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Cislunar (or Earth) Swingby for Interplanetary Missions

Cislunar swingby is the basic trajectory profile which initiates heliocentric flight from the Earth-Moon system by inception from cislunar space so that the spacecraft swings close by Earth prior to injection into its interplanetary orbit. In the present paper, the energetics of this Earth swingby are compared by means of parametric curves for the four alternative options of flight inception from a highly eccentric elliptic orbit about Earth, the cislunar libration point L_1, a lunar orbit, or the lunar surface.

In a previous paper, Gillespie compared the energetics of rockets launched directly into heliocentric space from the cislunar libration point with those of rockets launched from low circular orbits about the Earth. In this present paper, it is shown that, for a given payload, the fuel mass expenditure for perigee departure from a highly eccentric orbit about Earth is almost one-half that required for launching directly from the cislunar libration point. In both cases it is assumed that the rocket is refueled by reusable tanker rockets.

A comparison of the energetics for cislunar swingby (from lunar orbit), direct transfer, and planetary swingby (from low, circular, terrestrial orbits) shows that cislunar swingby alone offers fuel savings of about 50 percent over the other two profiles. In this comparison, only an outbound leg of heliocentric flight was considered. Additional savings could be realized by the combined use of cislunar and extraterrestrial planetary swingby profiles for a complete interplanetary flight.

It is indicated that the launch window for cislunar swingby will be significantly wider than that for direct or planetary swingby flight. For the last three options described, the launch opportunities occur in "sidebands" which are distributed over the launch window as a function of the lunar period.

INTRODUCTION

The vehicle staging design of an interplanetary spacecraft is principally determined by the onboard fuel mass required for all maneuvers from inception of outbound interplanetary flight until termination of inbound flight at terrestrial orbit or surface landing. Significant fuel savings may be provided by means of planetary swingby (see ref. 1) during interplanetary flight. In addition, considerable fuel economies may be realized by effective refueling within the Earth-Moon system prior to inception of the outbound interplanetary flight. Gillespie (ref. 2) has compared the energetics of rockets launched directly into heliocentric space from the cislunar libration point, with rockets launched from low circular orbits about the Earth. In general, Gillespie has demonstrated that refueling modes within the Earth-Moon system are very advantageous and enable the design efficiency of the interplanetary vehicle staging to be impressively increased. Refueling may occur from the Moon, assuming the availability of lunar resources by lunar-surface operations, or from the Earth. The availability of the Apollo system for the retanking capability offers a realistic basis for the immediate and planned use of trajectory profiles using refueling prior to inception of the interplanetary flight.

Study of the dynamic principles behind planetary swingby and cislunar retanking indicates that a significant increase in fuel savings is generally realized by flight inception from cislunar space, remote from the Earth's surface, after refueling so that the interplanetary vehicle swings close past the Earth for injection then into heliocentric orbit. Aside from the increased efficiency of the rocket energetics, the attendant flight characteristics and subsequent system-design objectives appear quite favorable in several major aspects. This cislunar (or Earth) swingby for interplanetary missions is explored briefly in this paper by description and discussion of its outstanding performance characteristics. The realizable fuel savings are estimated as a function of the alternative and selectable classes of flight inception point, which may be located at various stable dynamic sites in cislunar space.

Cislunar swingby past Earth for launch into heliocentric orbit may be provided, as shown schematically in figure 1, from any one of four options of flight inception points:

(1) Perigee of a highly eccentric Earth orbit
(2) Cislunar libration point
(3) Lunar orbit
(4) Lunar surface

Refueling at the cislunar libration point or in lunar orbit might be provided from Earth or the Moon. Flight from the lunar surface would best utilize refueling from lunar resources, whereas refueling on the highly eccentric Earth orbit could be provided from Earth, from a cislunar-libration-point station, or even from the Moon.

The subsequent heliocentric flight in interplanetary space after cislunar swingby may be either direct to the target planet (or asteroid) or by extraterrestrial planetary swingby.

