SATELLITE LAUNCHING

ECE 514E- RADAR & SATELLITE ENGINEERING

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KENYA SPACE AGENCY
Possibilities beyond our skies

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Kenya Space Agency Strategic Plan

Kenya Space Agency staff pose for a photo during strategic plan 2020 - 2025 launch.

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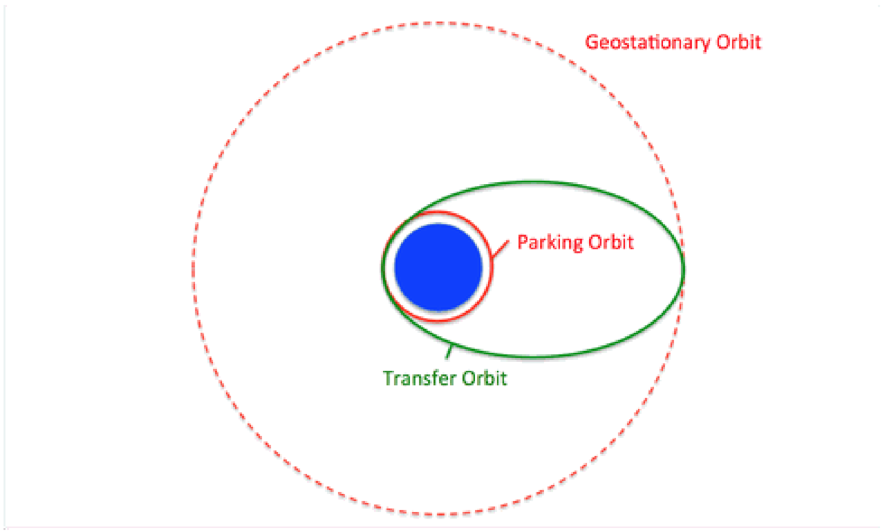
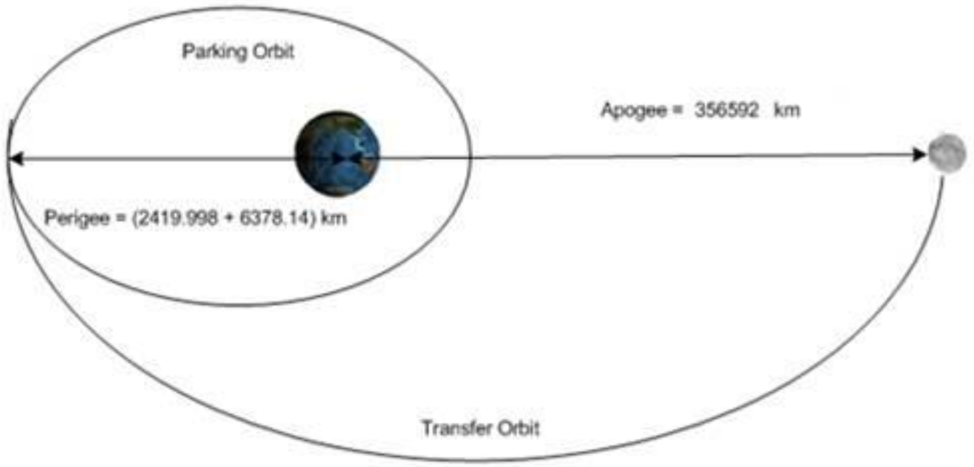
PARKING ORBIT

- **A parking orbit** is a temporary orbit used during the launch of a spacecraft. A launch vehicle boosts into the parking orbit, then coasts for a while, then fires again to enter the final desired trajectory.



- While in parking orbit, the Satellite is checked out and its trajectory measured to determine the velocity and time required to send it to the final orbit or into space is a specific direction.

