GLOBAL MOON COVERAGE VIA HYPERBOLIC FLYBYS

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Abstract: The scientific desire for global coverage of moons such as Jupiter's Galilean moons or Saturn’s Titan has invariably led to the design of orbiter missions. These orbiter missions require a large amount of propellant needed to insert into orbit around such small bodies, and for a given launch vehicle, the additional propellant mass takes away from mass that could otherwise be used for scientific instrumentation on a multiple flyby-only mission. This paper will present methods—expanding upon techniques developed for the design of the Cassini prime and extended missions—to obtain near global moon coverage through multiple flybys. Furthermore we will show with proper instrument suite selection, a flyby-only mission can provide science return similar (and in some cases greater) to that of an orbiter mission.

Keywords: gravity assist, trajectory design, mission design, and orbital mechanics.

1. Introduction

Traditionally the challenge of global coverage of moons such as Jupiter's Galilean moons or Saturn's Titan has led to the design of orbiter missions. Such orbiter missions are very expensive in propellant as a result of the large maneuver required insert into orbit around such small bodies, and can be plagued by orbit stability complexities. Furthermore, for a given launch vehicle, the large amount of propellant associated with orbit insertion takes away from mass that could otherwise be used for scientific instrumentation, and in the case of a Europa mission, shielding—in the form of tantalum or other dense materials—to protect onboard electronics and instrumentation.

As part of modifying the Cassini Prime Mission and designing the Cassini extended missions (referred to as the Equinox and Solstice Missions), a new method for the placement of the groundtrack of a hyperbolic orbit over desired latitudes and longitudes was developed [1]. This method has been expanded such that a number of flybys can be used to systematically cover a specific hemisphere of a moon.

2. Gravity Assist Trajectory Design

The enabling mechanism for complicated missions such as Galileo [2,3] and Cassini [4-6] is a concept understood for over a century and employed in a number of missions during the past forty years—the gravity assist. A gravity assist entails a spacecraft using a massive moving celestial body to significantly modify its trajectory. Depending on the flyby speed and distance, and how the spacecraft flies by the large gravitating body (above/below, behind/in-front), the spacecraft’s orbit size (period, energy, and distance relative to the central body) and orientation
(inclination and line of apsides relative to the central body to Sun line) can be altered in an incremental and predictable manner such that a wide range of geometries can be attained to meet myriad, often disparate, scientific goals.

2.1 Pump and Crank Angles

By the patched-conic assumption [7-9], the spacecraft's velocity relative the central body ($v_{sc}$) is given by the vector sum of the gravity assist body velocity ($v_{ga}$) and the spacecraft's v-infinity ($v_\infty$) with respect to the gravity assist body. This sum is shown in Fig. 1.

![Figure 1: Pump angle.](image)

The angle depicted in Fig. 1 is referred to as the pump angle ($\alpha$). Applying the law of cosines, we may write the following equation for the pump angle:

$$v_{sc}^2 = v_\infty^2 + v_{ga}^2 + 2v_\infty v_{ga} \cos(\alpha)$$

(1)

We may also use the vis-viva equation to transform Eq. 1 into a relation for the spacecraft’s semi-major axis ($a_{sc}$) of its orbit relative to the central body. Here, $r_{enc}$ is the radius of encounter, i.e., the distance from the central body to the gravity assist body at the time of the gravity assist.

$$\frac{r_{enc}}{a_{sc}} = 2 - \sqrt{\frac{\mu_{cb}}{r_{enc}} \left[ v_\infty^2 + v_{ga}^2 + 2v_\infty v_{ga} \cos(\alpha) \right]}$$

(2)

This then also yields a relation for orbit period ($T_{sc}$) as a function of pump angle:

$$T_{sc} = 2\pi \sqrt{\frac{r_{enc}^3}{\mu_{cb}} \left( 2 - \sqrt{\frac{\mu_{cb}}{r_{enc}} \left[ v_\infty^2 + v_{ga}^2 + 2v_\infty v_{ga} \cos(\alpha) \right]} \right)^{-3}}$$

(3)

The velocity triangle in Fig. 1 can also be rotated out of the orbit plane of the gravity assist body when the spacecraft’s orbit is inclined relative to the orbit of the gravity assist body. Figure 2 shows this, where the crank angle ($\kappa$) is used to describe the rotation of the velocity triangle out-of-plane. The basis in Figure 2 is defined by [10]:

$$\hat{q}_2 = \frac{v_{ga}}{|v_{ga}|}$$

(4)
\[ \hat{q}_3 = \frac{\vec{r}_{enc} \times \vec{v}_{ga}}{|\vec{r}_{enc}| |\vec{v}_{ga}| \cos(\gamma_{ga})} \]  

(5)

\[ \hat{q}_1 = \hat{q}_2 \times \hat{q}_3 \]  

(6)

where \( \gamma_{ga} \) is the flight path angle of the gravity assist body at the time of the encounter (this is zero for gravity assist bodies in circular orbits).

