VENUS EXPLORATION USING PILOTED FLYBYS AND TELEPRESENCE

GRANT BONIN* AND TARIK KAYA
Department of Mechanical and Aerospace Engineering, Carleton University, 1125 Colonel By Drive, Ottawa, Ontario, Canada, K1S 5B6.

Email: gbonin@connect.carleton.ca*

This paper investigates near-term, low-cost human expeditions to Venus using gravity assisted trajectories and telepresence. In this mission design, a crewed spacecraft is injected onto a near minimum-energy transfer to Venus, and upon arrival is targeted to perform a polar flyby manoeuvre which transforms its outbound trajectory into a heliocentric orbit that maintains close proximity to the planet for an extended period of time. From this vantage point, astronauts can utilize real-time surface communications to remotely operate a series of robotic Venusian explorers. At the conclusion of the mission, a second polar flyby is used to re-inject the spacecraft onto its original trajectory, which returns the crewed vehicle to Earth exactly two years after launch. Employing such gravity-assisted trajectories, an entire round-trip piloted flight to Venus can be accomplished using a single low-energy injection burn leaving Earth, with only minute (<1500 m/s velocity change) adjustments required subsequently. The proposed mission design can be accomplished with either dedicated spacecraft and Evolved Expendable Launch Vehicle (EELV)-class boosters, or alternatively, with combinations of spacecraft and launch systems planned for NASA’s current exploration systems architecture. It is concluded that the mission design proposed herein represents a simple, safe, and cost-effective method of exploring one of the most compelling worlds in our solar system, while simultaneously allowing humankind to extend its reach further into space.

Keywords: Venus, human exploration, flybys, patched conics

1. INTRODUCTION: WHY VENUS?

Venus is at once the most similar and most different planet from Earth in the solar system. The Venusian surface environment is characterized by temperatures in excess of 730 K (hot enough to melt lead, and with hot spots approaching 1000 K), an average surface pressure of 96 bars, and sulphuric acid rain clouds; all of which have been precipitated by a stunning example of runaway planetary greenhousing. Yet in spite of its extremely hostile surface, above the thick Venusian cloud cover at an altitude of approximately 50 km, the most Earth-like environment in the solar system can be found, with an average temperature of approximately 310 K and an atmospheric pressure of 1 bar—essentially “shirt sleeve” weather for human beings, comparable to equatorial conditions on Earth [1].

However, Venus is still relatively mysterious compared to other worlds in our solar system. Earth’s sister planet still poses a number of unanswered questions in several key areas of planetary science, including:

- The elemental and mineralogical composition of its surface;
- Interactions at both the surface/atmosphere and atmosphere/space boundaries;
- Atmospheric composition (particularly isotope ratios of key species);
- Planetary volcanism, seismicity, and surface morphology; and
- Surface and upper atmosphere meteorology [2].

While much closer to the Sun than Earth, Venus absorbs far less solar energy; and contrary to present terrestrial models, Venus experiences hurricane-like storms which can circumnavigate the entire planet in as little as 4 days. Additionally, the planetary surface apparently lacks any major surface features older than 500 million years, suggesting that Venus may actually behave as a ‘volcanic pressure cooker’, periodically causing sudden and violent reshaping of surface features on a planetary scale.

Venus is indeed a world shrouded in mystery. The failure of extrapolated terrestrial models to accurately explain its characteristics highlight significant gaps in our current understanding of planetology. However, the corollary of this is that a better understanding of Venus may provide unique and important insights into how Earth works, with broad implications spanning subjects from climatology and weather forecasting to geology and planetary science.

Missions to Venus conducted by the United States and Russia have thus far been significantly limited in capability by the severity of the planet’s surface conditions. Survival rates for past surface probes such as Venera 4 have only been on the order of hours at best. However, increasingly technologies are emerging which may allow us to explore Earth’s sister planet for much greater durations, and with far greater accuracy and detail. The most recent expedition to Venus—the European Space Agency’s Venus Express probe, launched in November 2005—will be the first spacecraft to perform a global investigation of the Venusian atmosphere and its chain of surface-atmosphere-solar interactions. Unfortunately, this investigation will still be limited by time delays and the probe’s position in...
orbit rather than in the atmosphere or on the surface. While both JAXA and ESA are currently investigating the possibility of more extended investigations with Planet-C (and the BepiColombo mission may also perform a Venus flyby [3]), no firm plans exist at present to further explore Venus from its surface.