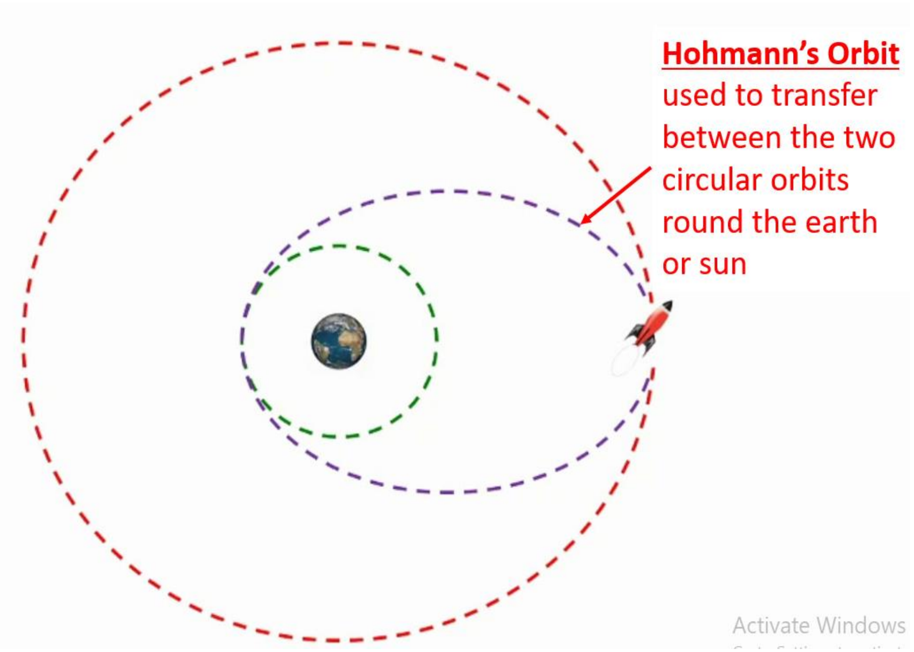


REASONS FOR USING PARKING ORBITS

1. **Parking increases the launch window.** For earth-escape missions, these are often quite short (seconds to minutes) when no parking orbit is used. With a parking orbit, these can often be increased up to several hours.
2. For **non-LEO** missions, the desired location for the final burn may not be in a convenient spot. In particular, for earth-escape missions that want good northern coverage of the trajectory, the correct place for the final burn is often in the southern hemisphere.
3. For **geostationary orbit** missions, the correct spot for the final (or next to final) firing is normally on the equator. It is therefore necessary to hold the satellite in a parking orbit until it is over the equator, then fire again into a geostationary transfer orbit.
4. For **lunar missions**, a parking orbit allowed some checkout while still close to home, before committing to the long lunar trip.
5. Parking is also necessary when the desired orbit has a **high perigee**. In this case the booster launches into an elliptical parking orbit, then coasts until a higher point in the orbit (apogee), then fires again to raise the perigee.

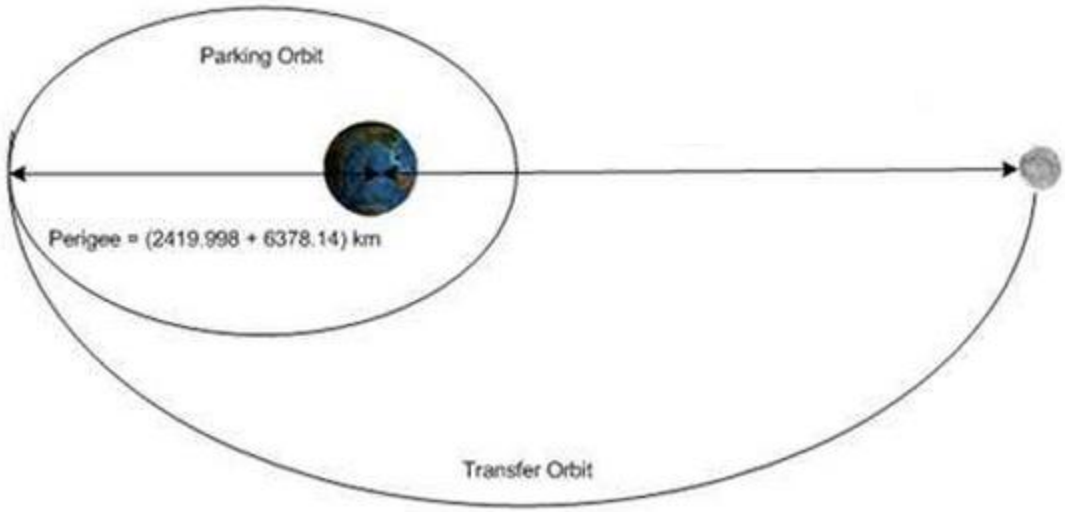
HOHMANN TRANSFER ORBIT

1. The Hohmann transfer orbit is an elliptical orbit used to transfer between two circular orbits of different altitudes, in the same plane.
2. The orbital manoeuvre to perform the Hohmann transfer uses two engine impulses, one to move a spacecraft onto the transfer orbit and a second to move out.
3. The manoeuvre was named after Walter Hohmann, the German scientist who published a description of the orbit in his 1925 book.