![Figure 2: Crank angle.](image-url)

The \( v_\infty \)-infinity vector in this reference frame is then given by:

\[ \vec{v}_\infty = v_\infty \sin(\alpha) \cos(\kappa) \hat{q}_1 + v_\infty \cos(\alpha) \hat{q}_2 - v_\infty \sin(\alpha) \sin(\kappa) \hat{q}_3 \]  

(7)

Let’s introduce two more vector bases: a p-frame tied to the gravity assist body’s orbit plane and an s-frame tied to the spacecraft’s orbit plane:

\[ \hat{p}_3 = \frac{\vec{r}_{enc} \times \vec{v}_{ga}}{|\vec{r}_{enc}| |\vec{v}_{ga}| \cos(\gamma_{ga})} \]  

(8)

\[ \hat{p}_2 = \hat{p}_3 \times \hat{p}_1 \]  

(9)

\[ \hat{p}_1 = \frac{\vec{r}_{enc}}{|\vec{r}_{enc}|} \]  

(10)

\[ \hat{s}_1 = \frac{\vec{r}_{enc}}{|\vec{r}_{enc}|} \]  

(11)

\[ \hat{s}_3 = \frac{\vec{r}_{enc} \times \vec{v}_{sc}}{|\vec{r}_{enc}| |\vec{v}_{sc}| \cos(\gamma_{sc})} \]  

(12)

\[ \hat{s}_2 = \hat{s}_3 \times \hat{s}_1 \]  

(13)
The gravity assist body’s velocity vector and the spacecraft’s velocity vector are then given by:

\[ \vec{v}_{ga} = v_{ga} \hat{q}_2 = v_{ga} \sin(\gamma_{ga}) \hat{p}_1 + v_{ga} \cos(\gamma_{ga}) \hat{p}_2 \]  

(14)

\[ \vec{v}_{sc} = v_{sc} \sin(\gamma_{sc}) \hat{s}_1 + v_{sc} \cos(\gamma_{sc}) \hat{s}_2 = v_{sc} \sin(\gamma_{sc}) \hat{p}_1 + v_{sc} \cos(\gamma_{sc}) \cos(i_{sc}) \hat{p}_2 + v_{sc} \cos(\gamma_{sc}) \sin(i_{sc}) \hat{p}_3 \]  

(15)

We may use these two equations to write the v-infinity in the p-frame:

\[ \vec{v}_\infty = \vec{v}_{sc} - \vec{v}_{ga} \]

\[ = [v_{sc} \sin(\gamma_{sc}) - v_{ga} \sin(\gamma_{ga})] \hat{p}_1 + [v_{sc} \cos(\gamma_{sc}) \cos(i_{sc}) - v_{ga} \cos(\gamma_{ga})] \hat{p}_2 + v_{sc} \cos(\gamma_{sc}) \sin(i_{sc}) \hat{p}_3 \]  

(16)

We can also transform Eq. 7 into the p-frame:

\[ \vec{v}_\infty = v_{\infty} [\sin(\alpha) \cos(\kappa) \cos(\gamma_{ga}) + \cos(\alpha) \sin(\gamma_{ga})] \hat{p}_1 + v_{\infty} [\cos(\alpha) \cos(\gamma_{ga}) - \sin(\alpha) \cos(\kappa) \sin(\gamma_{ga})] \hat{p}_2 - v_{\infty} \sin(\alpha) \sin(\kappa) \hat{p}_3 \]  

(17)

Comparing Eqs. 16 and 17, the radial (p1) components give a relation for the crank angle:

\[ \cos(\kappa) = \frac{v_{sc} \sin(\gamma_{sc}) - v_{ga} \sin(\gamma_{ga}) - v_{\infty} \cos(\alpha) \sin(\gamma_{ga})}{v_{\infty} \sin(\alpha) \cos(\gamma_{ga})} \]  

(18)