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ROCKET ENERGETICS

The study of the complete interplanetary flight may be conducted in two discrete and successive "legs": outbound to the target planet (or planets) and return, if required. In this paper, for brevity, only the outbound leg is discussed, so that the available parametric tradeoffs are demonstrated. The outbound leg can be considered in two discrete steps:

(1) The rocket energetics within the Earth-Moon system until transition into the desired heliocentric orbit.

(2) The heliocentric transfer energetics from the point of Earth-Moon system injection into heliocentric orbit, until capture (propulsive or atmospheric), or fly by the target planet.

The rocket energetics for cislunar swingby out of the Earth-Moon system (step (1) above) from each of the four alternative inception points are presented in figures 2 and 3, in parametric form, for specific impulse $I_{sp}=444$ seconds. Comparable sets of parametric curves may be generated for other realizable values of specific impulse. The parametric format for mass-ratio tradeoff used in figure 3 is Gillespie's concise method (ref. 2) of presenting energetics working curves for system synthesis and evaluation.

With all four cases, the required velocity increment and the remaining mass (expressed in percent of initial mass) are plotted against hyperbolic excess speed in EMOS (i.e., units of Earth-mean-orbital-speed), where 1 EMOS = 29.8 km/sec. Note that these working curves are dissociated from staging design variations, since the remaining mass includes the onboard fuel left after injection into heliocentric orbit, the basic rocket staging, and the payload. Tradeoff between fuel, staging, and payload of the remaining mass can then be separately evaluated and optimized.

The total velocity increment for each alternative inception of cislunar swingby was determined in accordance with the equations given below:

(1) From Earth-orbit perigee:

$$\Delta V = \Delta V_{\oplus\male} = \sqrt{V_{e}^{2}+V_{HEEV}^{2}} - V_{swby} \qquad (1)$$

where

V_e parabolic escape velocity from Earth $=(2r_{\oplus}/R_{\oplus})^{1/2}$
V_{HEEV} hyperbolic excess velocity = (heliocentric transfer injection velocity) − (Earth's mean heliocentric velocity)
V_{swby} Earth swingby velocity $=V_P$
$R_{\oplus}$ Earth's spherical radius

