Orbit Design Concepts for Jupiter Orbiter Missions

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Advanced mission and orbit planning efforts are in progress for a Jupiter orbiter mission in the early 1980's. Baseline spacecraft and orbit design criteria are the goals of a NASA effort to define such a mission. Orbit design concepts that have been discovered during the early stages of mission planning are both challenging and exciting. This paper is a description of several such concepts that may greatly increase the flexibility and scientific return of orbiters designed for close study of the Galilean satellites and exploration of the Jovian system. Some new jargon is introduced in discussions to describe the exploitation of gravity-assist trajectories using the giant satellites for orbit control. Orbit pumping and cranking and resonance hopping are defined and shown to be dynamically feasible means of controlling the orbit and, thus, the scientific return. A candidate encounter sequence is presented for an equatorial tour of the Galilean moons. The analytic development of the orbit design concept is performed by zero-sphere of influence patched conics. The navigation feasibility and the accuracy of this work are discussed and shown to yield high confidence in the applicability of the concepts to the ultimate mission design.

Introduction

In recent years, substantial efforts have been directed toward definition of and advance planning for a Jupiter orbiter-satellite tour mission aimed at extensive study of Jupiter, its satellites, and its environment.

The preliminary mission analyses associated with these efforts have revealed gravity-assist techniques for orbit insertion and control that are not particularly surprising in fundamental theory but are remarkable in degree. Of more importance is the discovery that the combination of these techniques, particularly in repeated transfer from one satellite to the other, opens up possibilities that seem incredible to experienced analysts accustomed to ballistic missions with chemical propulsion or single swingby missions.

In pursuit of a baseline satellite tour mission, we have discovered such a multiplicity of possible encounter sequences that new ways of thinking about orbit mission analysis have evolved. Part of this evolutionary process has been the coining of new terminology that helps describe the methods of satellite-to-satellite transfer that have proved effective in the search for encounter sequences that simultaneously enhance performance and achieve various scientific objectives. These concepts are useful not only for semantic reasons but also, and much more importantly, for the practical evaluation of what is possible and what is not.

This paper is a description of several of the more useful concepts of gravity-assist orbit control and a parallel discussion of how the techniques are being used to establish Jupiter orbiter mission profiles that have previously been considered outrageously expensive, technically unfeasible, or downright science fiction.

The following discussion is divided into three sections. In the section on gravity-assist concepts, descriptions of orbit cranking, orbit pumping and resonance hopping are accompanied by a discussion of how the method of analysis is based on a firm theoretical foundation that conserves the Jacobian integral in the circular restricted three-body problem. Also presented in this section is a discussion of what we call $n\pi$ transfers. These are transfer trajectories that require an integer multiple of 180 deg for the transfer angle.

A second section includes a discussion of the types and magnitudes of the perturbations, some comments on the importance of the navigation problem with estimates of correction maneuver requirements, and some performance considerations that serve as background for a later discussion of mission design concepts.

A final section is devoted to discussion of how the ideas are being used in mission design. The orbit control available through gravity assist is cited as an excellent way to achieve many simultaneous science objectives, and an equatorial encounter sequence is presented as a candidate in the mission design.

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Gravity-Assist Concepts

The study of the use of the gravity of the Galilean satellites to control the orbit of the Jupiter orbiter spacecraft has led to a number of concepts which, while not theoretically new (see, for example, Refs. 1-3), represent remarkable advances in design practice. These concepts have previously been bypassed either because they appeared to represent an insignificant set of cases requiring special computer programs, or because they seemed impractical when used on the scale of the solar system, or because Venus and Mars have no gravitationally useful satellites available. The concepts we wish to discuss are orbit pumping and cranking, resonance hopping, and $n\pi$ transfers. These are defined roughly here and discussed more fully in the succeeding paragraphs. Orbit pumping, a term coined by Beckman,[4] is use of satellite gravity to change the energy of a spacecraft orbit. Orbit cranking is the use of satellite gravity to change the inclination or flight path angle at encounter without changing the spacecraft energy. $N\pi$ transfers are trajectories between satellites where the transfer angle about Jupiter is an integer multiple of π, and resonance hopping is a form of orbit pumping using $n\pi$ transfers where n is an even integer.

Geometry of Zero-Sphere-of-Influence Patched Conic

The zero-sphere-of-influence patched conic method treats the encounter of a spacecraft with a satellite as a collision of two point particles which conserves energy in the frame of the satellite but impulsively changes the spacecraft velocity. The results of the method are accurate enough for preliminary mission analysis, and the method is easy to analyze. The concepts of this section result from such analysis.

Figure 1 shows the swingby geometry. The notation here will be used throughout the paper. Let $\boldsymbol{V}$ be the satellite's jovicentric velocity; $\boldsymbol{V}_{b,a}$, the spacecraft's jovicentric velocity; and $\boldsymbol{V}_{\infty b,a}$ the spacecraft's satellite-centered excess velocity before, after the swingby.§

Because the $\boldsymbol{V}_\infty$ vectors before and after have the same length, the locus of all possible $\boldsymbol{V}_\infty$'s after swingby lie on a sphere centered at the head of $\boldsymbol{V}$. If $\boldsymbol{V}_a$, $\boldsymbol{V}_b$ and $\boldsymbol{V}$ all lie in the same plane and if $V_b \neq V_a$ then the jovicentric orbital energy changes in the swingby. This is designated "orbit pumping." If $V_a = V_b$ but $\boldsymbol{V}_a \neq \boldsymbol{V}_b$, then the triangle $(\boldsymbol{V}_a, \boldsymbol{V}, \boldsymbol{V}_{\infty a})$ is congruent to $(\boldsymbol{V}_b, \boldsymbol{V}, \boldsymbol{V}_{\infty b})$ and is just rotated about $\boldsymbol{V}$ from $(\boldsymbol{V}_b, \boldsymbol{V}, \boldsymbol{V}_{\infty b})$. This sort of encounter, which does not change the jovicentric energy, we call "orbit cranking." Any actual encounter may be resolved into a pumping part and a cranking part.