When the gravity assist body is in a circular orbit, this reduces to:

\[ \cos(\kappa) = \frac{v_{sc} \sin(\gamma_{sc})}{v_{\infty} \sin(\alpha)} \]  

(19)

Quadrant ambiguities in Eqs. 18 and 19 can be resolved by looking at the out-of-plane (p3) components of Eqs. 16 and 17, which yields:

\[ \text{sign}(\kappa) = \text{sign}(i_{sc} \cos(\gamma_{sc})) \]  

(20)

\[ \text{2.2 Maximum Inclination} \]

The p3 component in Eqs. 18 and 19 yields a relation for the spacecraft’s inclination with respect to the gravity assist body’s orbit:

\[ \sin(i_{sc}) = -\frac{v_{\infty} \sin(\alpha) \sin(\kappa)}{v_{sc} \cos(\gamma_{sc})} \]  

(21)

Note that in the equation above, inclination can be negative. This corresponds to cases when the flyby is at the descending node of the spacecraft’s orbit. Although, inclination is traditionally a
strictly positive quantity, it is useful to allow it to go negative when talking about gravity assists, as otherwise we’d have to derive separate relations for flybys at ascending and descending nodes.

The \( p_2 \) component in Eqs. 18 and 19 yields a relation for the spacecraft flight path angle (\( \gamma_{sc} \)):

\[
\cos(\gamma_{sc}) = \frac{1}{v_{sc} \cos(i_{sc})} [v_{ga} \cos(\gamma_{ga}) + v_{\infty} \cos(\alpha) \cos(\gamma_{ga}) - v_{\infty} \sin(\alpha) \cos(\kappa) \sin(\gamma_{ga})]
\]

(22)

This can be combined with Eq. 21 to yield the following relation for spacecraft inclination:

\[
\tan(i_{sc}) = \frac{v_{\infty} \sin(\alpha) \sin(\kappa)}{v_{ga} + v_{\infty} \cos(\alpha)} [v_{\infty} \sin(\alpha) \cos(\kappa) \sin(\gamma_{ga}) - v_{ga} \cos(\gamma_{ga}) - v_{\infty} \cos(\alpha) \cos(\gamma_{ga})]
\]

(23)

When \( \gamma_{ga} = 0 \), this reduces to:

\[
\tan(i_{sc}) = -\frac{v_{\infty} \sin(\alpha) \sin(\kappa)}{v_{ga} + v_{\infty} \cos(\alpha)}
\]

(24)

The maximum value of inclination happened when the crank is \( \pm \pi/2 \). Therefore the inclination is bounded by:

\[
|\tan(i_{sc})| \leq \frac{v_{\infty} \sin(\alpha)}{v_{ga} \cos(\gamma_{ga}) + v_{\infty} \cos(\alpha) \cos(\gamma_{ga})}
\]

(25)

The dependence on the pump angle in Eq. 25 means that each spacecraft orbit period will have an associated maximum inclination.

2.3. Flyby Groundtracks

If the gravity assist body is in a circular orbit, and tidally locked so the prime meridian always points towards the central body (true for almost all satellites including Europa), the incoming or outgoing v-infinity can be written in terms of latitude (\( \phi \)) and longitude (\( \lambda \)) in the p-frame:

\[
\vec{v}_\infty = v_{\infty} [\cos(\phi) \cos(\lambda) \hat{p}_1 + \cos(\phi) \sin(\lambda) \hat{p}_2 + \sin(\phi) \hat{p}_3]
\]

(26)

Equation 26 can be compared with Eq. 16 or Eq. 17 to give a latitude and longitude corresponding to a pump and crank angle or to a spacecraft inclination, semi-major axis, and flight-path angle. However, in doing so we must remember that the v-infinity vector on the inbound-asymptote is pointed towards the surface and we want to use the negative of the inbound v-infinity to find the point on the surface under the trajectory. If we do this we arrive at the following relations for the inbound latitude and longitude (\( \phi_1, \lambda_1 \)) and the outbound latitude and longitude (\( \phi_2, \lambda_2 \)) in terms of pump and crank before a flyby (\( \alpha_1, \kappa_1 \)) and after a flyby (\( \alpha_2, \kappa_2 \)).
The flyby’s ground track is then a great circle connecting the inbound sub-point \((\phi_1, \lambda_1)\) to the outbound sub-point \((\phi_2, \lambda_2)\). The flyby periapsis occurs at the midpoint between these two \((\phi_p, \lambda_p)\), this is given by:

\[
\phi_p = \phi_1 + \frac{1}{2}(\phi_2 - \phi_1)
\]

\[
\lambda_p = \lambda_1 + \frac{1}{2}(\lambda_2 - \lambda_1)
\]

Note that in the above, \((\phi_p, \lambda_p)\) may need to be adjusted by \(\pm 2\pi\) in order to get the regular quadrants for latitude and longitude.