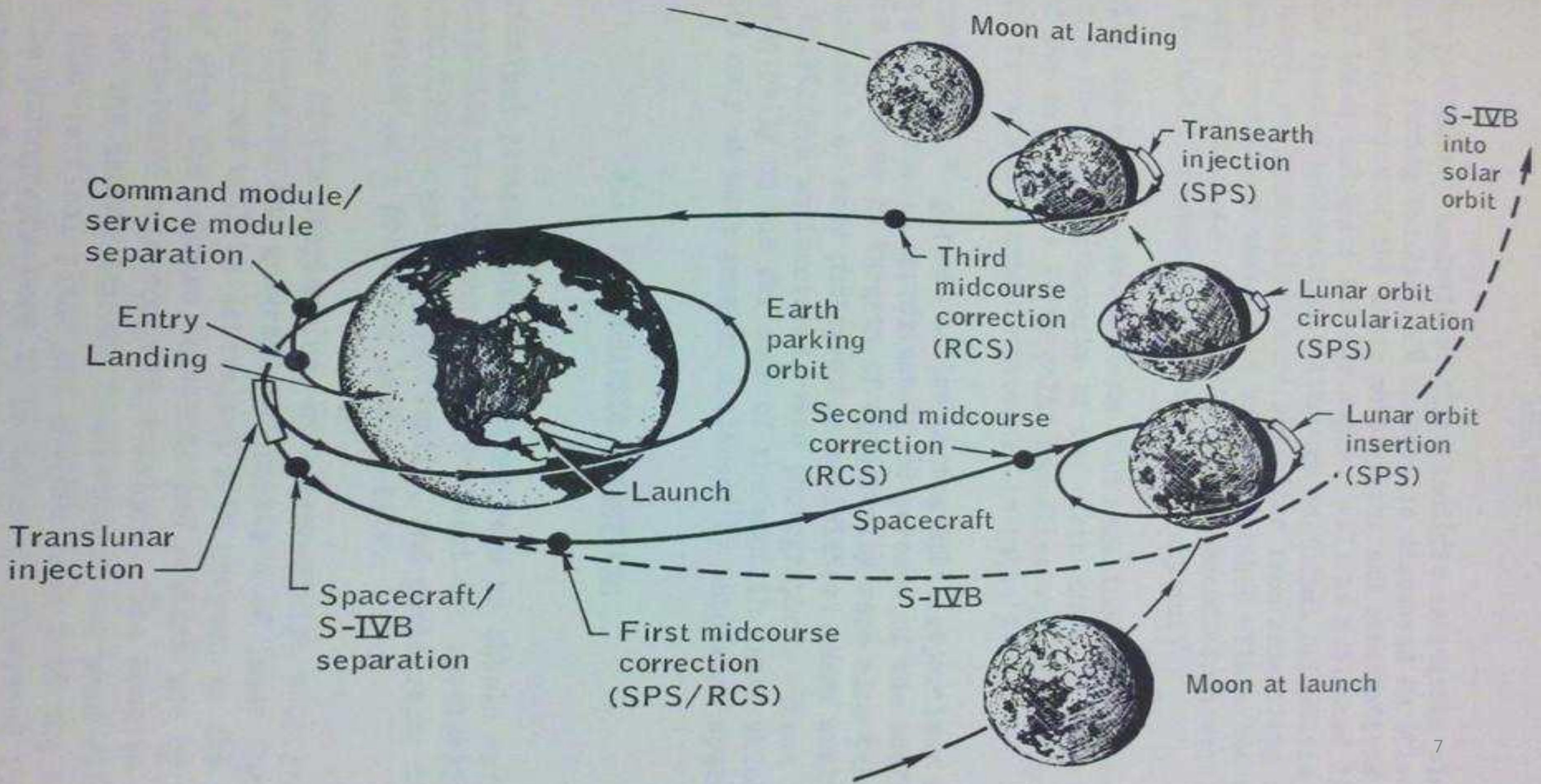


USE OF PARKING ORBITS IN LUNAR MISSIONS

1. The space station is first placed in an elliptical orbit-the parking orbit.
2. The next stage of the journey to the Moon requires firing from parking orbit to transfer orbit.
3. This is accomplished by firing the spacecraft’s engine when it reaches its apogee.
4. The goal is to make the apogee of the parking orbit serve as the perigee of the mission orbit.
5. The apogee of the transfer orbit now corresponds to the Moon.