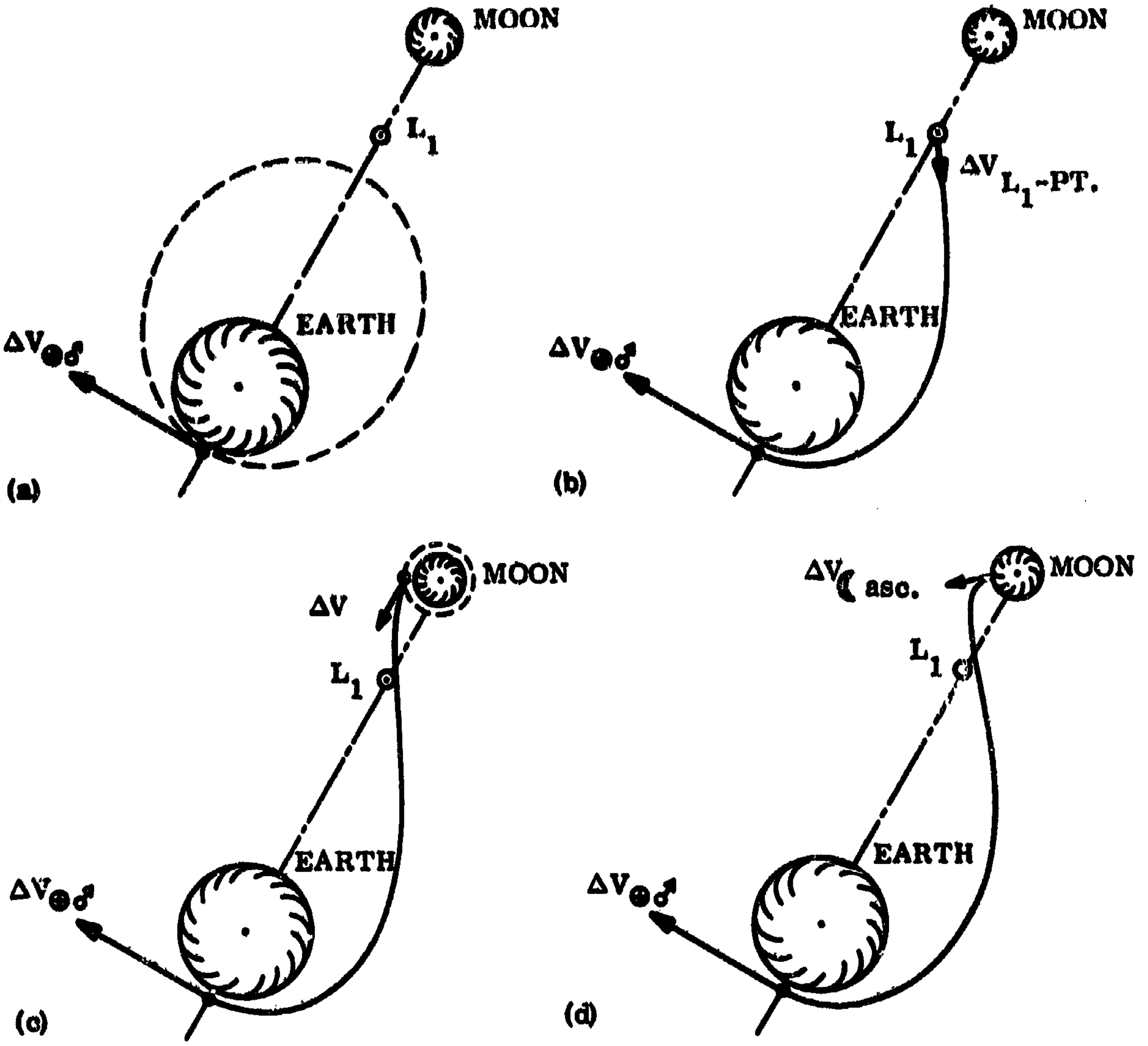

FIGURE 1.—Options of cislunar swingby inception (schematic only).

(*a*) Departure from perigee of highly eccentric orbit about Earth.
(*b*) Departure from cislunar libration point L_1.
(*c*) Departure from lunar orbit.
(*d*) Departure from lunar surface.

(2) From cislunar libration point L_1:

$$\Delta V_2 = \Delta V_{L_1\text{-}PT} + \Delta V_{\oplus \text{♂}} \tag{2}$$

where

$\Delta V_{L_1\text{-}PT}$ velocity increment to leave L_1 to swing by Earth, approximated by the apogee velocity V_A of the highly eccentric Earth orbit

(3) From lunar orbit:

$$\Delta V_3 = \Delta V_{\text{☾}} + \Delta V_{\oplus \text{♂}} \tag{3}$$

where

$\Delta V_{\text{☾}}$ velocity increment needed to leave lunar orbit to swing by Earth, approximated by the difference between the lunar escape velocity and the lunar orbital velocity $= V_{\text{☾}e} - V_{\text{☾}o}$

(4) From lunar surface:

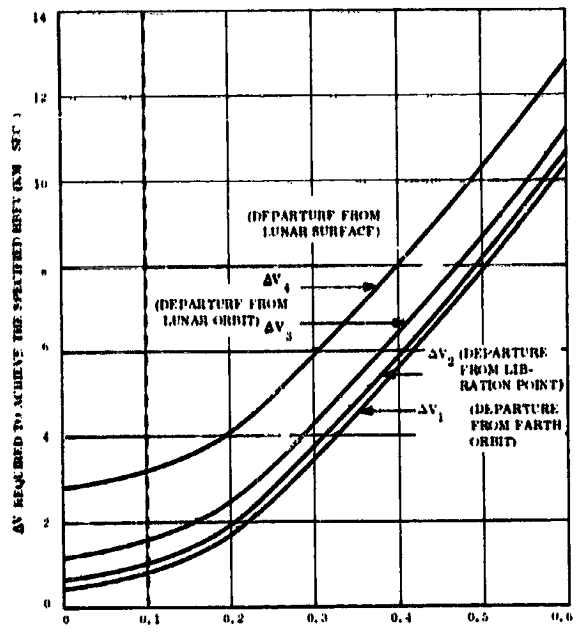

FIGURE 2.—Required velocity increments for alternative cislunar swingby options. $I_{sp}=444$ seconds.

$$\Delta V_4=\Delta V_{\text{☾} asc}+\Delta V_{\oplus\sigma} \tag{4}$$

where

$\Delta V_{\text{☾} asc}$ velocity increment to ascend from the lunar surface to swing by Earth, approximated by the lunar escape velocity $V_{\text{☾}}$.

These velocity increment equations approximate the actual fuel expenditure, by use of the conventional conic approximation of impulsive transition of a spacecraft from the gravisphere domain of one force center into that of a second force center. The Earth orbit for perigee inception (case (1)) was selected as a highly eccentric orbit which lies realistically within the terrestrial gravisphere. The definitive orbital data are as follows:

- r_A orbit apogee, 300×10^3 km
- r_P orbit perigee, 6.7×10^3 km
- a orbit semimajor axis, 153.35×10^3 km
- e orbit eccentricity, 0.956

In swingby inception from the cislunar libration point L_1 (case 2), the velocity increment to leave the libration point was approximated by

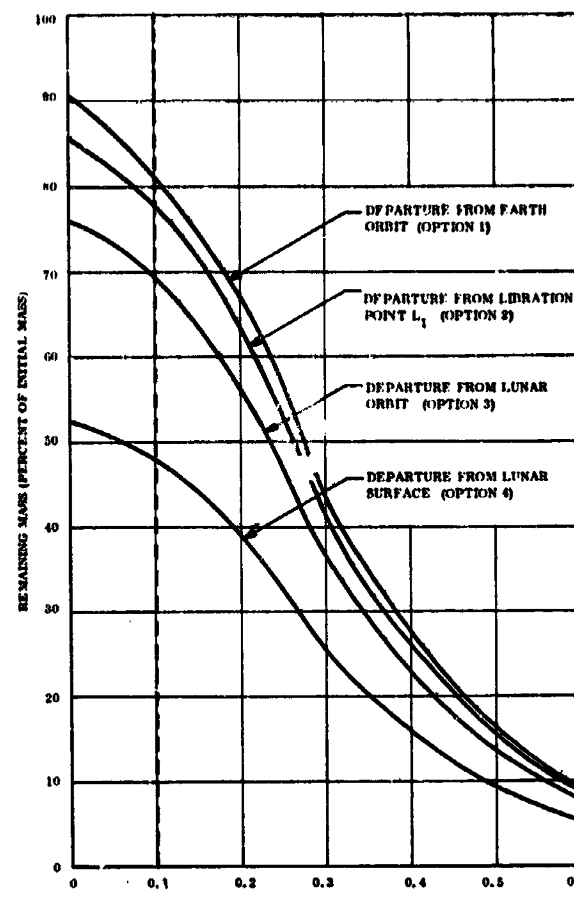

FIGURE 3.—Remaining mass ratio for alternative cislunar swingby options. $I_{sp}=444$ seconds.

the apogee velocity ($V_A=0.24$ km/sec) of the highly eccentric Earth orbit, in order to assure a reasonable transit time from inception until swingby. In cases (3) and (4), the velocity increments for injection into cislunar flight to Earth swingby were approximated by use of the lunar escape velocity, since the hyperbolic velocity may vary for different lunar orbital altitudes and Earth swingby altitudes. In general, the actual additional velocity will vary from about 0.21 to 0.25 km/sec, depending upon the hyperbolic velocity to be attained on cislunar departure.