### 2.4. Same-Body Transfers

Depending on the planetary system and a spacecraft’s relative velocity with respect to the bodies in that system, one or more bodies exist that can be utilized for gravity assists. At Saturn and Uranus, Titan and Triton, respectively, are most easily utilized to design gravity assist tours. At Jupiter, the four Galilean moons, Io, Europa, Ganymede, and Callisto are all massive enough to be utilized. Apart from transfers between different moons, three basic* types of same-body transfers exist [8,10-11]: resonant, non-resonant, and pi-transfers.

A resonant transfer has a time-of-flight that is an integer multiple of the gravity-assist body’s period and the flybys at either end of the transfer occur at the same place in the gravity-assist body’s orbit. Because of this, the encounters’ longitude occurs on a fixed line from the central body and the resonant transfer may be inclined. A resonant transfer is typically labeled as \(m:n\), where \(m\) is the number of gravity-assist body revs and \(n\) is the number of spacecraft revs.

A non-resonant transfer’s time-of-flight is not an integer multiple of the gravity-assist body’s orbit, and the flybys occur at different longitudes in the gravity-assist body’s orbit. Other than the special case of a pi-transfer, the flybys of a non-resonant transfer do not occur at the same longitude and, therefore, constrain the spacecraft’s orbit plane to be the same as the gravity-assists body’s orbit plane.

* With large maneuvers or large third-body perturbations, leveraging transfers or other techniques are possible.
A pi-transfer is a special case of a non-resonant transfer where the time-of-flight of the transfer is \( m \) plus one-half times the gravity-assist body's period. The flybys of a pi-transfer occur on a line passing through the central body; hence these transfers can be inclined. In fact, they typically must be inclined with a specific inclination determined by the \( v \)-infinity magnitude [10]. A pi-transfer changes the longitude of the encounter by \( 180^\circ \).

### 2.5. Crank-over-the-top Sequence

Beginning from an equatorial orbit (\( \kappa = 0 \) or \( \kappa = \pi \)), a “crank-over-the-top” (COT) sequence is defined as a set of resonant transfers (\( N \)) used to crank the spacecraft inclination up to a maximum inclination (\( i_{\text{max}} \)) for a given orbit period (\( \kappa = \pi/2 \)), and continue cranking in the same direction—where the inclination will now decrease—until the spacecraft’s orbit plane has returned to an equatorial orbit.

Figure 3 shows the geometry of one flyby of a COT sequence. Note that the incoming and outgoing \( v_\infty \) have the same pump angle, hence the flyby \( \Delta v \) is perpendicular to the moon velocity. Since the line of apsides of a flyby hyperbola is aligned with the flyby \( \Delta v \), also the flyby closest approach must lie in a plane perpendicular to the moon velocity, and passing by the moon center of mass. Therefore, when the moons are tidally locked, and the moon orbit has zero eccentricity, the closest approach of a COT flyby occurs at \( 0^\circ \) or \( 180^\circ \) longitudes.

![Diagram](image)

**Figure 3: Geometry of one flyby in a COT sequence.**

The flyby \( \Delta v \) can be written equivalently as,

\[
\Delta v = 2v_\infty \sin\left(\frac{\delta}{2}\right)
\]

(33)

If the same period is maintained, and the gravity assist is used to only crank, the flyby \( \Delta v \) can also be written as,

\[
\Delta v = 2v_\infty \sin(\alpha) \sin\left(\frac{\Delta \kappa}{2}\right)
\]

(34)

Using Eqs. 33 and 34, and by definition for a COT, \( \Delta \kappa = \pi/N \), the flyby bending angle can be expressed as function of the pump angle and the number of COT flybys:

\[
\sin\left(\frac{\delta}{2}\right) = \sin(\alpha) \sin\left(\frac{\pi}{2N}\right)
\]

(35)
Using the definition of the flyby bending angle:

\[ \sin\left(\frac{\delta}{2}\right) = \frac{\mu}{\mu + r_p v_\infty^2} \]  \hspace{1cm} (36)

Equation 36 can be rearranged such that,

\[ \sin\left(\frac{\pi}{2N}\right) = 1 + \frac{r_p v_\infty^2}{\mu} \sin(\alpha) \]  \hspace{1cm} (37)

Equations 3 and 37 are used combined to find the number of flybys of the altitude of a COT given the \( v_\infty \) and the resonance \((m:n)\) period. Equations 37 and 38 can also be solved to find the required \( v_\infty \) for a given number of minimum-altitude flybys and a given resonance.

