



RESEARCH ARTICLE

ΔV CALCULATION FOR FOUR DIFFERENT CASES IN GEOSTATIONARY ORBIT

*Ramya Preethi, S.

School of Aeronautical Sciences, Hindustan Institute of Technology and Science, Chennai-603103, India

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ABSTRACT

From the previous discussions of the paper we studied various orbit types under inclination and eccentricity classifications and chosen to work on GEO based on two major ideas (i) fuel consumption and (ii) building a constellation in GEO. To obtain a desired geostationary orbit, we need to calculate its ΔV budget, keplerian orbital elements and orbital perturbations. In this paper, ΔV calculation will be done four different cases and obtain a low ΔV total, which is more efficient for less fuel consumption. In this paper four major cases are assumed and made calculations based on it and obtained low ΔV total. Apart from this also described about the need and importance of ΔV Budget in a launching. Most importantly to build an orbit for a communication satellite we need to place a satellite. For that the GSLV F04 of MK.II series is chosen.

INTRODUCTION

GEO satellites are unlikely to be launched onto their orbits because of the restriction of the geographic latitudes of spacecraft launching spots. At an altitude between 200-400 km GEO satellite is usually sent up into the parking orbit. Above the equator, as the satellite approaches the space the upper stage gets ignited and the satellite breaks away from the tail stage. With both the perigee and apogee of the orbit above the equator, the satellite enters the highly elliptical orbit here. The orbit-insertion point and the GEO will be equalled by the altitudes of perigee and apogee. The apogee engine on the satellite will be ignited at the apogee of the highly elliptical orbit which drives the satellite into GEO. Delta-V budget is the first and for most calculation to be done, to evaluate the amount of fuel requirement for a specific mission. It is also used to evaluate fuel consumption during the two impulsive burns. There are numerous impulsive burns is used for different orbit types to enumerate various classes of manoeuvres like sum the total delta-V (financial budget), delta-V per manoeuvre and number of manoeuvres required over the life time of the mission. Launch windows and the relative position of the gravitating bodies are calculated from pork chopplots (delta-V plotted against launch time) to attain the mission at a required delta-V.

ΔV Budget

Delta-V budget is required for the propulsive manoeuvres during the mission. To determine the amount of propellant required for a spacecraft of given mass and propulsion system. It depends only on the desired trajectory and not on the mass of the vehicle because it is a scalar quantity. It is required to move between different space venues which place a major role in planning for space missions. Delta-V is contrasted to rocket burning time because when more fuel is used in the latter, it will be a greater effect in the mission later. Eventually, large amount of fuel is needed to transfer a heavier communication satellite from LEO to GSO than for a light weight satellite but the delta-V requirement is same. Need of delta-V, will be more when the space vehicle is near the atmosphere; therefore it drags the vehicle by aero braking.

ΔV Budget calculation

The delta-V of a rocket stages is been shown by Tsiolkovsky rocket equation, is proportional to the logarithm of the fuel to empty mass ratio of the vehicle and to the specific impulse of the rocket engine. Major goal is expected when designing a space- mission trajectory i.e. to minimize the delta-V to reduce the expense and size of the rocket. As a result payload can be successfully reaching its destination. A simple transfer like hohmann transfer needs a less required delta-V budget. Whereas in some complex transfers (the orbits are not coplanar) needs an additional delta-V to change its plane. At the intersection of the two orbital planes, the delta-V is

*Corresponding author: Ramya Preethi, S.

School of Aeronautical Sciences, Hindustan Institute of Technology and Science, Chennai-603103, India.

extremely high because substantial burns used depending on the velocity of the vehicle. If the deflection occurs due to the mass of a planetary body or its gravity, delta-V is not needed in this case (the vehicle will move freely). But in other cases, in high altitude apoapsis gives low speed before the plane change and thus leads to a lower total delta-V. In interplanetary mission, slingshot effect is used to give a boost up speed. Likewise, Oberth effect is used to decrease the delta-V budget. In Bi-elliptic transfers very little delta-V is required because they have high non-linear effects and their trajectories are close to Lagrange points.