Conservation of Jacobian Integral

In Fig. 1 the angle between $\boldsymbol{V}$ and $\boldsymbol{V}_b$ is arccos ($\cos i_b \cos \gamma_b$) where i_b and γ_b are the jovicentric inclination to the satellite orbit plane and the flight path angle before the swingby, respectively. From the triangle $(\boldsymbol{V}_b, \boldsymbol{V}, \boldsymbol{V}_{\infty b})$ we have $V_{\infty b}^2 = V_b^2 + V^2 - 2V_bV \cos i_b \cos \gamma_b$, and similarly for the triangle after. If the orbit of the swingby satellite is circular, then, because V_∞ is the same before and after, $V_b^2 - 2V_bn'r \cos\gamma_b \cos i_b = V_a^2 - 2V_an'r \cos\gamma_a \cos i_a$, where r is the radius of the swingby satellite orbit and n' is its mean motion. Subtraction of $2\mu/r$ from each side yields

$$E_b - n'h_b \cos i_b = E_a - n'h_a \cos i_a \qquad (1)$$

where E and h represent the jovicentric energy and angular momentum per unit mass of the spacecraft's orbit and μ is the gravitational strength (GM) of Jupiter.

Thus, we have arrived at the expression for the conservation of the Jacobian integral in the circular restricted three-body problem because Eq. (1) is a statement of that constancy for situations where the spacecraft is well away from the swingby satellite before and after having made an arbitrarily close approach to that disturbing body. Thus, we see that the method of analysis has a firm theoretical foundation insofar as the energy relationships are concerned.

Orbit Control

The satellite-centered orbit is just a hyperbola whose asymptotes are $\boldsymbol{V}_{\infty b}$ and $\boldsymbol{V}_{\infty a}$. Orbit control is achieved by control of the $\boldsymbol{V}_{\infty a}$ vector through suitable selection of the encounter conditions. Let the minimum permissible swingby radius be r_p, then the maximum angle α between $\boldsymbol{V}_{\infty a}$ and $\boldsymbol{V}_\infty$ is given by

$$\sin[\alpha/2] = 1/(1 + r_p V_\infty^2/\mu_s) \qquad (2)$$

where the satellite gravitational constant is μ_s.

The maximum ΔV available in the swingby is a useful quantity because ΔV is the same in both jovicentric and satellite-centered systems

$$\Delta V = 2V_\infty \sin[\alpha/2] = 2V_\infty/(1 + r_p V_\infty^2/\mu_s) \qquad (3)$$

It can be shown that ΔV cannot be greater than the velocity V_c of a circular orbiter about the satellite at radius r_p. This maximum can occur only when $V_\infty = V_c$, at which time $\alpha = 60$ deg. For cases of interest in the Jupiter orbiter analysis $V_\infty > V_c$ so $\Delta V < V_c$. Moreover, this ΔV caused by the swingby is approximately perpendicular to $\boldsymbol{V}_{\infty b}$ so it may not be compared to a propulsive ΔV except with due caution.

§Symbols in bold type represent vector quantities; the same symbols in light type represent the lengths of these vectors.

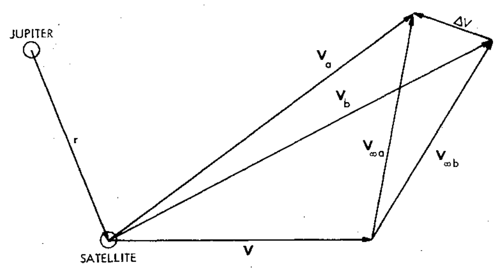

Fig. 1 Zero-patched conic geometry.

Figure 1 shows that if V_∞ is small then $\boldsymbol{V}_a$ cannot be much different from $\boldsymbol{V}$, so that the spacecraft jovicentric orbit cannot be much different from that of the satellite. On the other hand, if $V_{\infty b}$ is large, then ΔV is small by Eq. (3) and $\boldsymbol{V}_{\infty a}$ must be close to $\boldsymbol{V}_{\infty b}$ in a given swingby. ΔV is often small in the Jupiter orbit analysis, but a large ΔV may still be accumulated with several swingbys. It should be noted that if $V_\infty < V$, then the inclination must be less than 90 deg, and a retrograde orbit after swingby is impossible. Also, if $V_\infty < (\sqrt{2} - 1)V$, then escape from Jupiter after swingby is impossible.

In the preceding paragraphs, the basic gravity-assist concepts have been defined and the most useful auxiliary equations derived. The following paragraphs extend these ideas to more specific areas of interest to Jupiter orbiter missions. Reference 5 presents these concepts in much greater detail.

Even $n\pi$ Transfers: Resonance Hopping and Orbit Cranking

A great deal can be learned from studying transfers with central angles of $n\pi$ with n even. The transfers are, thus, from a satellite to itself. When the orbital period of a spacecraft is a simple rational multiple of that of a satellite, it is said to be in resonance with the satellite. The object of resonance hopping is to change the orbital period from one resonance to another by an encounter with the satellite itself.

In resonance hopping a relation exists for the encounter velocity of a spacecraft making successive $m \times 360$-deg transfers to successive encounters with a satellite in a circular orbit about Jupiter. The period of the spacecraft P_{sc} must certainly be a rational multiple of that of the satellite: $P_{sc} = n/m\ P_s$. Let the corresponding spacecraft orbital semimajor axis be $a_{n,m}$ then $a_{n,m}/a_{1,1} = (n/m)^{2/3}$. Now the encounter distance r is just $a_{1,1}$ because the satellite orbit is circular, so the vis viva equation for the speed $V_{n,m}$ of the spacecraft gives

$$V_{n,m}/V = \sqrt{2 - (m/n)^{2/3}}$$

The important thing to realize here is that resonance hops correspond to jumping from one velocity to another, and the difference between any two ratios is a measure of the ΔV required and, thus, the difficulty of achieving the transfer.