A COT sequence beginning with an inbound flyby will cover the sub-planet facing hemisphere of a tidally locked moon. At \( t_{\text{max}} \), the flyby will occur at \( \gamma = 0^\circ \) (periapsis when \( T_{sc} < T_{ga} \), apoapsis when \( T_{sc} > T_{ga} \), \( e = 0 \) when \( T_{sc} = T_{ga} \) and all subsequent flybys will be outbound (referred to as an inbound-to-outbound COT, or I/O COT). Likewise, at O/I COT sequence beginning with outbound flybys will cover the anti-planet facing hemisphere of a tidally locked moon and will return to the equatorial plan with inbound flybys.

Figures 4 and 5 exhibit the characteristics of COT sequences, where the minimum altitude is 100 km. Namely:

- For a given spacecraft orbit period, the number of flybys increases/decreases as \( v_\infty \) increases/decreases (Fig. 4).
- For a given \( v_\infty \), the number of flybys increases/decreases as the spacecraft orbit period decreases/increases (Fig. 5).

Figure 6 shows the parametric curves \( P(v_\infty; N) \). For \( N=1 \), one single gravity assist flips the \( V_\infty \) from inbound to outbound (or vice versa) and is referred to as “\( v_\infty \)-flipping”. If the gravity assist body is a circular orbit, a sequence of \( v_\infty \)-flipping non-resonant transfers can be used to change the location of the flyby on the moon’s orbit. The resulting trajectory is periodic in the rotating frame defined by the moon and the planet. Figure 7 shows a sequence of 36 \( v_\infty \)-flipping at Europa.

Lastly, as previously mentioned, when the same period resonant transfers are used throughout a COT sequence, all closest approaches will lie very near the prime or 180° meridians (i.e., 90° away in longitude from gravity assist body’s velocity vector). However, alternating the period of resonant transfers during a COT sequence (i.e., cranking and pumping), the closest approach can be placed away from the prime or 180° meridians (see Section 3.2).
Figure 4: O/I COT sequences at Europa with constant period (14.2 days, 4:1 resonant transfer) and a $V_\infty$ of: (a) 4.127 km/s, (b) 3.949 km/s, (c) 3.802 km/s, and (d) 3.702 km/s. Black: 1,000<alt<10,000 km; Yellow: 400<alt<1000 km; Green: alt≤400 km; Red: Closest approach.

Figure 5: O/I COT sequences at Europa with constant $V_\infty$ =4.266 km/s and varying period: (a) 10.65 days (3:1 resonance), (b) 14.2 days (4:1 resonance) (c) 17.75 days (5:1 resonance), and (d) 21.25 days (6:1 resonance). Black: 1,000<alt<10,000 km; Yellow: 400<alt<1000 km; Green: alt≤400 km; Red: Closest approach.
Figure 6: Number of flybys required by a COT sequence, as function of the period and of the $v_\infty$. For $N=1$, the graph shows the $v_\infty$ flipping solutions.

Figure 7: 3:1 O/I non-resonant flyby $v_\infty$-flipping example. $v_\infty = 3.2$ km/s, $-10^\circ$ regression in true anomaly per flyby. Rotates line of apsides $\sim 360^\circ$ using 36 flybys over 380 days.
3. Application

COT sequences can be used to obtain near global coverage for different moons in the solar system. The following two examples show the usefulness of these developed techniques, the latter showing that a multiple-flyby mission architecture can carry out a comprehensive investigation of Europa, exhibiting a number of potential advantages over an orbiter mission, and, is the preferred path by the scientific community to explore Europa under current fiscal constraints [13,14].

3.1 Titan Example

While Titan is one of the most fascinating moons in our solar system, it is also very useful in designing gravity assist trajectories at Saturn. For Cassini, given the high velocities the spacecraft encounters the various moons of Saturn, Titan is the only Saturnian satellite massive enough to significantly alter the spacecraft’s trajectory. A single 1000 km altitude Titan flyby provides the spacecraft a gravity-assist $\Delta v$ in excess of 800 m/s. As a result, gravity-assist tours are built as a sequence of Titan-to-Titan transfers, each flyby tuned to optimize not only Titan science, but also the science of the other four science discipline working groups on Cassini: Icy Satellites, Saturn, Rings, and Magnetosphere and Plasma.

