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Cycler Orbit Between Earth and Mars

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A periodic orbit between Earth and Mars has been discovered that, after launch, permits a space vehicle to cycle back and forth between the planets with moderate maneuvers at irregular intervals. A space station placed in this cycler orbit could provide a safe haven from radiation and comfortable living quarters for astronauts en route to Earth or Mars. The orbit is largely maintained by gravity assist from Earth. Numerical results from multiconic optimization software are presented for a 15-year period from 1995 through 2010.

Introduction

NUMEROUS studies have been made involving circulating trajectories between planets, particularly between Earth and Mars.[1–7] Of special interest is an orbit first suggested by Aldrin,[8] which has been referred to as the Aldrin cycler. The Aldrin cycler is the simplest circulating orbit possible between Earth and Mars. Gravity assist at each Earth encounter is used as the main driver to periodically reshape the trajectory. The cycler has the special advantage of (nearly) regular encounters at Earth and Mars, which allows for much simpler mission planning.[9] A space station placed in this cycler orbit could provide a safe haven from radiation and comfortable living quarters for astronauts en route to Earth or Mars. The disadvantages are primarily that moderately large propulsive maneuvers are required at irregular intervals to maintain the circulating orbit and that the velocities at the planetary encounters are larger than those of other circulating trajectories.

In this paper a complete analysis of the Aldrin cycler is presented. Numerical results from a multiconic optimization program[10,11] provide the minimum ΔV solution for the 15-year period 1995–2010.

Analysis of the Cycler

Orbits of Earth and Mars

Earth and Mars revolve about the sun with orbit periods of 1 year and 1.8808 years, respectively. Earth travels in the ecliptic plane with an orbit eccentricity of 0.0168, whereas Mars travels in a plane inclined by 1.85 deg to the ecliptic with an eccentricity of 0.0934. To understand the properties of circulating orbits, we will ignore for the moment the noncircular noncoplanar motion of the real world. In addition, we will approximate the period of Mars by 1.875 years, so that every 15 years Earth makes 15 revolutions about the sun whereas Mars makes exactly eight revolutions. Since the relative geometry between Earth and Mars repeats with the synodic period of $15/7 = 2.1429$ years, trajectories connecting them will re-occur with this period also. The inertial geometry repeats every 15 years. These relationships have important implications in the design and repeatability of circulating orbits between these planets.

Circular Coplanar Analysis

The cycler type of circulating orbit evolved from the desire to have more frequent and regular encounters with both Earth and Mars. Since encounter locations on a 15/7 synodic period orbit rotate 1/7 of a circle per revolution, there exists a precession of the line of apsides equal to 51.4 deg per orbit that must be accommodated either by propulsive maneuvers (which are not desirable) or, to the extent possible, by gravity-assist swingbys of the planets (primarily Earth since it is more massive). Given this effective trajectory shaping, both Earth and Mars will be encountered sequentially every 2-1/7 years. The encounter speeds at both planets (but principally Mars), however, will be significantly higher than those of the VISIT orbits proposed by Niehoff[7] because the 2-1/7 year period transfer orbit has an aphelion distance of about 2.32 AU, thereby crossing the orbit of Mars at steeper (nontangential) angles.

The outbound (Earth-to-Mars) and inbound (Mars-to-Earth) cycler orbits defined here are essentially mirror images of each other. The outbound orbit will have a short transfer to Mars and a long transfer back, whereas the reverse is true for the inbound orbit. Figure 1 illustrates the concept of rotating the orbit. The space station leaves Earth at E1 and encounters Mars at M2 (for the outbound cycler). When the station crosses Earth's orbit again (slightly over 2 years later), Earth is not there but is re-encountered at E3 (2-1/7 years later). At E3, Earth's gravity rotates the orbit so as to encounter Mars at M4 while maintaining heliocentric energy. Similarly, the inbound cycler is defined by the orbit from E1 to M2′ to E3 to M4′. The advance of the perihelion per orbit is defined as $\Delta\psi = 2\pi/7$ rad.

