite is launched, a tremendous amount of energy is imparted to it. Such forces are necessary to overcome the gravitational force of the earth as discussed previously. To maintain a minimum circular orbit at an altitude of 90–100 miles, the satellite has to travel at about 17,500 mph. Due to the earth's rotation, more or less kinetic energy may need to be supplied, depending on launch azimuth. The starting velocity at the launch site varies with latitude and can be determined by multiplying the cosine of the latitude by 1,037 mph. For example, at an altitude of 45° north latitude, the starting velocity would be determined in the following manner:

$$\boxed{\cos(45) \text{ x } 1{,}037 \text{ mph} = 0.7071068 \text{ x } 1{,}037 \text{ mph} = 733.3 \text{ mph}}$$

A satellite launched from the equator in the same direction as the earth's rotation (due east) has an initial speed of 1,037 mph. Therefore,16,463 mph must be supplied (17,500 mph – 1,037 mph = 16,463 mph) to launch a satellite into that particular orbit

(90–100 mile altitude). If launched from the equator in a retrograde orbit (against the rotation of the earth), 18,537 mph must be supplied. Launching with the earth's rotation saves energy and allows for larger payloads for any given booster. In addition, the above equations show substantial energy savings when locating launch sites close to the equator.

Launch Costs. Launching a satellite into space is an extremely expensive venture. A very common standard used to estimate the cost of putting a satellite in orbit has been $10,000 per pound. In reality, the cost per pound varies greatly. Factors such as the type of launch payload, the launch booster, and orbit type (LEO, GEO, etc.) affect the costs. In one study of the current commercial launch costs, it was determined that the average cost per pound was between $3,632 and $4,587 for LEO launches and between $9,243 and $11,243 for GEO launches.[50] The actual launch costs used to determine these averages ranged from $5 million for a Russian Strategic Arms Reduction Treaty (START) launch vehicle (LEO) on the lower end to $180 million for a European Space Agency *Ariane 5* launch vehicle (GEO) on the higher end.[51]

Orbital Maneuvers

An orbital maneuver is a deliberate change in the size, shape, and/or orientation of a satellite's orbit. The reasons for conducting an orbital maneuver include (but are not limited to) the following: increasing the satellite's field of view, counteracting the effects of atmospheric drag or other perturbations, increasing imaging resolution, rendezvousing with another satellite, or deorbiting a satellite.[52] Perturbations and deorbits will be discussed further in later sections.

Delta-v. As previously mentioned, a satellite's velocity and position determine its orbit. To change one of these requires the application of force, which then accelerates the vehicle by Newton's second law. This acceleration produces an impulsive change in velocity, known as delta-v (Δv), which changes the size of the orbit by either adding or subtracting energy.[53] For any single Δv orbital change, the desired orbit must intersect the current orbit, and the point of intersection is where the change is applied. Otherwise it will take at least two Δv's to achieve the final orbit, one to leave the current orbit and another to join the final desired orbit. The amount of Δv required can be determined by subtracting the present vector from the desired vector.

Mission Considerations. Mission planners must ensure that a satellite is provided with sufficient fuel to perform the above maneuvers once in orbit. Additional fuel on board a satellite results in a heavier payload and may require a more powerful booster to place the satellite in orbit, so these maneuvers must be planned carefully. There are two types of orbital maneuvers: in plane and out of plane.

In-plane maneuvers are the most common type of orbital maneuvers performed since they require much less fuel and energy to perform. These maneuvers are conducted to change a satellite's period (size), argument of perigee, or true anomaly.[54] The majority of in-plane maneuvers are performed to counter the external forces, or perturbations, that are constantly acting upon the satellite and changing its orbit.

Out-of-plane maneuvers result in a change in inclination or right ascension of the ascending node.[55] This type of maneuver requires a much larger amount of fuel to generate the sufficient velocity vectors (Δv) to change the satellite's orbital plane. For example, a 28° plane change, such as would be necessary for a Kennedy Space Center–launched satellite to become equatorial, requires a Δv of about 3.5 km/s. This same Δv

applied in-plane would be enough for the two burns needed to raise an LEO satellite to geostationary.