During the development of the Cassini Solstice Mission, the following trajectory was built to exhibit to the Radar Team the amount of Titan coverage possible via two COT sequences (Fig. 8). Both COT sequences used 16 1:1 resonant transfers ($T_{sc}=15.9$ days); the first COT is an O/I COT sequence covering the anti-Saturnian hemisphere of Titan and occurs at Cassini’s descending node. The second COT is an I/O COT sequence covering the sub-Saturnian hemisphere of Titan and occurs at Cassini’s ascending node. Both COT sequences occur at the same Saturn local solar time (i.e., same location in Titan’s orbit relative to the Saturn-Sun line), and since the Cassini radar uses microwaves to map Titan’s surface, lighting conditions of the flybys were of no concern. Figure 9 show the very three-dimensional nature and symmetric characteristics of COT sequences with relatively high v-infinities at a gravity assist body.
Figure 9: Two COT sequences using 1:1 resonant transfers (15.9 day period) with a $V_\infty = 5.8$ km/s. (a) View from Saturn N. pole (sun-fixed, towards top), (b) Oblique view (inertial). Orange: Spacecraft orbit; Red: Titan’s orbit; Black: Orbit of six other inner icy satellites.

3.2 Europa Example

As reinforced by the 2011 NRC Decadal Survey [15], Europa remains one of the most scientifically intriguing targets in planetary science due to its potential suitability for life. However, based on JEO cost estimates and current budgetary constraints, the Decadal Survey recommended—and later directed by NASA Headquarters—a more affordable pathway to Europa exploration be derived. In response, a flyby-only proof-of-concept trajectory (referred to as 11-F5) has been developed to investigate Europa. See references 13 and 14 for a detailed description of the 11-F5 trajectory, and more generally, the proposed multiple-Europa flyby mission concept.

3.2.1 Science Objectives

The conceived model payload for a flyby-only Europa spacecraft contains an Ice-Penetrating Radar (IPR), Topographical Imager (TI), Shortwave Infrared Spectrometer (SWIRS), Ion and Neutral Mass Spectrometer (INMS), and. This notional payload is not meant to be exclusive of other measurements and instruments that might be able to meet the scientific objectives in other ways†. Refer to the Europa 2012 Study Report [14] for the details mapping the specific instruments to their corresponding Europa investigation(s).

The following summarizes geometric constraints levied on the mission design in order to fulfill required scientific objectives for a compelling Europa multiple-flyby mission:

† NASA would ultimately select the payload through a formal Announcement of Opportunity (AO) process.
Ice Penetrating Radar (IPR)
- Closest approach (c/a) relative velocity: < 5 km/s
- c/a altitude: 100 km
- Coverage: Satisfy the following constraints in 11 of 14 panels (Fig. 10)
  - Three 800 km groundtracks in anti-Jovian panels, and two 800 km groundtrack segments in each sub-Jovian panel (altitude ≤ 400 km)
  - Each groundtrack must intersect another groundtrack (intersection may be outside the panel of interest) below 1,000 km (when altimetry mode begins)
  - Cover anti-Jupiter hemisphere first (preferred, not required)
- Requires simultaneous stereo imaging to provide topographic information necessary to process the IPR data

Topographic Imager (TI)
- c/a relative velocity: < 5 km/s
- c/a altitude: 100 km
- Solar phase: 50-70° (10-80° acceptable) when alt ≤ 400 km

Shortwave Infrared Spectrometer (SWIRS)
- c/a relative velocity: < 6 km/s
- c/a altitude: 100 km
- Local Solar Time: 9 am - 3 pm (the closer to noon the better)
- Solar phase angle: <45 degrees (preferred)
- Ability to target specific geologic features that are globally distributed (300 m/pixel, 11 of 14 panels)
≥ 70% coverage at ≤ 10 km per pixel

Ion and Neutral Mass Spectrometer (INMS)
- c/a relative velocity: < 7 km/s
- c/a altitude: 25 km (or more generally, as close as navigationally possible)

Figure 10. 14 panels defined by the Science Definition Team (SDT) used to assess global-regional coverage. Since Europa is tidally locked, the same hemispheres always face towards (sub-Jovian) or away from (anti-Jovian) Jupiter.
3.2.2 Multiple-Flyby Trajectory (11-F5)

The 11-F5 trajectory is a fully integrated trajectory from Earth launch (2021) through a notional end-of-mission (Ganymede impact). The trajectory design goal was to maximize IPR, TI, SWIRS and INMS coverage while minimizing TID‡, mission duration (and hence operations costs), and \(\Delta V\).