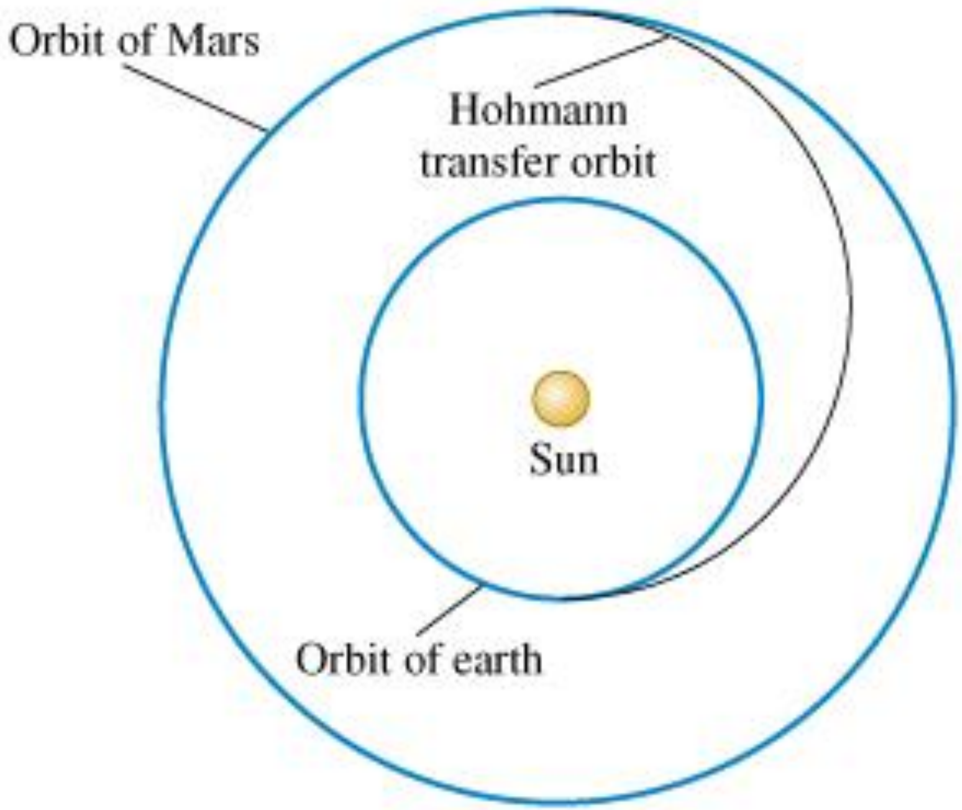


LUNAR MISSION



EARTH TO MARS MISSION

- The most efficient way to send a spacecraft from the earth to mars is by using a **Hohmann transfer orbit**.
- The Hohmann transfer orbit is an elliptical orbit whose perigee and apogee are tangent to the orbits of the two planets.
- The rockets are fired briefly at the departure planet to put the spacecraft into the transfer orbit; the spacecraft then coasts until it reaches the destination planet.
- The rockets are then fired again to put the spacecraft into the same orbit round the sun as mars.