Perturbations

Some orbit maneuvers are done simply to maintain the given orbit in the light of perturbations, which were ignored earlier in our discussion to simplify the orbital elements. However, in the real world, all satellites are subject to external forces acting upon a satellite that affect its otherwise constant orbital parameters. These forces have a variety of causes, origins, and effects. For instance, because of drag, the eccentricity of a satellite orbiting the earth can never truly equal zero. These forces are named and categorized in an attempt to model their effects. The major perturbations are:

- Earth's oblateness
- Atmospheric drag
- Third-body effects
- Solar wind/radiation pressure
- Electromagnetic drag

Earth's Oblateness. The earth is not a perfect sphere. It is somewhat asymmetrical at the poles and bulges at the equator. This squashed shape is referred to as oblateness, or the J2 effect. The north polar region is more pointed than the flatter south polar region, producing a slight "pear" shape. Also, the equator is not a perfect circle; it is slightly elliptical when looking down on it from the top. The effects of the earth's oblateness are gravitational variations or perturbations, which have a greater influence the closer a satellite is to the earth. For low to medium orbits, these influences are significant.[56]

One effect of the earth's oblateness is nodal regression. Westward regression due to the earth's rotation under the satellite was discussed above in the section on ground tracks. Nodal regression is an actual rotation of the orbital plane about the earth (the right ascension changes) relative to the fixed reference line—the first point of Aries. If the orbit is prograde, the orbital plane rotates westward around the earth (right ascension decreases); if the orbit is retrograde, the orbital plane rotates eastward around the earth (right ascension increases).

In most cases, perturbations must be counteracted. However, in the case of sun-synchronous orbits, perturbations can be advantageous. Picking a specific slightly retrograde orbit, the angle between the orbital plane and a line between the earth and the sun remains constant and thus "sun-synchronous." This works because as the earth orbits eastward around the sun, the orbital plane drifts due to the J2 effect around the earth at the same rate.

A sun-synchronous orbit is beneficial because it allows a satellite to view the same place on Earth with the same sun angle (or shadow pattern) every day. This is very valuable for remote sensing missions because they use shadows to measure object height. With a constant sun angle, the shadow lengths give away any changes in height, or any shadow changes give clues to exterior configuration changes.[57]

Another significant effect of Earth's asymmetry is apsidal line rotation. This effect appears as a rotation of the orbit within the orbital plane, that is, the argument of perigee changes. This is true for all orbits except at an inclination of 63.4° (and its retrograde complement, 116.6°), where this rotation happens to be zero. The Molniya orbit was specifically designed with an inclination of 63.4° to take advantage of this perturbation. With the zero effect at a 63.4° inclination, the stability of the Molniya orbit improves, limiting the need for considerable onboard fuel to counteract this rotation. Without this effect, the apogee point would rotate away from the desired communications zone (i.e., from the Northern to Southern Hemisphere), and the satellite would be useless.[58]

The ellipticity of the equator has an effect that shows up most notably in geostationary satellites (also in inclined geosynchronous satellites). Because the equator is elliptical, most satellites are closer to one of the lobes and experience a slight gravitational misalignment. This misalignment affects geostationary satellites more because they view the same part of the earth's surface all the time, resulting in a cumulative effect. The elliptical force causes the subpoint of the geostationary satellite to move east or west with the direction depending on its location. There are two stable points at 75° east and 105° west and two unstable stable points 90° out (165° east and 5° west). This movement would be bad not only because the satellite would no longer "hover" over the point of interest, but also because it would cause collisions if all the GEO satellites drifted to these two nodes.

Atmospheric Drag. The earth's atmosphere does not suddenly cease; rather it trails off into space. The current atmospheric model is not perfect because of the many factors affecting the upper atmosphere, such as the earth's day-night cycle, seasonal tilt, variable solar distance, fluctuation in the earth's magnetic field, the sun's 27-day rotation, and the 11-year sun spot cycle. Even a very thin atmosphere causes a drag force due to the high orbital speeds of the satellites. The drag force also depends on the satellite's coefficient of drag and frontal area, which varies widely between satellites.[59] Up to 1,000 km (620 miles), the slowing effect it has on satellites must be taken into account.