After a 6.37-year Venus-Earth-Earth gravity assist (VEEGA) interplanetary, five Ganymede flybys (include Ganymede-0 prior to JOI) would be used to lower the spacecraft’s orbital energy with respect to Jupiter and set up the correct flyby conditions (lighting and relative velocity) at Europa. First, since Europa is tidally locked (i.e., the prime meridian always faces towards Jupiter), the terrain illuminated by the Sun is simply a function of where Europa is in its orbit. By implementing a nonresonant G0–G1 transfer followed by three outbound resonant transfers, the spacecraft’s line of nodes can be rotated clockwise such that the first set of Europa flybys would occur very near the Sun–Jupiter line, and hence, Europa’s anti-Jovian hemisphere is sunlit. This is necessary since visible wavelength stereo imaging must be done in unison with IPR measurements as outlined in Section 3.2.1. Second, to meet the science coverage requirements, but also minimize the number of Europa flybys (and hence minimize TID), the first COT sequence (O/I COT-1) would use a combination of 4:1 \((T=14.3\ \text{days})\) and 7:2 \((T=12.4\ \text{days})\) resonant transfers with a \(V_\infty\) of approximately \(3.9\ \text{km/s}\). While alternating between the two resonances takes more time and leads to a higher TID (7:2 resonance has two perijove passages between Europa flybys) as opposed to using only 4:1 resonant transfers, it would result in the closest approaches being pulled away from the 180º meridian far enough to place a large portion of the groundtrack in the equatorial leading and trailing sectors of the anti-Jovian hemisphere (Fig. 6).

Once COT-1 is complete (which would change the Europa flybys from outbound to inbound), a nonresonant Europa transfer would be used to get back to an outbound flyby such that another O/I COT sequence could be implemented to cover the anti-Jovian hemisphere of Europa again. This nonresonant transfer would also change the local solar time (LST) of the Europa flybys.

All flybys in COT-1 occur at the ascending node. COT-2 (O/I, using strictly 4:1 resonant transfers) instead cranks in the opposite direction, placing the flybys at the descending node. This results in the COT-2 groundtracks intersecting the COT-1 sequence groundtracks (instead of running nearly parallel), hence fulfilling the IPR requirements in all seven anti-Jovian hemisphere sectors to have groundtracks with intersections (Fig. 7).

Before IPR, TI and SWIRS data could be collected on Europa’s sub-Jovian hemisphere, the observational lighting conditions need to be changed by 180º. That is, the location of the Europa flybys needs to be moved to the opposite side of Jupiter so that Europa’s sub-Jovian hemisphere would be sunlit. To do this, “switch-flip” was implemented [13]. A switch-flip involves first cranking up the inclination and pumping down the orbit period with Europa flybys to set up the correct geometry for a Europa-to-Ganymede pi-transfer. Once at Ganymede, a Ganymede pi-transfer is executed (3.5-day TOF), followed by a 1:1 resonant Ganymede transfer that cranks

\(\dagger\) Total ionizing dose Si behind a100-mil Al, spherical shell.

\(\S\) Variations in \(V_\infty\) occur due to Europa’s eccentricity and apsidal precession.
down the inclination and sets up the Ganymede-to-Europa pi-transfer. The result: All subsequent Europa flybys are located ~180° away from the last Europa flyby in COT-2 and the sub-Jovian hemisphere of Europa is sunlit.

Immediately following the Ganymede-to-Europa transfer, Europa flybys would be used to pump-up the orbit and crank-over-the-top. Like COT-1, the goal of I/O COT-3 is to minimize the number of flybys while still providing adequate coverage for science. However, since the $V_\infty$ is ~3.5 km/s (instead of 3.9 km/s in COT-1), the COT-3 sequence would need to instead alternate between 3:1 ($T=10.7$ days) and 5:2 ($T=8.8$ days) resonant transfers to accomplish this. Lastly the first four Europa flybys in COT-3 (Europa27 to Europa30), would be in Jupiter’s shadow; hence no stereo imaging can be performed in unison with IPR measurement (see Fig. 13). This is something that wasn’t noticed during the design of the 11-F5 trajectory and will be corrected in

Once COT-3 is complete, a nonresonant Europa transfer would be used to get back to an inbound flyby such that another I/O COT sequence can be implemented to cover Europa’s sub-Jovian hemisphere. This nonresonant transfer also changes the LST of the Europa flybys, as is shown in Figure 9.