To resonance hop to resonance n,m it is necessary to achieve an outgoing jovicentric speed equal to $V_{n,m}$. This means that all $\boldsymbol{V}_{n,m}$ vectors lie on a sphere of radius $V_{n,m}$ centered at the tail of $\boldsymbol{V}$. Recall that all $\boldsymbol{V}_\infty$ lie on a sphere of radius V_∞ centered on the head of the $\boldsymbol{V}$ vector. The actual vectors $\boldsymbol{V}_{n,m}$ and $\boldsymbol{V}_\infty$ must meet along the intersection of the two spheres, assuming they intersect for our choice of n,m. Figure 2 shows a velocity-space diagram of the relationships.

The angles β_b and β_a ($=\beta_{n,m}$) are given by

$$\cos\beta_{b,a} = (V_\infty^2 + V^2 - V_{b,a}^2)/2V_\infty V$$

The angle α in Fig. 2 is the maximum bending angle allowed by Eq. (2) for a swingby. To find where $\boldsymbol{V}_{\infty a}$ should be placed to achieve resonance one could rotate $\boldsymbol{V}_{\infty b}$ about $(\boldsymbol{V}_{\infty b} \times \boldsymbol{V})$ through angle $\epsilon = \beta_b - \beta_{n,m}$. This is possible provided $|\epsilon| \leq \alpha$, and will guarantee a return orbit. Now if $|\epsilon| < \alpha$, it will be

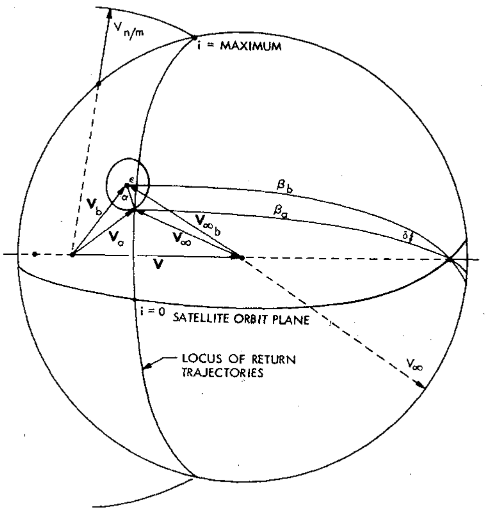

Fig. 2 Detailed cranking and pumping geometry.

possible to change the inclination and/or flight path angle by rotating $V_{\infty b}$ about V through angle δ such that

$$\cos\delta \leq (\cos\alpha - \cos\beta_b \cos\beta_a)\ /\sin\beta_b \sin\beta_a$$

These two rotations together allow the creation of any outgoing vector $V_{\infty a}$ which gives a return encounter with bending at the encounter of no more than α. In the case of pure cranking we must already be in resonance with the satellite, thus, $\epsilon = 0 < \alpha$, and the maximum crank will be

$$\delta = \arccos\left[(\cos\alpha - \cos^2\beta)\ /\sin^2\beta\right]$$

Now, in orbit cranking, $V_a = V_b$ and Eq. (1) simplifies greatly to

$$\cos\gamma_b \cos i_b = \cos\gamma_a \cos i_a$$

which is to say that the angle between the jovicentric velocity vector and the swingby satellite's velocity is conserved. The maximum inclination or flight path angle with the energy so constrained is simply

$$i_{\max} = \gamma_{\max} = \arccos(\cos\gamma_b \cos i_b)$$

$$= \arccos\left[(V^2 + V_a^2 - V_\infty^2)\ /2VV_a\right]$$

But at this inclination, the flight path angle must be zero, and vice versa. The zero flight path angle implies that the spacecraft is at apojove ($V_{n,m} < V$) or perijove ($V_{n,m} > V$). It is, thus, theoretically possible to increase the inclination of the orbit from zero to very large values (of order 30-60 deg for realistic orbits). The number of swingbys required to increase the inclination to $i_{\max}$ from 0 is just 90 deg/δ.

Figure 3 shows the relation between inclination, periapsis radius, and flight time for cranking at Callisto in a 1/1 resonant orbit. It can be seen that, for a flyby altitude of, e.g., 1300 km(0.5 R_{Callisto}) the inclination can be raised to 50 deg in 0.75 years and that the periapsis goes from 4.2 R_J to 20 R_J at the same time. The entire sequence requires about 20 encounters. Similarly, if the inclination were lowered, the periapsis would also be lowered by cranking.

Effects of Even $n\pi$ Transfers on Perijove

Because lowering the perijove altitude increases the risk of radiation damage, it is important to consider how pumping and cranking affect perijove. Equation (1) can be rewritten as follows

$$E_b - n'\sqrt{2(\mu r_{pb} + r_{pb}^2 E_b)}\cos i_b$$
$$= E_a - n'\sqrt{2(\mu r_{pa} + r_{pa}^2 E_a)}\cos i_a \qquad (4)$$

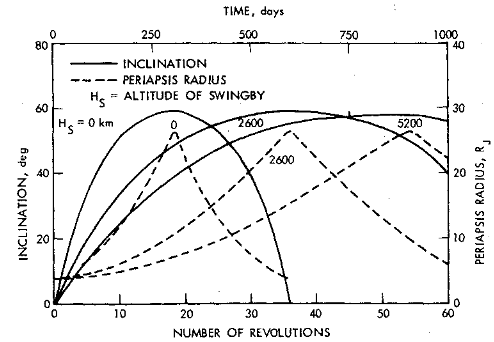

Fig. 3 One-to-one orbit cranking at Callisto.

where the angular momentum has been replaced by an expression for r_p. This equation shows that, unless E, r_p, and i are all constant, then, at least two of them must vary. If E is constant in an encounter, we are cranking and $E_b = E_a$. By taking an arbitrarily small variation in inclination, we have

$$\left(\frac{a - r_p}{2a - r_p}\right)\frac{\delta r_p}{r_p} = \tan i\,\delta i \qquad (5)$$

where a is the (constant) semimajor axis of the jovicentric orbit.