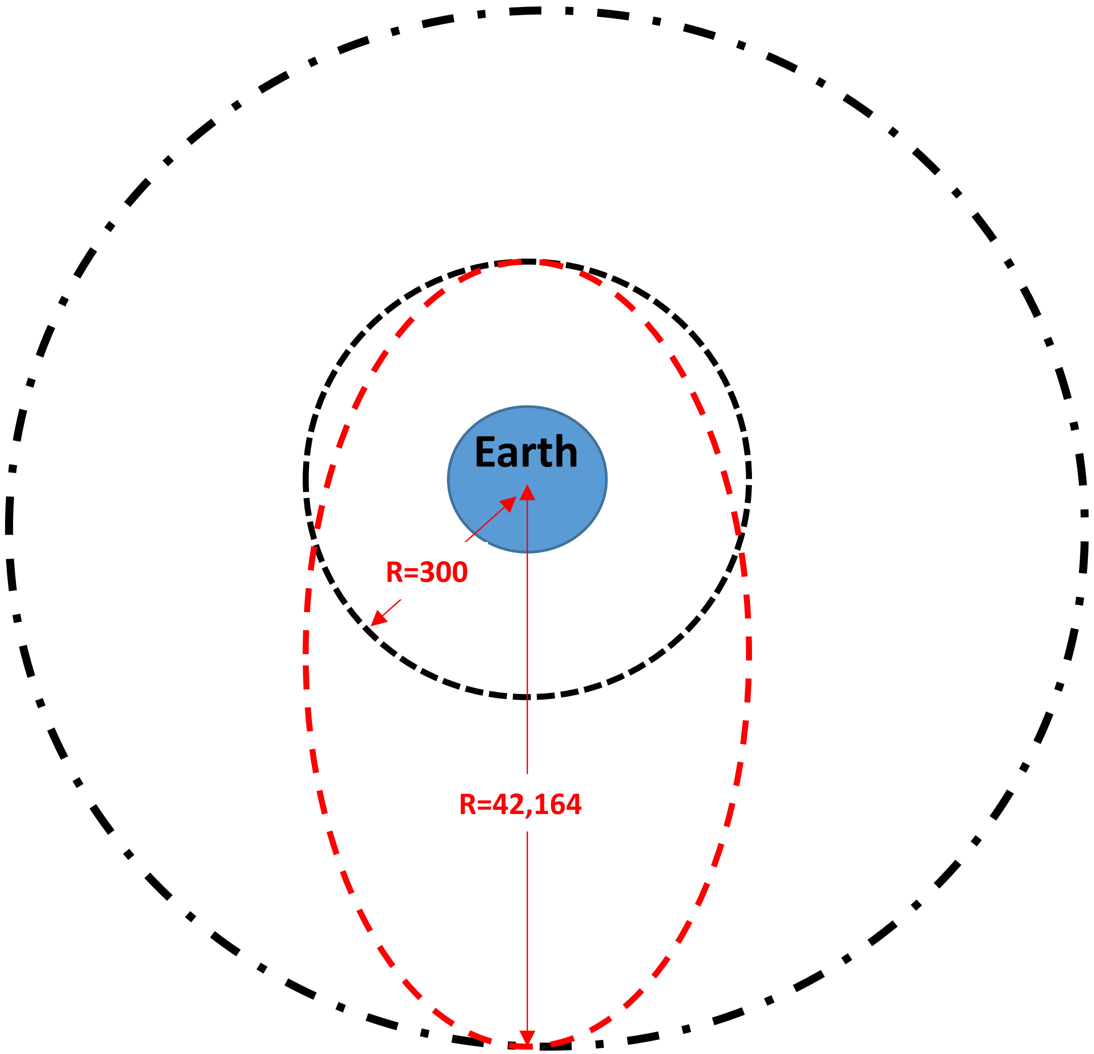
DISADVANTAGES OF USING PARKING ORBITS

The main disadvantage of using parking orbits is the need to use liquid fuel engines as opposed to solid fuel. The reasons are as follows.

1. A rocket needs to coast for a while in the parking orbit, then restart while under zero gravity conditions.
2. But the length of two of the burns (the initial injection burn, and the final burn) typically depend on where in the launch window of the launch occurs.
3. To do this without wasting fuel, a rocket system that can fire, then stop, then start again as needed.
4. This requires a liquid fuel engine since solid fuel rockets cannot be stopped or restarted - once ignited they burn to completion.