In general, four orbital elements are required to specify an orbit in a plane. If the set (a, e, ω, and T) is chosen, then ω, the argument of periapsis, can be arbitrarily set to zero since the orbits of Earth and Mars, for this approximation, are taken to be concentric circles. The epoch of periapsis T will be specified by the requirement of encountering Earth and Mars. This leaves only the eccentricity e and the semimajor axis a to be determined.

From Fig. 1, it is seen that the magnitude of the true anomaly at Earth encounter θ is exactly one-half of the required advance of perihelion, $|\theta| = \Delta\psi/2$. Thus, Earth is encountered at a true anomaly of 25.7 deg, and the flyby changes

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the true anomaly to -25.7 deg. The net effect is to rotate the orbit by 51.4 deg. It is interesting to note that the shape of the transfer orbit can be obtained by using the conic equation:

$$r = a(1 - e^2)/(1 + e\cos\theta) \tag{1}$$

where $r = 1$ AU and $\theta = -25.7$ deg. If a is approximated by assuming a period of 2 years

$$a \approx (2)^{2/3} = 1.59 \text{ AU}$$

then by solving the quadratic equation in e we have the approximation for eccentricity

$$e \approx 0.387$$

The exact solution for the circular coplanar problem is found by solving the Lambert problem.[12,13] For this particular case we note that a "multirevolution" solution is necessary, since the spacecraft completes more than one revolution about the sun in traveling from E1 to E3. By specifying initial and final radius vectors of 1 AU, with a transfer angle of $\Delta\psi = 51.4$ deg, we obtain

$$a = 1.60 \text{ AU} \tag{2}$$

$$e = 0.393 \tag{3}$$

with an orbit period of 2.02 years.

Having specified the heliocentric cycler orbit, the characteristics of Earth flyby can be determined. Figure 2 illustrates the rotation of the heliocentric velocity vector from V_{in} to V_{out}, resulting in the change in heliocentric velocity ΔV. The flight-path angle γ is found from

$$\tan\gamma = \{er/[a(1-e^2)]\}\sin\theta \tag{4}$$

Substituting values for a and e from Eqs. (2) and (3) and $r = 1$ AU into Eq. (4) gives $\gamma = 7.18$ deg. From Fig. 2 it is seen that the magnitude of ΔV is

$$\Delta V = 2V\sin\gamma \tag{5}$$

where $V = |V_{\text{in}}| = |V_{\text{out}}|$. The magnitude of the heliocentric velocity V at Earth is found from the energy equation:

$$V = \mu^{1/2}[(2/r) - (1/a)]^{1/2} = (\mu/r)^{1/2}(2 - r/a)^{1/2} \tag{6}$$

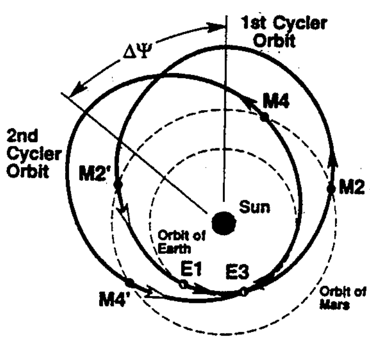

Fig. 1 Cycler orbit diagram.

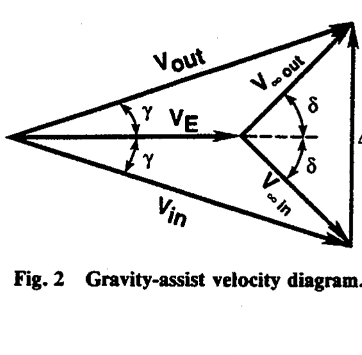

Fig. 2 Gravity-assist velocity diagram.