Equation (5) indicates that, for low inclinations, the tradeoff between inclination and periapsis radius is small. In a practical sense, this means that we can crank the orbit several degrees around zero without making an appreciable change in the periapsis radius. Because the coefficient of $\delta r_p / r_p$ in Eq. (5) is positive, the periapsis radius approaches a minimum as the inclination approaches zero.

Now if the inclination i is held constant, we may consider a small variation in E. The result may be written as

$$\left(\frac{2a}{r_p} - 1\right)\frac{\delta r_p}{r_p} = \left(\frac{V_p r}{V r_p \cos i} - 1\right)\frac{\delta E}{-E}$$

where r is the orbital radius of the swingby satellite and V_p is the spacecraft perijove speed. Since $V_p > V$, $r > r_p$ and $V_p^2 > -E > 0$, it can be seen that, as E becomes larger, r_p must also become larger, and vice versa. Since r/V is proportional to the satellite period and since the period is greater for the outer satellites, it can be seen that δr_p is larger for the same δ-E if an outer satellite can be encountered. This means that, if an inner and an outer satellite are used alternatively to pump the energy down and up, respectively, it is possible to have the net effect of pumping the energy down without lowering the periapsis radius.

Odd $n\pi$ Transfers and High Inclination Orbits

A restriction of the values of orbital angular momentum for Keplerian transfer orbits accompanies the natural restriction of transfer angles to odd multiples of π for high inclination transfers between different satellites of a circular coplanar system. If the transfer from one radius r_1 to another r_2 is an odd $n\pi$ arc, then it can be shown that

$$p = 2r_1 r_2 / (r_1 + r_2)$$

is the value of semilatus rectum which defines the orbital angular momentum $h = \sqrt{\mu p}$ required for the transfer.

Thus, we can classify a priori the discrete values of angular momentum that will permit the various high-inclination transfers from satellite to satellite. Because the Galilean satellites all have nearly circular and coplanar orbits, we can expect the angular momenta of actual transfers to lie close to these discrete values. In this way, it is possible to eliminate vast regions of transfer orbits from consideration simply because the angular momentum is too far from the ideal

Table 1 Values of p for odd $n\pi$ transfers

	Io	Europa	Ganymede	Callisto
Io	5.9	7.3	8.5	9.6
Europa	7.3	9.4	11.6	13.9
Ganymede	8.5	11.6	15.0	19.1
Callisto	9.6	13.9	19.1	26.4

values to allow a swingby to adjust the orbit to the required shape.

Table 1 gives the values of the semilatus rectum that permit odd $n\pi$ transfers between Galilean satellite pairs. The values of p are given in Jupiter radii R_J of 71372 km, and the satellites are idealized as circular and coplanar.

Sample High-Inclination Transfers

The ideas just discussed have been employed to identify some Ganymede-Europa transfers for an inclination near 30 deg. The preswingby orbit period is 37.4 days and the perijove radius is 6.0 R_J. The initial inclination is 27 deg with respect to the Jovian equator and the date at Ganymede is Aug. 24, 1984. After a 4300-km altitude outbound flyby at Ganymede, the spacecraft arrives at Europa on Sept. 28, 1984 for a flight time of 35 days. The transfer orbit perijove radius was 5.12 R_J, which gives a value of 11.396 R_J for the semilatus rectum of the transfer orbit, almost as predicted by the idealized concepts of the previous section. The flight time for the transfer indicates that the semimajor axis should be 45 R_J from the circular coplanar orbit analysis. The actual semimajor axis of the transfer orbit was 44.38 R_J.

Thus, we have found that the elusive high-inclination transfers exist in spite of the small eccentricities and inclinations of the Galilean satellites. But the immediate question arises whether or not a satellite tour can be continued. It was not possible to return directly to Ganymede from Europa, but it was quite easy to hop to the nearest resonance with Europa to ensure a second encounter. Temporizing in this way, we found it possible to wait until Ganymede moved into a position that would permit a high-inclination transfer back to the original satellite, this time by means of a 540-deg transfer from Europa inbound on Nov. 2 to Ganymede outbound on Dec. 9 with a flyby altitude at Europa greater than 6000 km.

The discovery of these transfers opens up a whole new regime of mission design possibilities, and while the high-inclination transfers are less abundant than equatorial trajectories, they allow us to consider the possibility of a combined satellite tour and planetology mission that has the entire Jovian environment for its objective.

Perturbations, Navigation, and Performance

This section is a condensation of material developed in the study[5] which was the basis for this paper. Included is a discussion of the types and magnitudes of perturbations that will affect the precise targeting patterns for an actual mission, some comments on the importance of the navigation problem and estimates of correction maneuver requirements, and some flight-mechanical performance considerations that serve as background for a later discussion of mission design concepts.

Perturbations

The gravity-assist concepts of the previous section have been developed in the context of the restricted three-body problem and, although the zero-sphere-of-influence patched conic analysis is correct in its prediction of the effects of a sequence of single encounters, there are other perturbations of the two-body orbit dynamics used for the targeting from one encounter to the next.

There are three principal disturbing forces that will affect the orbit during transfer between satellite encounters. They are 1) oblateness, 2) solar gravitational perturbations, and 3) third-body perturbations due to Galilean satellites not participating as a launch or target planet in a specific transfer.