Earth’s sister planet presents compelling questions, and a difficult challenge: and the best way to meet its challenge is with extended surface exploration and adaptability, neither of which can be achieved with orbiting probes operated remotely from Earth. Robotic explorers cannot be pre-programmed for the unexpected, yet human explorers cannot feasibly reach the surface of Venus. The mission design proposed herein represents an attempt to bridge this gap in exploration capability. By conducting telerobotic exploration from a position near Venus, with constant coverage and minimal signal delays, the best of human telescience and robotic exploration can be combined in a safe, cost-effective human expedition that can be accomplished with near-term technology.

2. MISSION AND TRAJECTORY DESIGN

2.1 Mission Overview

A successful implementation of the proposed mission would require a crewed spacecraft, launched to Venus on a transfer orbit with an approximate outbound flight time of 119 days. In this mission plan, the spacecraft is not captured into Venusian orbit. Instead, the crewed vehicle is targeted for a polar flyby upon arrival, which rotates its hyperbolic velocity vector such that its heliocentric velocity increases in magnitude to equal that of Venus, but is inclined slightly to the plane of Venus’ orbit. This new “proximity” orbit allows the crewed vehicle to loiter in the vicinity of Venus for approximately 15 months, permitting lengthy telerobotic exploration with constant planetary coverage. Round trip signal delays between the spacecraft and surface do not exceed 90 seconds throughout the duration of the mission. At the conclusion of proximity operations, the spacecraft is targeted for a second Venus flyby which returns its heliocentric velocity vector back to the ecliptic plane and restores its original magnitude, allowing the spacecraft to return to Earth. The entire round-trip flight requires two years to complete, and is illustrated in Fig. 1.

This mission design requires no significant innovations or new technology to undertake, nor is any propulsive or aerodynamic orbital capture required at Venus. Consequently, the entire flight can be accomplished using only one major propulsive manoeuvre departing Earth, with subsequent trajectory correction velocity changes less than 1500 m/s. Furthermore, this mission design can be accomplished using the same launch systems slated to be developed for NASA’s lunar return missions as proposed in reference [4]. Alternatively a dedicated set of Evolved Expendable Launch Vehicle (EELV)-class launchers can be used to deploy the spacecraft and injection propulsion system in stages. As a result, human explorers would be able to perform a comprehensive survey of Earth’s sister planet without the need to develop significant new space exploration assets: only the crewed vehicle actually being dispatched to Venus would require development, which itself would be far less complex than proposed lunar or Mars exploration vehicles from past studies [4].

2.2 Methodology

The principal assumptions used in the initial analysis of the proposed mission are listed below. Some of these assumptions were later relaxed where noted. Primary simplifications include:

![Fig. 1 Venus transfer and flyby profile (axes in kilometres).](image)
Venus Exploration using Piloted Flybys and Telepresence

• Circular and coplanar orbits of Earth and Venus;
• Two-body motion and point-to-point conic matching; and
• Instantaneous (impulsive) velocity changes, including gravity assists

The method of trajectory analysis employed in this paper makes use of the fundamental constraint in space mission design that, for any round-trip interplanetary mission, the difference between the change in mission anomaly of the traveler and the home planet (generally Earth) must be an integral number of revolutions (where mission anomaly refers simply to the angular displacement of either the traveler or home planet from a fixed datum located at the point of departure). This constraint, first investigated by Wertz [5], can be written as:

\[
\delta_{\text{trav}} - \delta_{\text{Earth}} = 2\pi n \tag{1}
\]

where the \( \delta \) terms represent the mission anomalies of the spacecraft (traveller) and Earth, respectively. The spacecraft mission anomaly in Eq. (1) is written first to keep \( n \) positive, since for inner solar system travel, the spacecraft mission anomaly will be larger than that of the Earth. Using \( \omega \) terms to denote the mean motions of each body, the total round-trip mission time will be:

\[
\tau = \frac{\delta_{\text{Earth}}}{\omega_{\text{Earth}}} = \frac{\delta_{\text{trav}}}{\omega_{\text{trav}}} - \frac{2\pi n}{\omega_{\text{Earth}}} \tag{2}
\]

\[
\tau = \frac{\omega_{\text{trav}}}{\omega_{\text{Earth}}} \cdot t_{\text{flight}} + \frac{\omega_{\text{Venus}}}{\omega_{\text{Earth}}} \cdot t_{\text{loiter}} - nP_{\text{Earth}} \tag{3}
\]