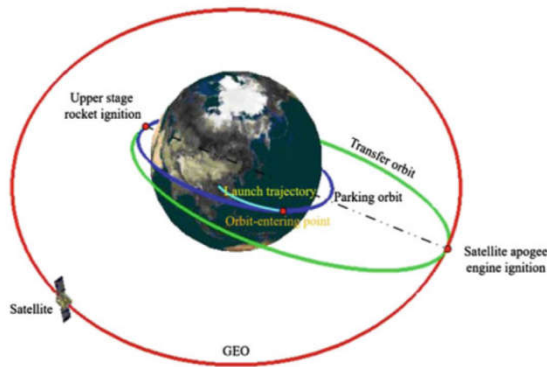


Figure 1. Launching process of GEO Satellites [Source: Authors-Zhang *et al.* <http://www.springer.com/978-981-10-2947-9>]

Launch vehicle

This project is about designing a geostationary orbit for a communication satellite, but to design an orbit we need a satellite to be placed. In order to launch a satellite to GEO, a Hohmann transfer is required. Our orbit is going to be a prograde orbit (it rotates in the same direction as the Earth's rotation which provides the orbital velocity with a consequent saving of launch energy). Hence GSLV launch vehicle is chosen for the satellite launch. GSLV F04 of MK.II series is chosen for a sample of its perigee and apogee kilometres. It can carry a maximum payload of about 2,500kg to GEO at an altitude of 35,786km from the Earth's surface.

Launch sequence

It is basically divided into two broad categories, one is employed by an expendable vehicle and the other is employed by the reusable vehicle. When a satellite is launched in irrespective weather either by a reusable or an expendable vehicle which is heading for a geostationary orbit, first is been placed in a transfer orbit. Its perigee altitude is between 200km and its apogee will sustain at its geostationary altitude. In other cases, the launch vehicle injects the satellite directly into a transfer orbit. In this type an apogee manoeuvre starts circulating the orbit at the geostationary altitude. Then the last step is to carry out the orbit correction for its inclination. In another possibility the satellite is first launched into a low Earth circular orbit. Further it is transformed into an elliptical orbit with a perigee manoeuvre. Then circularization of the transfer orbit correction of orbit inclination takes place. Prograde orbit which is also called as direct orbit have an easterly component of velocity to gain launchers from the Earth's rotational velocity. Then with the retrograde (westerly) launch, a significantly larger payload can be launched in an easterly direction. Thus easterly launches are used for the initial launch into the geostationary orbit.