In case the perijove radius of a satellite-to-satellite transfer is less than 2 or 3 R_J, the oblateness perturbations can cause changes in the orientation of the jovicentric ellipse of the order of a few tenths of a degree during the perijove passage These deviations can cause targeting errors at the post perijove encounter of order several thousand kilometers. These errors can grow to become several tens of thousands of kilometers in the case of a very low perijove radius ($R_p = 1.1$ R_J).

Solar gravitational perturbations can modify the jovicentric orbit for some of the large-apojove orbits under current consideration. The principal orbital variations come from medium-periodic eccentricity perturbations due to the motion of Jupiter around the sun. These variations were estimated, and shown to cause changes in the perijove radius of the order of a few tenths of Jupiter's radius. The resulting satellite-relative targeting errors can be expected to be several tens of thousands of kilometers, somewhat larger than the oblateness perturbations.

Resonance and standoff encounter perturbations due to Galiliean satellites also enter the precision targeting problem. The orbital changes caused by close satellite encounters are actually perturbations to the jovicentric orbit, but they are not small in the sense usually implied by the term. These changes are almost completely described by the zero-sphere-of-influence patched conic technique in the situation where the (close) swingby conditions are specified. But the gravitational perturbations due to the Galilean satellites not involved in the encounter can affect the orbit in the usual sense of the lunar and planetary theories.

The results of some previous work[6] on low-order orbital resonances with the Galilean satellites were used to obtain an estimate of the resonance perturbations. The principal variations were found in the semimajor axis. These variations had a period of about 400 days and an amplitude of 0.35 R_J ($\approx$25,000 km). Thus, for these low-order resonance situations, we can expect total variations in semimajor axis of order 50,000 km in about 200 days ($\approx$25 revolutions) or about 2000 km per revolution on the average. Variations in the other elements are smaller but combine with the variations in period to cause targeting errors of the order of 10,000-50,000 km. Thus, we find on the low end of the orbital spectrum, perturbations of the same order of magnitude as those due to the solar gravity on the very large orbits.

There is one more aspect of the targeting problem that should be dealt with before the full flexibility of gravity-assist trajectories can be realized. This aspect is the prediction and resolution of the effect of standoff or moderately close (100,000-500,000 km) encounters with Galilean satellites during the transfer to a close encounter with another satellite. It is often difficult to achieve a specific transfer to one satellite without passing fairly close to another. Our current solution to this problem is to avoid standoff encounters that are closer than about 100,000 km and the computer software is designed to scan the targeted transfer trajectories for standoff encounters. If the specific transfer cannot be achieved without the intervening standoff encounter, it is nearly always possible to select the imposing satellite as a target and, after what will then be a close (control) encounter, the spacecraft can be targeted on to its original objective. In this way, we force the encounter sequences to be realistic and use the perturbations rather than attempt to compensate for them with valuable maneuver fuel.

Navigation

Before the gravity assists can be built into the mission design, their navigation feasibility must be demonstrated. Demonstration consists of proving that the satellite targeting can be carried out accurately enough to permit close satellite flybys without danger of impact and without excessive propellant requirements (ΔV) for navigation maneuvers.

What levels of accuracy might we expect? Radio-alone navigation accuracy is dominated by ephemeris errors.[7] The Mariner-Jupiter-Saturn project is assuming 350 km a priori position errors, and a posteriori determination to 20 km.[8] While these latter levels cannot be maintained for several years it should be possible to assume a priori ephemerides for a 1981 Jupiter-orbiter mission of about 50-100 km in position. Thus radio-alone navigation will yield 1σ position errors of about 400 km before in-flight determination of the satellite ephemerides and about 50-200 km afterwards (depending on actual flyby geometries). A satellite tour mission is possible with these accuracies, although the number of flybys would be limited, or the passing distances (and hence the power) of the flybys would be highly constrained, or the navigation fuel requirements would be very high.

Mariner missions of the future will, however, also use on-board optical measurements of the target's inertial position to overcome ephemeris and geocentric distance limitations.[9] The basic science TV sensor is used to provide an angular measurement of the target satellite relative to a star background. The accuracy is about one pixel (1σ) or about 20 μrad in present systems as long as the satellite is a point target. When the satellite becomes extended the limiting accuracy is center finding–about 1% of the apparent diameter (in the future this limit might be as low as 0.2%[10]). From such accuracies it seems that we can count on position determinations of the spacecraft with respect to the target satellites of less than 50 km (1σ) and that very close flybys of less than 400 km will be permissible with the use of on-board optical measurements.

These values are rough and conservative in that they take account only of error sources and not of data filtering. However, they are also optimistic in that they ignore potential degeneracies, singularities, geometrical restrictions, and data surprises. Orbit determination simulations have begun,[7] but much more work in this area is needed.

The concern is not just with targeting, but with spacecraft mass implications. How much navigation fuel will be needed to correct for the errors and what are the effects of these requirements on the spacecraft? The maneuver strategy for the satellite tour is far more complex than the usual orbit trim analysis or linear guidance theory required for previous missions. We need to deduce optimal times and numbers of maneuvers simultaneously to minimize fuel expenditures (over several satellite encounters), control miss distances to certain tolerances, and insure future targeting objectives.

The solution to this problem will require a complex analysis and years of simulations. Work has begun[11] which has shown the feasibility of a three-maneuver strategy to insure a pumping or cranking sequence and to keep fuel errors in realistic bounds. These studies lead us to believe that the 99% ΔV requirement can be kept to only a few meters/second per orbit and that over a whole mission involving satellite touring, orbit pumping and cranking, the navigation ΔV load might be of the order of 300 m/s. This presupposes optical navigation accuracies of about 50 km or less with respect to the satellite.