Nonideal losses such as with finite-time thrust or with nonoptimal deviations from ideal flight parameter values will entail added losses which characterize the attainable efficiency of a given spacecraft system design. However, these losses and the above approximations do not affect the basic parametric variation of the energetics for the various Earth swingby modes available.

Obviously, departure from the highly eccentric Earth orbit is most advantageous and necessitates the smallest velocity increment to enter heliocentric orbit (fig. 2) so that the remaining mass is maximum (fig. 3). At zero hyperbolic excess speed, the relative ratios of remaining mass fractions are

$$m_1:m_2:m_3:m_4=1.75:1.65:1.45:1$$

These relative ratios are valid over the complete range of hyperbolic excess speed shown. Two salient features are evident:

(1) A very large penalty is imposed for lunar-surface launch, whereas all other inception point profiles lie within a much narrower envelope.

(2) The mass expenditure for Earth-orbit departure is only 0.64 of that for libration point departure.

The heliocentric velocity required upon exit from the Earth-Moon system will be determined by the transfer orbit for rendezvous with the target planet. In general, the heliocentric transfer orbit which requires the least total fuel expenditure between given endpoints (i.e., Earth at the time of departure and the target planet at the time of spacecraft arrival) is determined as the smallest positive real root of an eighth-order polynomial with constant coefficients (refs. 3 and 4). That is, all possible solutions of minimum-fuel heliocentric transfer orbits can be presented in parametric form. It can be visualized that Earth swingby will enable a lower total fuel expenditure than that of the corresponding direct transfer provided by heliocentric transfer from a terrestrial close circular orbit. Moreover, Earth swingby enables even a smaller fuel requirement than that with planetary swingby. As a simple example indicative of this favorable tradeoff, let us consider the corresponding velocity increments required for outbound flight by direct, planetary swingby and cislunar swingby (from lunar orbit) modes for transit to and propulsive capture by Mars. Direct and cislunar swingby are based upon standard opposition conditions of heliocentric transfer, whereas planetary swingby utilizes Venus en route to obtain the representative results reported in reference 5. All other definitive conditions are as follows:

Lunar circular orbit altitude =161 km (100 miles)
Terrestrial circular orbit altitude=300 km (186 miles)
Martian circular orbit altitude =800 km (496 miles)

The outbound transit data are as shown in table 1. Note that two different sets of standard opposition and Venus swingby are presented; these illustrate the tradeoff between trip time and total velocity increment ($\Sigma\Delta V$). In general, cislunar swingby requires about 50 percent of the velocity increment for either direct or Venus swingby, with a trip time almost the same as for direct transfer. Note that this heliocentric transfer requires about 0.1 EMOS to enter heliocentric space, as indicated by the broken vertical lines in figures 2 and 3.

TABLE 1.—*Velocity Increments for Alternative Basic Trajectory Profiles*

Flight mode	Trip time, days	$\Delta V_{LV\oplus}$, fps	$\Delta V_{AR\male}$, fps	$\Delta V_{outbound}$, fps
Direct	460	12 365	11 555	23 920
Outward Venus swingby	680	14 095	11 165	25 260
Direct	420	13 200	15 240	28 440
Outward Venus swingby	500	14 090	13 760	27 850
Cislunar swingby	Standard +3	3 444	11 395	14 839

Of course, the Earth and planetary swingbys could be combined to provide remarkable total fuel savings with almost the same trip time as for planetary swingby alone. It is noteworthy that computer study of planetary swingby solutions can be expedited by analytic approximation of this flight mode by a three-impulse orbital transfer optimization as shown in reference 6. In this reference, analytic formulation and partial reduction for solution of the three-impulse transfer optimization was accomplished. In planetary swingby, the gravity potential influence of the swingby planet (such as Venus) could be effectively approximated as the intermediate (or second) impulse. This analytic aid to mapping of the solution field can expedite future swingby studies, especially of hybrid swingby profiles (i.e., cislunar followed by planetary swingby). Significant reductions in solution running time can be expected.