Calculation and analysis

ΔV Calculation for various cases

CASE 1: First Impulse at Perigee of Parking Orbit

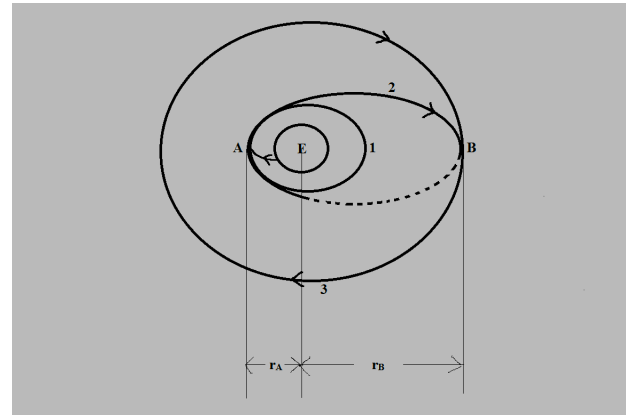


Figure 2. Impulse starts at Perigee of Parking Orbit

Elliptical parking orbit to GEO

Parking orbit: apogee=170 km

Perigee=168 km

ORBIT 1

Eccentricity $e_1 = 1.528080541 \times 10^{-4}$

Angular momentum $h_1 = 51073.5714 \text{ km}^2/\text{s}$

Speed at 'A' in orbit-1, $V_{A1} = 7.805645827 \text{ km/s}$

ORBIT 2

Eccentricity $e_2 = 0.731327552$

Angular momentum $h_2 = 67197.43077 \text{ km}^2/\text{s}$

Speed at 'A' in orbit-2, $V_{A2} = 10.26987796 \text{ km/s}$

Required velocity increment, $\Delta V_A = 2.464232129 \text{ km/s}$

Speed at 'B' on orbit-2, $V_{B2} = 1.593709548 \text{ km/s}$

ORBIT-3 is circular, so its constant orbital speed

$V_{B3} = 3.0794663614 \text{ km/s}$

$\Delta V_B = 1.480954066 \text{ km/s}$

$\Delta V_{\text{total}} = 3.945186195 \text{ km/s}$

CASE 2: First Impulse at Apogee of Parking Orbit

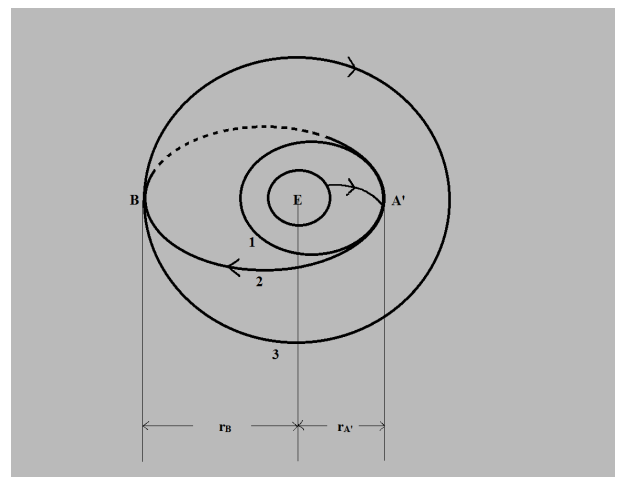


Figure 3. Impulse starts at Apogee of Parking Orbit

Elliptical parking orbit to circular GEO

Parking orbit: apogee = 170 km

Perigee = 168 km

ORBIT 1

Eccentricity $e_1 = 1.528080541 \times 10^{-4}$ Angular momentum $h_1 = 51073.5714 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-1, $V_{A1} = 7.80326066 \text{ km/s}$

ORBIT 2

Eccentricity $e_2 = 0.731256463$ Angular momentum $h_2 = 67206.32006 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-2, $V_{A2} = 10.26809794 \text{ km/s}$ Required velocity increment $\Delta V_A = 2.464837283 \text{ km/s}$ Speed at 'B' in orbit-2, $V_{B2} = 1.593920374 \text{ km/s}$

ORBIT 3

 $V_{B3} = 3.074663614 \text{ km/s}$ $\Delta V_B = 1.48074324 \text{ km/s}$ $\Delta V_{\text{total}} = 3.945580523 \text{ km/s}$

CASE 3: Circular Parking Orbit with Radius R=170 Km

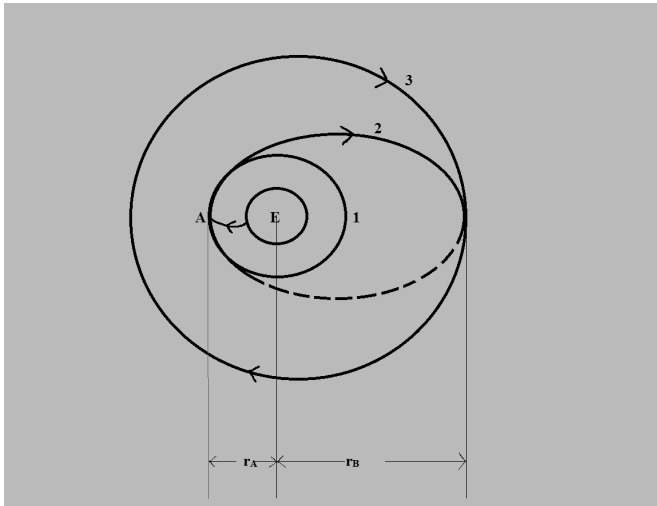


Figure 4. Radius of circular Orbit (170km)

ORBIT 1

Eccentricity $e_1 = 0$ Angular momentum $h_1 = 51077.47408 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-1, $V_{A1} = 7.803856929 \text{ km/s}$

ORBIT 2

Eccentricity $e_2 = 0.731256463$ Angular momentum $h_2 = 67206.32008 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-2, $V_{A2} = 10.26809795 \text{ km/s}$ $\Delta V_A = 2.464241017 \text{ km/s}$ Speed at 'B' in orbit-2, $V_{B2} = 1.5893920374 \text{ km/s}$

ORBIT 3

 $V_{B3} = 3.074663614 \text{ km/s}$ $\Delta V_B = 1.48074324 \text{ km/s}$ $\Delta V_{\text{total}} = 3.944984257 \text{ km/s}$

CASE 4: Circular Parking Orbit with Radius R = 168 Km

ORBIT 1

Eccentricity $e_1 = 0$ Angular momentum $h_1 = 51069.66962 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-1, $V_{A1} = 7.805049513 \text{ km/s}$

ORBIT 2

Eccentricity $e_2 = 0.731327552$ Angular momentum $h_2 = 67197.43077 \text{ km}^2/\text{s}$ Speed at 'A' in orbit-2, $V_{A2} = 10.26987796 \text{ km/s}$ $\Delta V_A = 2.464828443 \text{ km/s}$ Speed at 'B' orbit-2, $V_{B2} = 1.593709548 \text{ km/s}$