Another opportunity exists–this arises from the recognition that it is hard to avoid the satellites.[12] A multinominal navigation strategy can be devised, permitting the mission designer to have several options to meet his objectives. Specifically, he could retarget to minimize ΔV, knowing full well that within a few encounters he can recapture the target strategy. It is suggested that adaptive mission design is possible and that new nominals can be defined over the period of several orbits–ones that will take advantage of new science results and ones that can reduce ΔV requirements. The uncorrected errors (e.g., in velocity) could merely be absorbed into the definition of the following sequence of encounters and desired orbits. The situation, depicting multinominal and nominal navigation, is shown in Fig. 4.

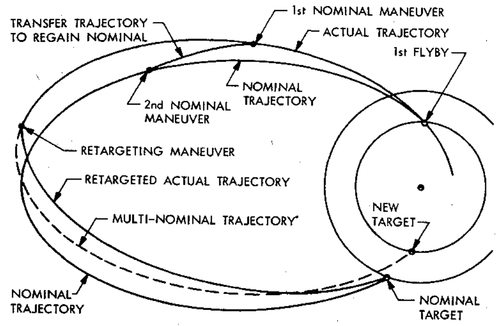

Fig. 4 Jupiter orbiter navigation concepts.

At present, given the above discussion, we conclude that: 1) Navigation accuracies at the satellites will be of order 400 km at first, and 50 to 200 km later for the radio-alone case and 30-50 km, assuming a planned Mariner optical measurement system. 2) Strict nominal path navigation analysis yields a minimum of two but probably three impulses per orbit required for navigation with mean ΔV_{TOTAL} per orbit of 5-10 m/s, assuming flybys in the range of 500-2000 km and on-board optical system targeting accuracies. 3) The total ΔV requirements should be less than 300 m/s (99%) for the mission, but this is of concern in that large ΔVs may result for close flyby, and hence the real limitations of an arbitrary ΔV are not yet known. 4) Multinominal navigation opens new vistas in mission design and probable reduction of ΔV_{TOTAL} requirements. We suspect that the tip of an interesting "iceberg" has been uncovered and that further research and application will be of great value.

Performance Considerations

As background for the discussion of the main concepts of this paper, some discussion of the performance requirements expected in Jupiter-orbiter missions will be presented here. The quoted propulsive capability estimates are indicative of the range of values under consideration at the time of publication and should not be thought to represent a specific statement of capability. Preliminary mission and spacecraft definition studies have demonstrated the existence of an exciting Mariner-class Jupiter-orbiter mission set in the 1981-1982 opportunity requiring launch vehicles and propulsion systems similar to those existing now.

Launch and Cruise Parameters

Reasonable launch opportunities occur in 1980, 1981-1982, and 1983, but the 1981-1982 launch is the most attractive in the next decade. Arrival dates range from July 1984 to Dec. 1984 for Type 1 (<180 deg transfer) trajectories and from July 1984 to April 1985 for Type 2 (>180 deg transfers) trajectories.

The 1981-1982 launch opportunity provides a 20-day launch window for $C_3 \leq 80$ km^2/sec^2 and allows arrival at Jupiter with approach excess velocities under 6 km/sec throughout the window. Earth departure trajectories required for the interplanetary transfers are near the ecliptic and permit a 90-deg azimuth (eastward) launch from ETR twice a day throughout the opportunity.

The interplanetary transfer requires about two and a half years, during which time the spacecraft leaves Earth at perihelion of its orbit, passes fairly close to Mars, moves on through the asteroid belt, and arrives at Jupiter with near-minimum jovicentric energy at aphelion on the heliocentric transfer. Thus, the trajectory takes the spacecraft to Jupiter's orbit where, because of its lower heliocentric speed, the spacecraft waits for Jupiter to come along in its orbit and the approach is along the direction of the dawn terminator.

Figure 5 is a presentation of orbit insertion performance characteristics showing the effects of the important parameters. Notice that, for a given ΔV, the lower the periap-

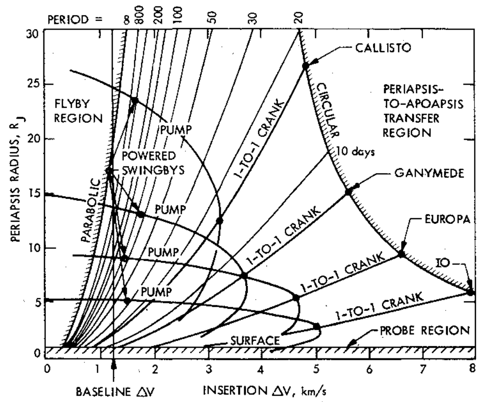

Fig. 5 Jupiter orbit insertion performance.

sis radius, the lower the postmaneuver orbit period. The vertical line indicates the 1290 m/s ΔV of an early baseline mission and shows the range of orbital period achievable by means of a perijove retro-maneuver at various radii.

Also shown on the chart is the performance gain available through the technique of applying the orbit insertion burn in the satellite's gravity field, i.e., with use of powered satellite swingby orbit insertion maneuvers. The points marked "powered swingbys" indicate the orbits achievable with the same ΔV (1290 m/sec) but applied at periapsis of a close encounter with one of the Galilean satellites. Note that the powered Ganymede swingby shown at 13 R_J and 1740 m/s yields a savings of 450 m/s over direct insertion into the same orbit at perijove. The powered swingbys were all constrained to pass no closer than 1000 km above the satellite surfaces. Thus, for the same ΔV, (1290 m/s), it is possible to establish an 80-day orbit instead of the 400-day orbit obtainable with a direct insertion at 13 R_J. Further performance improvement can then be obtained by pumping the orbit down. Reference 13 contains a detailed analysis of the optimal powered swingby orbit insertion.