For Earth, $V_E = (\mu/r)^{1/2} = 29.8$ km/s, where μ is the gravitational parameter of the sun, and so $V = 34.9$ km/s. Substituting values into Eq. (5) gives $\Delta V = 8.73$ km/s. Again from Fig. 2, the excess speed is given by

$$V_\infty = (V^2 + V_E^2 - 2VV_E\cos\gamma)^{1/2} \tag{7}$$

which in turn provides $V_\infty = 6.54$ km/s. The flyby of Earth turns the V_∞ by an angle of 2δ. The equation relating this turn angle and the radius of closest approach r_p is given by

$$\sin\delta = 1/[1 + (r_p V_\infty^2/\mu_E)] \tag{8}$$

where μ_E is the gravitational parameter of Earth. From Fig. 2 we have

$$\Delta V = 2V_\infty \sin\delta \tag{9}$$

From Eqs. (8) and (9) we find that $r_p = 4640$ km and the turn angle $2\delta = 83.8$ deg. Since Earth's radius is 6371 km, this value of r_p corresponds to a flyby of Earth beneath the surface. To have a physically realizable flyby, the space station must therefore perform a propulsive maneuver somewhere on the orbit. Using techniques described in the next section, it was found that for the circular coplanar problem a relatively small maneuver of about 230 m/s performed at aphelion of the cycler orbit can rotate the argument of periapsis by an amount sufficient to make the flyby of Earth occur at 1000 km above the surface.

Numerical Results

With an understanding of the cycler orbit from the circular coplanar problem, a search was made to find the equivalent trajectories based on actual planet ephemerides. The trajectories were begun in the 1995/96 time period. Initially, a point-to-point conic model was used to identify the opportunities and verify their existence in the real world. In this model heliocentric conics connect the flybys of Earth and Mars with the times of flyby varied to obtain a match of V_∞ magnitude at each flyby. Thus a sequence of flybys similar to those of the circular coplanar model is obtained (E1-M2-E3-M4 . . .), the difference being that the real world geometry does not repeat exactly. These sequences over a 15-year period were then used as first guesses for a computer program that uses the more accurate multiconic propagation technique along with optimization techniques[10,11] to generate a trajectory with minimum propulsive maneuvers subject to constraints on flyby altitudes. In this application the most important constraint was to maintain a minimum flyby altitude of 1000 km at each Earth encounter.

Figures 3 and 4 illustrate the circulating orbits for both cyclers as they evolve over a complete 15-year cycle. Corre-

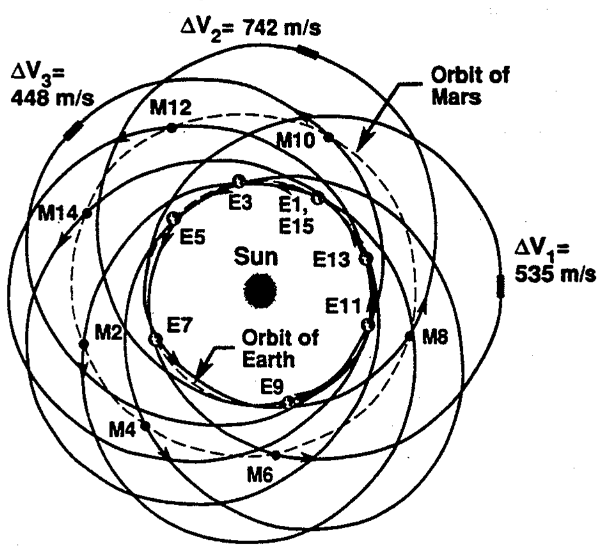

Fig. 3 Fifteen-year propagation of outbound cycler orbit.

sponding data on hyperbolic approach speed, closest approach distance, and propulsive ΔV requirements are listed in Tables 1 and 2. The orbit plots show the characteristic rotation of the encounter locations caused by the planetary flybys. (Note that each planetary encounter is indicated by an arrow along the trajectory path; the absence of an arrow indicates that no encounter occurs.) Because the orbital motion of Mars is not constant along its real elliptical path (and similarly for Earth to a lesser extent), the longitudinal difference between encounters is likewise not constant, nor are Earth-Mars and Mars-Earth flight times. This variability is related to different bending requirements of the trajectory during the successive planetary swingbys. The flight-time variation is not large: 147–170 days for both Earth-to-Mars transits on the outbound cycler and for Mars-to-Earth transits on the inbound cycler.