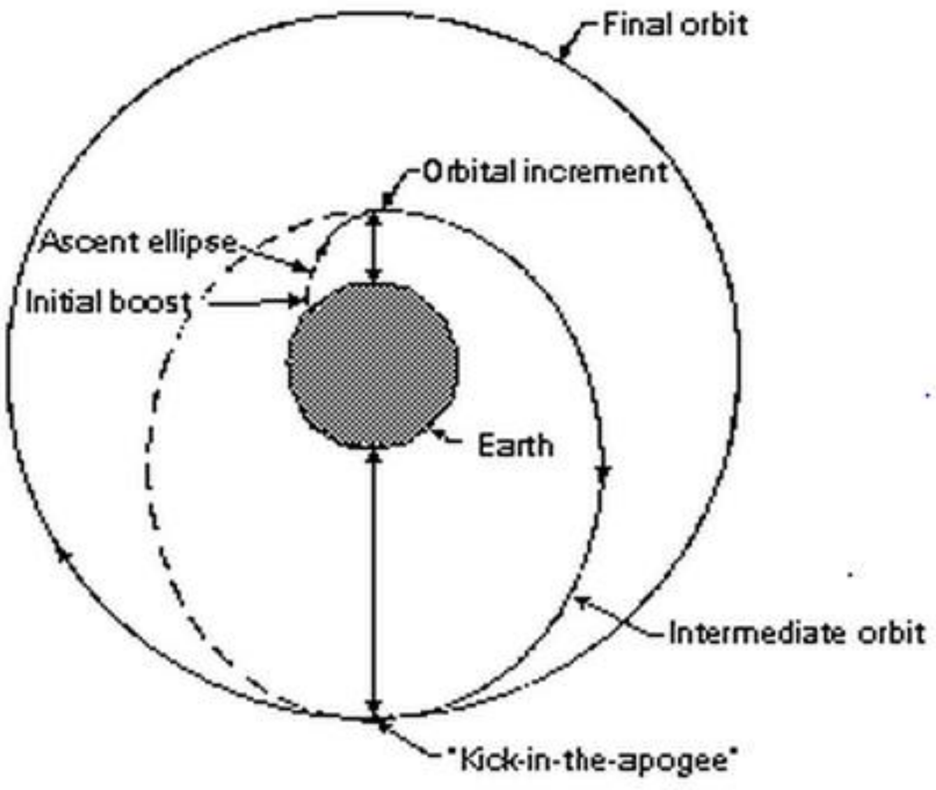
GEOSTATIONARY ORBIT SATELLITE LAUNCHING METHOD 1

1. The satellite is placed in a lower circular orbit at a radius of 300Km
2. The velocity of the satellite is then increased through various propulsion stages changing the orbit into an elliptical orbit with a perigees of 300Kms and an apogee radius of 42,164 Kms.
3. The second velocity increment is then used to make the orbit circular with a radius of 42,164 Kms.
4. This method is used by the Space Transport System (STS).



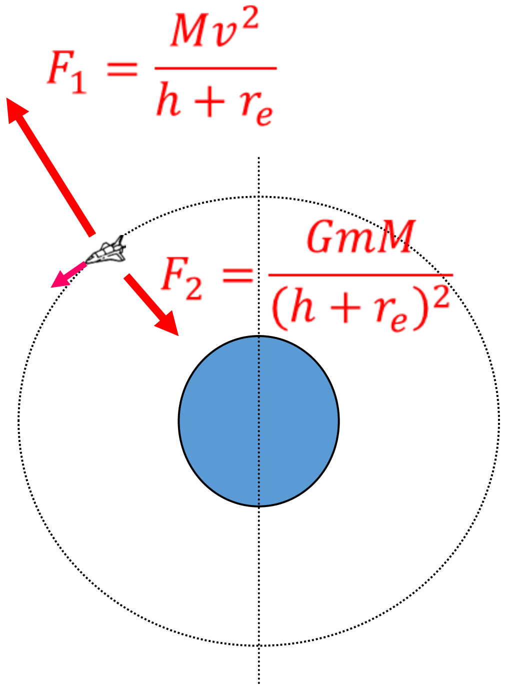
GEOSTATIONARY ORBIT SATELLITE LAUNCHING METHOD 2

1. The satellite is placed directly into an elliptical lower orbit with a perigee of 300Km and apogee of 42,164 Kms.
2. One velocity increment applied at the apogee pushes the satellite into the geostationary orbit.
3. This method is used by Expendable Launch Vehicle such as Ariane and Delta.