The curves labeled "pump," in Fig. 5, indicate the loci of repeated energy-changing encounters with each of the Galilean satellites. By returning again and again to the same satellite, it is possible to change the orbit period by amounts that would require in excess of 5 km/s if the maneuvers were performed propulsively. The curves labeled "1 to 1 crank" indicate what is possible by use of repeated orbit cranking at each of the Galilean satellites in the situation where the spacecraft is 1-to-1 resonant with the swingby satellite. For example, it is possible to start with a highly elliptic, equatorial orbit whose period is the same as that of Callisto (~16 days) and, after 20 encounters with the satellite, end up with a circular orbit at Callisto's orbit distance but inclined nearly 60 deg to the Jovian equator.

Furthermore, Fig. 5 does not tell the whole story. There is no restriction to return each revolution to the same satellite; it is equally feasible to transfer from one satellite to another exactly as we fly from one planet to the other in the solar system. The principal distinction is in the time scale of the transfers. The practical limits of orbit control through use of the Galilean satellites have not been established except in the gross sense of certain limits achievable with repeated encounters with a single satellite.

For orbiter missions, there is some concern about the selection of the initial perijove radius because of the desire for an extensive, many revolution mission, and it is important to estimate the minimum perijove radius that can be maintained for several years in the radiation environment. Current estimates indicate that orbit insertion can be performed in the 3-10 R_J region. Even in the unlikely case that orbit insertion will have to be higher, the Ganymede-powered swingby insertion provides a means for establishing an orbit well above the radiation. Additional performance gains are realized by insertion at a lower perijove and then, perhaps, raising the perijove by gravity assist. The options available to mission planners are sufficiently numerous that no serious radiation problems are anticipated.

The previous paragraphs contain a brief background on performance aspects for a Jupiter orbiter; the specifics are not firm but were presented to give the reader an understanding of the real-world context in which the concepts of this paper have been developed. In the next section we consider the more general questions of mission design.

Mission Design Concepts

The preceding discussions of gravity-assist concepts, perturbations, navigation, and performance considerations indicate the possibility of unprecedented orbit control through the use of multiple encounters with the Galilean satellites. The flexibility provided by this control opens up such an abundance of mission design options that we are hard-pressed, in the space available, to present discussions of even a few of the possibilities. The resolution of these options into a single or double mission will require a concerted dialog among the scientific, spacecraft design, and trajectory analysis elements of the spaceflight community. In what follows, we attempt to communicate expanded concepts of mission design rather than to enumerate all the specific options that have come to light during the preliminary mission analysis.

Scientific Objectives

The Jovian environment presents so many objectives of scientific study that the design of the flight profile will be strongly affected by the desire to obtain various types of information. The process of flight profile selection is a complex and dynamic one involving several iterations among the space exploration disciplines to determine what is possible, what is desirable, and what is both. As specific missions are proposed and studied for feasibility and cost, the iteration process begins automatically and, by the time preliminary mission analyses are published, the flight profile, payload, and science objective options are usually fairly well fixed in the sense of a baseline mission profile.

In the case of Jupiter-orbiter, these options are considerably more numerous than those for previous missions. We find specific interest in satellite surface and Jupiter atmospheric remote sensing, particles and fields measurements, radio occultation and radio emissions experiments, and celestial mechanics experiments in the Jovian system. For each of these measurement types there is a desire to study the variability of these phenomena in space and time and exploit the advantages of an orbiter. References 14-16 contain discussions of current thinking on scientific objectives for Jupiter orbit missions.

Orbit Selection and Science Return

The principal concern of mission design is the selection of science payload, mission objectives, and spacecraft performance capability in such a way as to achieve the highest science return for a given set of conditions including cost, state-of-the-art capability, and projection of that capability into the future. In Jupiter-orbiter mission design, we see this concern as one whose resolution is much more complex than for past missions; the question is not what orbit shall we select but, rather, what sequence of orbits shall we decide on and which areas of the Jovian environment shall we visit? Shall we attempt to do everything with one spacecraft, or shall we design for two or more relatively uncontrolled missions that could provide redundancy and simplicity of design at some justifiable cost increase for the additional launch vehicle(s). The questions of orbit selection and satellite encounter sequences interact continuously among many potential scientific objectives.

Table 2 Sample satellite tour

Orbit no.	R_P (R_J)	Period (days)	Date	Satellite	Inbound (I) or outbound (0)	Flyby altitude (km)	Latitude (deg)	Smear velocity (km/s)
Insertion	14.0		07-19-84	Ganymede	I	1000	0	6.8
1	14.0	104	11-08-84	Ganymede	I	1302	9	5.6
2	13.4	43	12-19-84	Callisto	I	111,000	−83	6.4
			12-21-84	Ganymede	I	1268	−78	5.6
3	13.2	38	01-31-85	Callisto	0	5996	0	6.4
.								
.								
.								
20	11.8	15	12-20-85	Ganymede	0	10575	0	5.0
			12-30-85	Callisto	I	442000	0	5.9
21	11.4	13	01-01-86	Ganymede	I	1155	0	5.5
.								
.								
33	8.4	12	07-08-86	Europa	I	1114	−1	5.8
34	8.5	14	07-23-86	Europa	I	1389	−86	5.8
35	8.5	14	08-06-86	Europa	I	1388	86	5.8
.								
.								
.								
40	9.8	20	10-27-86	Ganymede	I	1750	−26	7.3
41	10.5	29	11-25-86	Ganymede	I	850	0	7.4
					(Bow shock excursion ~ 12-25-85)			
42	11.3	57						

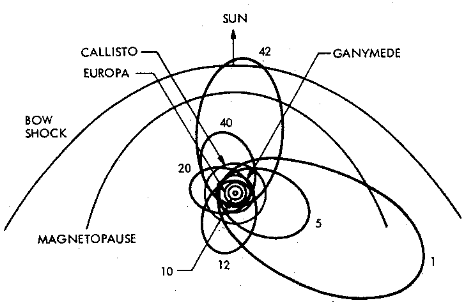

Fig. 6 Sample satellite tour profile (after Ref. 14).