Finally, COT-4 (I/O) cranks in the opposite direction from COT-3 (i.e., switches the node at the Europa flybys from descending to ascending) with 3:1 resonant transfers to intersect the COT-3 sequence groundtracks, fulfilling the IPR requirements in six of the seven sub-Jovian hemisphere sectors (Fig. 13).

At the conclusion of COT-4, 13 of the 14 sectors have been covered sufficiently to meet the observational and measurement requirements of all four instruments on board as defined by the SDT. Figures 14-16 show the cumulative topographic imaging coverage, petal plot, and

![Figure 11. Europa COT-1 (O/I) nadir groundtracks. Closest approach is marked with an “x” and numbered in accordance with Table 4. Red: 0<alt≤25 km; blue: 25<alt≤400 km; white: 400<alt≤1,000 km.](image)
Figure 12. Europa COT-1 (O/I) and COT-2 (O/I) nadir groundtracks. Europa nadir groundtrack plot with COT-1 and COT-2. Green check marks indicate IPR requirements are met in specified sector. Closest approach is marked with an “x” and numbered in accordance with Table 4. Red: 0<alt≤25 km; blue (COT-1) and cyan (COT-2): 25<alt<400 km; white: 400<alt<1000 km.

Figure 13. Europa COT-1–COT-4 nadir groundtracks. Europa nadir groundtrack plot for entire 11-F5 baseline trajectory. Green check marks indicate IPR requirements are met in specified sector. Red circles with “e” indicate flybys in eclipse. Closest approach is marked with an “x” and numbered in accordance with Table 4. Red: 0<alt≤25 km; blue (COT-1), cyan (COT-2), orange (switch-flip), magenta (COT-3), and green (COT-4): 25<alt<400 km; white: 400<alt<1000 km.
Figure 14. Nadir pointed cumulative topographic stereo imaging map for altitudes $\leq 4,000$ km and solar incidence angles between 0-90°.

Figure 15. 11-F5 Petal Plot. View from Jupiter’s north pole (Sun-fixed, towards top) of the 11-F5 baseline trajectory. Black: pump-down; blue: COT-1; cyan: COT-2; orange: switch-flip; magenta: COT-3; green COT-4; gray: orbits of the four Galilean satellites.
3.2.3 Multiple-Flyby Advantages

A variety of scientific investigations are required to address and answer key questions about Europa’s habitability. For a given launch vehicle such as the Atlas V 551 (or smaller), a multiple-flyby mission can exhibit many potential advantages over an orbiting spacecraft including:

- Given the finite capability of a chosen launch vehicle, more mass is available for scientific instrumentation and electronic component shielding (effectively increasing the lifetime of the mission) by forgoing EOI (i.e., the large amount of propellant needed to dissipate the spacecraft’s energy such that Europa orbit is reached)

- The ability to utilize a “store and forward” approach (i.e., collect, store, and eventually downlink data) enabling the use of higher power instruments in the vicinity of Europa since the spacecraft would never have to simultaneously operate the instruments and a high power telecom system.

- The large amount of time the spacecraft would spend away from Europa (in Jupiter orbit) would allow ample time to downlink the large amounts of data collected during each flyby without accumulating radiation dosage
• Data return is less susceptible to spacecraft or DSN anomalies due to much less compressed/stressed operations at Europa

• Since the spacecraft is not constrained to permanently residing in Europa’s gravity well, new Jupiter system campaigns could be executed once the spacecraft expected TID limits are reached (i.e., similar global-regional coverage campaigns at Ganymede and/or Callisto)

• The mission has many spacecraft disposal options [14], none of which include Europa impact

4. Conclusions

A trajectory design technique to obtain near global coverage of one or more moons in a system via multiple flybys has been derived. This method was used to design a complex network of 34 Europa flybys to efficiently investigate the habitability of Europa—previously thought infeasible—and has uncovered that a multiple-flyby mission architecture exhibits a number of potential advantages over an orbiter mission. The developed multiple-flyby Europa mission is now the preferred path by the scientific community to explore Europa in the near future given current fiscal constraints, and, the quality and the quantity of science return that is desired.

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References