INFLUENCE OF ORBITAL PLANE INCLINATION

Since the planets and other major bodies within the solar system lie in different orbital planes inclined to the ecliptic, the nonplanar (or three-dimensional) nature of the actual interplanetary transfer must be considered in evaluation of the energetics for heliocentric trajectories. The inclination to the ecliptic of the major bodies of immediate interest are as follows:

Body	Orbital plane inclination, deg
Mercury	7.00
Venus	3.38
Earth	0
Mars	1.85
Jupiter	1.31
Saturn	2.49
Earth's Moon	5.15

In view of the relatively large scalars of interplanetary transfer velocities, these inclinations can cause significant differences in fuel expenditure for accommodating maneuvers, even though the inclination angles appear to be small. However, the orbital inclination of the Moon may be employed to advantage in reducing or eliminating the nonplanar fuel penalty. Of course, the time of trajectory inception must be properly selected in order to utilize this inherent capability of cislunar transit within the Earth-Moon system in the course of Earth swingby.

Note that Earth swingby by inception from a highly eccentric Earth orbit need not necessarily occur within the plane of Earth-Moon system rotation, since the Earth orbit could be inclined to it, if desirable or realizable from the available launch and range facilities.

LAUNCH WINDOW AND FLIGHT OPPORTUNITIES

In general, spacecraft mass limitations upon the permissible fuel expenditure impose a severe operational constraint upon the launch window. This critical constraint results in the restriction, for current system capabilities, of interplanetary missions to a few crucial "opportunity years." It is eminently desirable to be able to "open the launch window" for mission launch from the Earth (or Earth-Moon system).

The cislunar swingby would obviously enable this launch window problem to be alleviated by virtue of the fuel savings indicated by the parametric curves and examples of the preceding sections. However, the cislunar swingby has another effect upon the launch window because of the recurrence of launch opportunity with lunar period. That is, direct and planetary swingby launch windows are principally determined by the Earth-orbit period and the relative phasing between planetary ephemerides, whereas the cislunar swingby window is determined principally by the lunar period and the relative phasing between planetary ephemerides. The structure of the launch window is shown schematically in figure 4, which presents the tolerance in delay time at any given launch opportunity as a function of the launch time. That is, a characteristic tolerance band for timing of launch into heliocentric orbit is available at each launch opportunity. Note that the delay tolerance is of the order of minutes, whereas the total width of the apparent launch window is of the order of months.