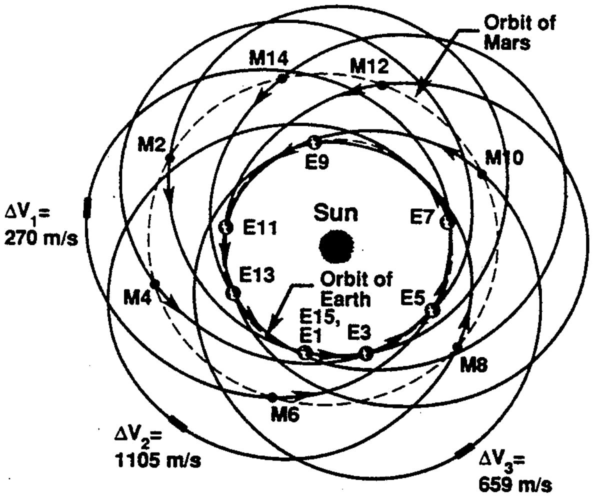

Fig. 4 Fifteen-year propagation of inbound cycler orbit.

Table 1 Outbound cycler encounter conditions

Encounter	Date	Approach V_∞, km/s	Closest approach distance, planet radii
Earth-1	19 Nov. 1996	6.19 (launch)	—
Mars-2	01 May 1997	10.69	5.8
Earth-3	01 Jan. 1999	5.94	1.3
Mars-4	28 May 1999	11.74	29.1
Earth-5	08 Feb. 2001	5.67	1.2
Mars-6	06 July 2001	10.22	1.3
Maneuver	13 March 2002	0.54 (ΔV_1)	—
Earth-7	16 April 2003	5.67	1.2
Mars-8	12 Sept. 2003	7.28	1.3
Maneuver	17 May 2004	0.74 (ΔV_2)	—
Earth-9	07 July 2005	5.87	1.2
Mars-10	13 Dec. 2005	6.05	3.4
Maneuver	23 July 2006	0.45 (ΔV_3)	—
Earth-11	06 Sept. 2007	5.87	1.8
Mars-12	16 Feb. 2008	7.43	6.4
Earth-13	10 Oct. 2009	5.89	1.9
Mars-14	28 March 2010	8.66	5.0
Earth-15	13 Nov. 2011	5.81	1.9

Table 2 Inbound cycler encounter conditions

Encounter	Date	Approach V_∞, km/s	Closest approach distance, planet radii
Earth-1	05 June 1995	5.88 (launch)	—
Mars-2	20 Jan. 1997	8.52	5.5
Earth-3	09 July 1997	5.95	1.8
Mars-4	07 March 1999	7.35	9.4
Earth-5	17 Aug. 1999	6.01	1.4
Maneuver	28 Sept. 2000	0.27 (ΔV_1)	—
Mars-6	15 May 2001	6.60	5.2
Earth-7	08 Oct. 2001	5.88	1.2
Maneuver	04 Dec. 2002	1.11 (ΔV_2)	—
Mars-8	07 Aug. 2003	7.30	1.3
Earth-9	02 Jan. 2004	5.39	1.4
Maneuver	02 Feb. 2005	0.66 (ΔV_3)	—
Mars-10	10 Oct. 2005	9.96	1.3
Earth-11	12 March 2006	5.48	1.5
Mars-12	19 Nov. 2007	11.59	8.4
Earth-13	16 April 2008	5.96	1.5
Mars-14	13 Dec. 2009	10.55	5.0
Earth-15	22 May 2010	5.93	1.8

Recall that in the circular coplanar problem, more bending is required of Earth than is available, so a propulsive maneuver is required on each orbit. In the real world, the optimal trajectory covering this 15-year cycle requires propulsive maneuvers on only three of the seven orbits. Interestingly enough, the sum of these three maneuvers is approximately seven times the per-orbit requirement of the circular coplanar problem. The propulsive maneuvers are made near aphelion on the cycler orbit, which occurs about eight months after Mars encounter on the outbound orbit and eight months before Mars encounter on the inbound orbit. Note that these maneuvers occur on the orbits that encounter Mars in the general region of its perihelion passage where its heliocentric speed is greatest.

Conclusions

The Aldrin cycler is the simplest circulating trajectory between Earth and Mars. It permits regular and frequent encounters with both planets and has the desirable characteristic of short transits (150–170 days) to Mars on the outbound cycler and, similarly, rapid returns to Earth on the inbound cycler.

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