ORBIT 3

Speed at 'B' in orbit-3,

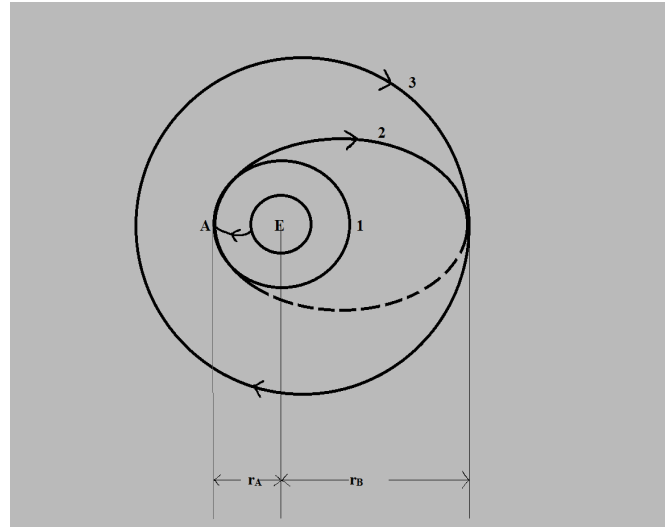
 $V_{B3} = 3.074663614 \text{ km/s}$ $\Delta V_B = 1.480954066 \text{ km/s}$ $\Delta V_{\text{total}} = 3.945782509 \text{ km/s}$ 

Figure 5. Radius of Circular Orbit (168km)

Formulae used

$$\text{Eccentricity } e = \frac{\text{major axis} - \text{minor axis}}{\text{major axis} + \text{minor axis}}$$

$$\text{Angular momentum, } h = \sqrt{(r \times \mu(1 + e \cos \theta))} \quad (\text{At perigee})$$

$$\text{Angular momentum, } h = \sqrt{(r \times \mu(1 - e \cos \theta))} \quad (\text{At apogee})$$

Speed at specific point in the orbit

$$= \frac{\text{angular momentum}}{\text{radius of the orbit at that point}}$$

 $\Delta V_{\text{total}} = \text{sum of the velocity increments}$

Radius of Earth above a particular point

$$= \sqrt{\frac{(a^2 \cos^2 \theta) + (b^2 \sin^2 \theta)}{(a \cos \theta)^2 + (b \sin \theta)^2}}$$

a = semi major axis, b = semi minor axis, θ = latitude of the point on Earth

Radius of Earth above India

a = 6378.1370 km, b = 6356.7523 km, $\theta = 22^\circ \text{N}$ $R_1 = 6375.157675 \text{ km}$ Newton's gravitational law, $F = \frac{GMm}{r^2}$

RESULTS

 ΔV Summarization

Table 1. Delta-V for various cases

CASE NO	CHECKLIST	ΔV_{TOTAL}
1	Elliptical parking orbit with first impulse at perigee	3.945186195km/sec
2	Elliptical parking orbit with first impulse at apogee	3.945580523km/sec
3	Circular parking orbit with radius 170km	3.944984257km/sec
4	Circular parking orbit with radius 168km	3.945782509km/sec

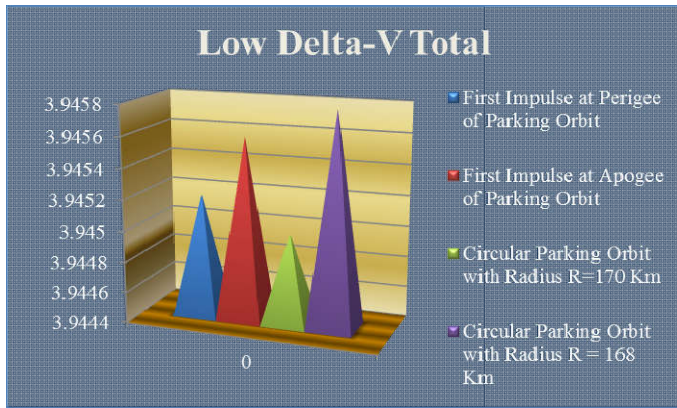


Figure 6. Represents circular parking orbit with R= 170km has low Delta-V Total

Conclusion

From the considered four cases, circular parking orbit with radius 170km is proved to have a low delta-V total. Therefore, It results in less fuel consumption. If launching a heavier payload into the GEO with less fuel consumption, helps to reduce the cost of the mission. Low delta-V budget is not only used for placing a satellite but also for mars missions and other future deep space missions. Continuation of the work will be carried out further.

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