SATELLITE IN CIRCULAR ORBIT

1. Let M be the mass of the earth and m be that of the satellite.
2. If v_s is the velocity of the satellite, h is the height of the satellite and r_e is the radius of the earth then we can write:
$$v_s = \omega(r_e + h)$$
3. If F_1 is the centripetal force on the satellite, then we can write:
$$F_1 = \frac{mv_s^2}{r_e + h} = \frac{m\omega^2(r_e + h)^2}{(r_e + h)} = m\omega^2(r_e + h)$$
4. If the period of the satellite is T, then $\omega = \frac{2\pi}{T}$ and we can write:
$$F_1 = m\left(\frac{2\pi}{T}\right)^2 (r_e + h)$$
5. But the gravitational pull from the earth is given by:
$$F_2 = \frac{GMm}{(r_e + h)^2}$$
6. In equilibrium $F_1 = F_2$
$$m\left(\frac{2\pi}{T}\right)^2 (r_e + h) = \frac{GMm}{(r_e+h)^2}$$
or
$$\left(\frac{2\pi}{T}\right)^2 = \frac{GM}{(r_e+h)^3}$$



For the earth:
$G = 6.672 \times 10^{-11}$ Newton Metre/kg^2
$M = 5.97 \times 10^{24} Kg$
Gravitational Coefficient (or Kepler's Coefficient) is:
$g_0 = GM = 3.9861 \times 10^5$ Km^3/s^2

PERIOD OF SATELLITE IN CIRCULAR ORBIT

$$\left(\frac{2\pi}{T}\right)^2 = \frac{GM}{(r_e+h)^3}$$

This Confirms Kepler's Second law, i.e

$$R^3 = \frac{\mu}{\omega^2}$$

- Rearranging the equation and substituting g_o = GM gives:

$$T^2 = \frac{4\pi(r_e + h)^3}{g_0}$$

$$T = \frac{2\pi}{\sqrt{g_o}}(r_e + h)^{3/2}$$

VELOCITY OF SATELLITE IN CIRCULAR ORBIT

If v_s is the velocity of the satellite, h is the height of the satellite and r_e is the radius of the earth then we can write:

$$v_s = \omega(r_e + h) = \frac{2\pi}{T}(r_e + h)$$

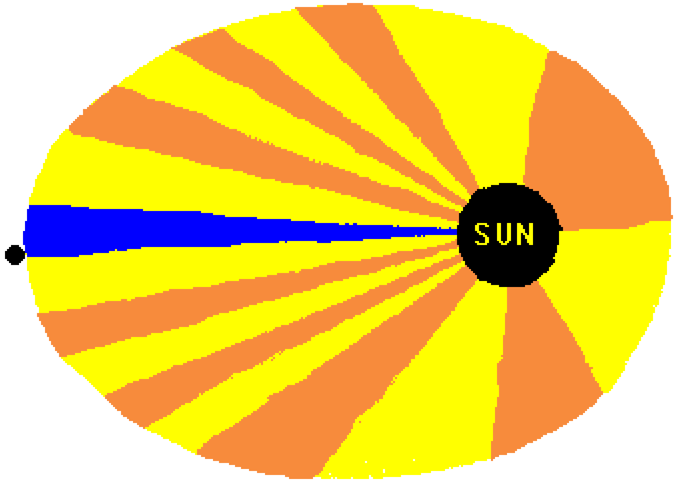
But

$$T = \frac{2\pi}{\sqrt{g_o}}(r_e + h)^{3/2}$$

Therefore

$$v_s = \frac{(r_e+h)}{(r_e+h)^{3/2}/\sqrt{g_o}} = \sqrt{\frac{g_o}{(r_e+h)}}$$

This confirms Kepler's Second law that an increase in radius of the orbit leads to a decrease in the velocity of the satellite.



ESCAPE VELOCITY FOR CIRCULAR ORBIT

1. Escape velocity is the speed at which the kinetic energy plus the gravitational potential energy of an object is zero.
2. It is the speed needed to "break free" from the gravitational attraction of a massive body, without further propulsion.

$$\frac{1}{2}mv_e^2 = \frac{GMm}{r}$$

$$v_e = \sqrt{\frac{2GM}{r}}$$

3. The term escape velocity is actually a misnomer, and it is often more accurately referred to as escape speed since the necessary speed is a scalar quantity which is independent of direction