Thus, in addition to the obvious performance tradeoffs with science payload, which we discuss further in the following, there are continual choices for the mission designer to make relative to encounter and orbit shape selection.

The discovery of resonance-hopping encounters that rapidly reduce the orbit period led to reconsideration of earlier orbit insertion strategies. It became apparent that a substantial performance gain could be achieved if the initial orbit were selected to be as large as is considered operationally feasible. The remaining energy change required to achieve the lower period orbits can then be obtained from gravity-assist encounters with the Galilean satellites.

The large initial orbit strategy was reinforced by science objectives to study the magnetopause and bow-shock region in the 150-200 R_J range and by the possibility of imaging experiments for some of the outer Jovian satellites (particularly Demeter, JX, and Hera, JVII). Further incentive came from the discovery of the performance gain achievable through the Ganymede-powered swingby orbit insertion. Recent studies[13] indicate a savings of at least 450 m/s in orbit insertion requirements over direct ballistic perijove insertion into the same orbit. But the powered swingby gives more than just performance; it supplies the performance at a high altitude (13 R_J for Ganymede) and provides us with the opportunity to pump the orbit down rapidly without descending into the radiation belts early in the mission. The option is still available to lower the perijove down to the orbit of Io (JI) or even Amalthea (JV) but the descent can be controlled, at the discretion of the mission director, in such a way as to allow a thorough, in-flight evaluation of the radiation at various altitudes before low-perijove orbits are selected.

There is still another aspect of gravity-assisted orbit control that affects the potential for scientific exploration; the line of apsides can be turned in either direction by means of repeated close encounters. A good rule of thumb is that the orbit will be turned in the prograde direction on inbound encounters that lower the perijove or outbound encounters that raise the perijove (and vice versa for apsidal rotation in the retrograde sense). This ability to reorient the line of apsides provides a great deal of flexibility with respect to the areas of the Jovian environment that can be visited in a two-to-three year time span, and in particular, permits the rapid examination of the particles and fields environment in all directions relative to the sun line.

This discussion is not intended to represent analyses or evaluations of science return; those kinds of judgments must come from a joint effort involving all the spaceflight disciplines. It indicates our suspicion that the full potential for scientific exploration of the Jovian system will exceed our earlier expectations and that further analysis of our goals and capabilities in this regard will be worthwhile.

Sample Satellite Tour

As a demonstration of the power of orbit control and the flexibility of encounter sequence selection, a sample encounter sequence has been designed that 1) provides for close (1000-2000 km) encounters with each of the three outer Galilean satellites, including passes above and below each satellite for study of the polar regions; 2) begins with perijove at 14 R_J and provides for slow lowering to a sequence minimum of 8.4 R_J as the orbit period is decreased from 105 to 14 days, and 3) provides for inertial rotation of the line of apsides so as to move the apojove through the antisolar direction and on around to permit a bow shock excursion two and one half years after orbit insertion.

Table 2 gives the encounter sequence and some parameters of interest for a short portion of the tour. It gives orbit number, perijove radius R_P, orbit period, date, satellite encountered, an inbound or outbound encounter code, satellite flyby altitude, latitude of flyby periapsis (satellite centered), and smear velocity. There are 28 close (control) encounters with Ganymede, 9 with Callisto, and 5 with Europa. The tour does not go to Io, illustrating the flexibility available by showing that the orbit can be turned 180 deg in an inertial frame without requiring the orbit to be pumped down into the severe radiation. The minimum perijove radius achieved during the tour is 8.4 R_J, which would allow imaging of Io from a distance of 186,000 km. The complete sequence is given in Ref. 5.

Figure 6 is a diagram of the tour, showing selected orbits from a 42-encounter sequence. The concept provides enough flexibility that more excursions out to the magnetopause can easily be included. Related work[16] has led to what has been called a "flower" orbit with four large orbits which sample the magnetopause 90 deg apart around a full circle from the sun-line.

The tour is only one of perhaps a hundred that would accomplish the same objectives. The tour could have been designed to provide control encounters with Io, or it could have included a several-month cranking sequence to increase the inclination up to 30 or 40 deg for high-latitude imaging of Jupiter's clouds. The orbit could have been shrunk to 14×26 R_J (Hohman transfer between Ganymede and Callisto) providing, if the science community were so disposed, a relatively inexpensive Ganymede orbiter for an additional 1 km/s retromaneuver at Ganymede. The possibilities are practically limitless.

Conclusions

A number of ideas for the practical application of gravity-assist techniques to Jupiter orbiter missions has been presented. While there is nothing conceptually surprising about the techniques, the strength and the flexibility they provide are quite remarkable. The techniques have been presented in the context of a preliminary mission analysis for a Jupiter-orbiter which, we are now confident, is a viable and exciting planetary mission option.

The discussions of gravity-assist techniques were accompanied by a discussion of dynamic perturbations that will affect the precise patterns of encounter sequence selection. Performance considerations for a representative mission profile were presented, along with a discussion of the application of the gravity-assist concepts to real-world mission design. Flexibility in orbit selection and potential for scientific exploration were cited as significantly more pronounced than has been the case in previous orbit missions, and an example encounter sequence was presented.

Given the control available through multiple-gravity assist, however, we must be concerned with the difficulty and cost of realizing that control. Our principal concerns in this regard are the navigation capabilities required to establish and continue extended satellite tour encounter sequences. The navigation problem was discussed and some preliminary estimates of navigation requirements were presented with reference to studies now in progress on this complex problem.

We feel that the assumptions and ground rules under which the concepts and techniques were developed have been quite conservative and that a great deal more flexibility and control is available. We submit that further analysis in gravity-assisted orbit insertion and orbit control, particularly when taken together with a concerted effort for science sequence planning, will be most fruitful.

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