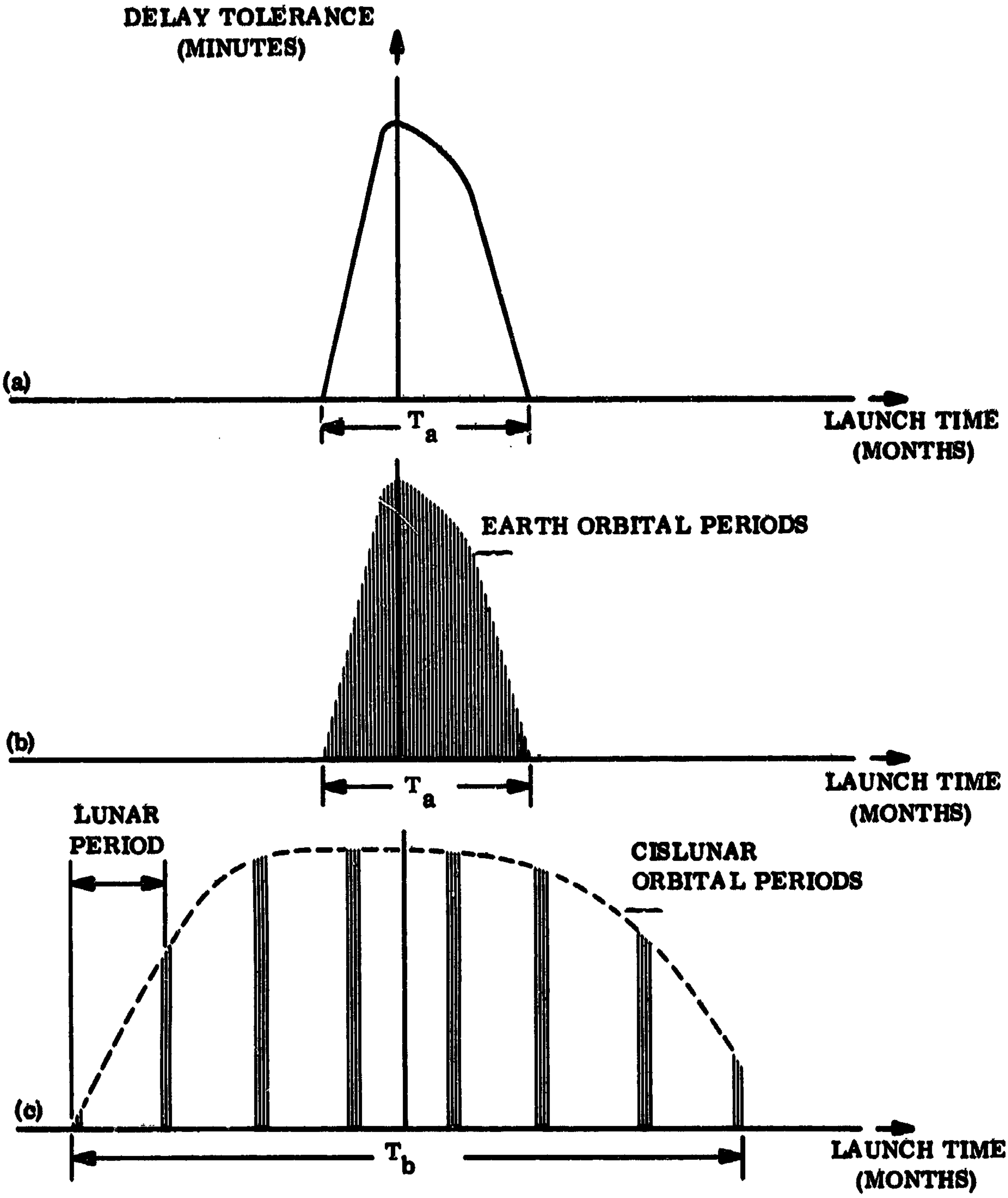

FIGURE 4.—Structure of launch window (schematic only).

(*a*) Apparent launch window; direct or planetary swingby.
(*b*) Actual launch window; direct or planetary swingby.
(*c*) Actual launch window; cislunar swingby, cases 2 to 4.

Figures 4(*a*) and 4(*b*) describe the launch windows for direct and planetary swingby in schematic form. As shown in figure 4(*a*), the "apparent" launch window is defined by the envelope of actual launch opportunities. The actual opportunities will recur with Earth-orbit period, as shown in figure 4(*b*); that is, the launch opportunities (presented discretely along the abscissa) are separated from one another by the period of the Earth orbit from which launch occurs, in the order of hours. The launch window for cislunar swingby is shown schematically in figure 4(*c*) to the same relative scale of launch time. The major "sidebands" of launch opportunities recur with the lunar period, whereas the discrete opportunities may occur at any time within them. It is estimated that the cislunar swingby spreads the envelope or timespan of launch opportunities in the relative ratio of $T_a : T_b$ as shown schematically in figure 4, and with the periodic recurrence pattern indicated in figure 4(*c*).