The uncertainty in these variables causes predictions of satellite decay to be accurate only for the short term. An example of changing atmospheric conditions causing premature satellite decay occurred in 1978–79, when the atmosphere received an increased amount of energy during a period of extreme solar activity. The extra solar energy expanded the atmosphere, causing several satellites to decay prematurely, most notably the US space station Skylab.[60]

The highest drag occurs when the satellite is closest to the earth (at perigee) and has an effect similar to performing a retro-rocket delta-v at perigee; it decreases the apogee height, circularizing the orbit. On every perigee pass, the satellite loses more kinetic energy (negative delta-v), circularizing the orbit more and more until the whole orbit is experiencing significant drag and the satellite spirals in, enters the earth's atmosphere, and falls back to the earth.[61] For example, the International Space Station currently drops in altitude 30 km per month and thus requires a reboost at every shuttle rendezvous.

Third-Body Effects. According to Newton's Law of Universal Gravitation, every object in the universe attracts every other object in the universe. The greatest third-body effects come from those bodies that are very massive and/or close, such as the sun, Jupiter, and the moon. These forces affect satellites in orbit as well. The farther a satellite is from the earth, the greater the third-body forces are in proportion to Earth's gravitational force, and therefore, the greater the effect on the high-altitude orbits.[62]

Radiation Pressure. The sun is constantly expelling atomic matter (electrons, protons, and Helium nuclei). This ionized gas moves with high velocity through interplanetary space and is known as the solar wind. Satellites are like sails in this solar wind, alternately being speeded up and slowed down, producing orbital perturbations.[63]

Electromagnetic Drag. Satellites are continually traveling through the earth's magnetic field. With all their electronics, satellites produce their own localized magnetic fields which interact with the earth's, causing torque on the satellite. This torque mainly turns the satellite within its orbit rather than affecting the orbit itself as the other perturbations do.

Deorbit and Decay

So far the concern has been with placing and maintaining satellites in orbit. Low Earth orbit satellites have an expected mission duration (life expectancy). Once a payload has completed its mission, it is essentially "taking up space" in space. In addition, when a payload is launched into orbit, other pieces from that launch such as the rocket body, platform, or debris may also remain in orbit. Due to the effects of perturbations, most of these objects will eventually reenter the earth's atmosphere. The only questions are when and how. The answers can be determined by mission planners, who are responsible for deciding whether to deorbit an object or allow it to naturally decay.

A deorbit is the deliberate, controlled reentry of an object into the earth's atmosphere to a specific location.[64] This is usually done to recover something of value, such as people in the case of the space shuttle returning from the International Space Station. It is also done to protect civilians by controlling the reentry of large objects that may survive reentry through the earth's atmosphere as was the case with the deorbit of the Russian Mir space station in March 2001.[65] Most LEO objects are not payloads but rather space junk and therefore cannot be controlled by satellite operators for a possible deorbit. These objects are left to decay naturally back to the earth.

A decay is the uncontrolled reentry of an object into the earth's atmosphere. The effects of perturbations, most notably atmospheric drag, will eventually reduce a satellite's orbital altitude to the point where it can no longer remain in orbit. As discussed in a previous section, this altitude is approximately 150 km (93 miles). It is possible for these decaying objects to be detected through the Space Surveillance Network, discussed in chapter 19. In addition, predictions for reentry dates and locations for decaying objects can be determined by USSTRATCOM's Joint Space Operations Center, as discussed in chapter 12.

In some situations, the satellites are in such stable orbits that natural perturbations will not do the disposal job. In these instances, the satellite must be removed from its operational orbit to another location. To return a satellite to Earth without destroying it takes a considerable amount of energy. Obviously, it is impractical to return old satellites to Earth from a high Earth orbit. The satellite is usually boosted into a slightly higher orbit to get it out of the way, and there it will remain for thousands of years. This practice is common for geosynchronous satellites. By boosting the orbit even higher (> 22,236 miles) above the earth, the satellite is placed in what is called a supersynchronous orbit.[66]

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