The delay tolerance for cislunar swingby appears to be about that for direct and planetary swingby, as indicated by the ordinate values of figure 4. However, brief consideration suggests that the cislunar swingby delay tolerance might be significantly larger. Of course, the dynamic model is more complex and analytical knowledge of the variational characteristics of the useful classes of cislunar orbits must be extended in order to determine this trajectory characteristic conclusively.

SYSTEM DESIGN AND MISSION OPERATIONS

The flight performance characteristics of a trajectory profile must fulfill the requirements of the available space system designs and mission operations. In addition to favorable energetics, flight time, and launch windows, Earth swingby has several other attractive aspects.

Earth approach and swingby for subsequent injection into heliocentric orbit will occur within Earth's planetary test range, which consists of the extremely accurate command and operations complexes (tracking stations, operations control center, etc.) used for space missions. The solid base of the space program is the Apollo system complex and missions. In general, the Apollo system and operations will be available and proven for hyperbolic approach phases within the near future.

In particular, the most sensitive system design performance and operations problems encountered in cislunar swingby will necessarily have been solved in the Apollo mission. Moreover, the operational experience and procedures will be almost directly applicable to the interplanetary mission with the cislunar swingby profile. The Apollo guidance and control system will solve the critical guidance and control requirements of swingby around Earth. Although the high approach velocity requires great control accuracy, swingby will be extra-atmospheric and can be carried out under optimum system conditions of ground tracking and joint onboard/ground control of injection into heliocentric orbit. Moreover, a number of alternative abort options would be available well at the start of the complete trajectory so that system and crew recovery will be possible with minimal risk or hardware loss.

The transport system, payload instruments, and tests can be exercised as the spacecraft approaches Earth for subsequent swingby and injection into heliocentric orbit. Consequently, prior calibration of all onboard subsystems with the actual working instruments is possible at the beginning of the long interplanetary flight. Any observed deficiencies can then be corrected by the operations team (on the Earth and/or onboard) prior to arrival at the target planet. In particular, the experiment scenario of the Earth swingby test exercise of payload operations can yield invaluable evaluation criteria for subsequent data processing and reduction of the target-planet observational data.

The selection of the inception point for the interplanetary flight leg will be determined by the costs, risks, and many design considerations for the given mission. The spectrum of design characteristics implied by each of the four alternative options is sufficiently broad to enable the selection and optimization of an effective system for various missions.

SUMMARY

The four alternative options of the cislunar

(or Earth) swingby trajectory profile for interplanetary flight have been compared briefly by means of parametric curves of the energetics for departure from the Earth-Moon system. While the fuel expenditure for flight inception from the lunar surface is significantly greater than for the other three inception options, the use of perigee departure from a highly eccentric elliptic orbit about the Earth is most favorable. Refueling may be accomplished in the neighborhood of apogee, either from the Earth, a cislunar libration station, or the Moon.

Fuel savings of about 50 percent over direct transfer or planetary swingby alone are indicated with cislunar swingby from lunar orbit (option 3) and subsequent direct heliocentric flight. Additional savings by the combined use of cislunar and planetary swingby profiles could be realized.

The inclination of the orbital plane of the Earth-Moon system to the ecliptic may be advantageous in minimizing the spacecraft energetics, provided the required endpoint conditions for heliocentric transfer in three-dimensional space can be realized.

The launch-window envelope and opportunities have been described as functions not only of the relative phasing between the planetary ephemerides (of the Earth and target planet), but also of the period of the Moon about the Earth. In general, the timespan of the Earth swingby window will be significantly greater (than that of direct or planetary swingby flight) with more frequent discrete launch opportunities, although they occur only within sidebands of lunar periodic frequency. However, heliocentric injection from a highly eccentric terrestrial orbit will depend upon the spacecraft's orbital period rather than on the lunar period about the Earth.

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