However, a mission using two planetary flybys separated by 180 degrees is also constrained, such that the loiter time must be an integral multiple of half the Venustian period. Thus, we can also write:

\[
t_{\text{loiter}} = \alpha P_{\text{Venus}} \tag{4}
\]

\[
\tau = t_{\text{flight}} + \alpha P_{\text{Venus}} \tag{5}
\]

where

\[
\alpha = \frac{k - 1}{2}
\]

and \( k \) is the total number of flyby opportunities at Venus. For a transfer orbit with a total sidereal period \( P_{\text{trav}} \), of which some total fraction \( \lambda \) is used by the traveller, Eqs. (3) and (5) can be combined to yield the expression:

\[
\lambda P_{\text{trav}} \left( 1 - \frac{\alpha_{\text{trav}}}{\omega_{\text{Earth}}} \right) + \alpha P_{\text{Venus}} \left( 1 - \frac{\alpha_{\text{Venus}}}{\omega_{\text{Earth}}} \right) = -nP_{\text{Earth}} \tag{6}
\]

which is equivalent to:

\[
\lambda P_{\text{trav}} \left( 1 - \frac{P_{\text{Earth}}}{P_{\text{trav}}} \right) + \alpha P_{\text{Venus}} \left( 1 - \frac{P_{\text{Earth}}}{P_{\text{Venus}}} \right) = -nP_{\text{Earth}} \tag{7}
\]

and can also be expressed as:

\[
\lambda \left( \frac{a_{\text{trans}}}{a_{\text{Earth}}} \right)^{\frac{3}{2}} + \alpha \left( \frac{a_{\text{Venus}}}{a_{\text{Earth}}} \right)^{\frac{3}{2}} = \lambda + \alpha - n \tag{8}
\]

Thus, for a given mission duration \( n \) and number of flyby opportunities \( k \), Eq. (8) yields the semi-major axis \( a_k \) required for the Earth-Venus transfer. Equations (7) and (8) can be used to search for trajectories of this variety between any two bodies in the solar system, and furthermore do not require analysis of Lambert’s Problem as in previous works [6,7]. If the Earth return flyby is used to restore the spacecraft onto its original heliocentric trajectory (i.e. restores all of the orbital elements of the original transfer), then \( \lambda = 1 \). It is noteworthy that the right-hand side of Eq. (8) is equal to the total mission time as a multiple of Earth sidereal years.

By imposing a single additional constraint—that the apheion of the transfer orbit is at Earth—the entire trajectory can be completely defined using the algorithm presented in Table 1. It is interesting to note that the outbound and return flight times can be reversed if desired, depending on which leg of the total mission is most desirable to expedite.

While Eq. (8) permits a relatively simple search for candidate trajectories, each \( a_k \) and its corresponding elements returned by the above algorithm must be checked against a series of additional constraints. In particular, \( a_k \geq 0.5(a_E + a_V) \), and that the approach velocity of the spacecraft at Venus is sufficiently small to permit a gravity-assisted turn of the required magnitude.

2.3 Ballistic Mission Profile

Ideally, the proposed mission would be accomplishable using only ballistic gravity assists. The geometry for a ballistic flyby with an approach velocity approximately collinear to that of Venus is illustrated in Fig. 2. Approaching Venus with a hyperbolic velocity \( V_r \), the spacecraft performs a swing-by manoeuvre which increases the magnitude of its heliocentric velocity to equal that of Venus, but with a different inclination, allowing it to effectively “hover” near Venus until a second flyby, occurring an amount of time \((k-1)P/2\) later. The second flyby is used to restore the spacecraft’s heliocentric trajectory to its original elements. In Fig. 2, the \( V_r \) vectors are in the same directions as the asymptotes of the hyperbolic flyby trajectory. From this figure, we can characterize the post-flyby geometry by a simple expression derived by applying the cosine law:

\[
\cos(\pi - \delta) = \frac{V_{\text{Earth}}}{2V_r} \tag{9}
\]

implying that, for an approach velocity vector tangential to that of Venus, the required turn angle \( \delta \) depends only on the spacecraft’s hyperbolic velocity and the planet’s orbital velocity. But it can also be shown , as in Ref. [8], that:

\[
\frac{1}{e} = \sin \left( \frac{\delta}{2} \right) = \left( 1 + \frac{V_{\text{Earth}}}{\mu V} \right)^{-1} \tag{10}
\]

where the eccentricity of the hyperbolic approach trajectory \( e \) is related to the radius of closest approach to Venus, \( r \) and the gravitational parameter of Venus, \( \mu_V \). Equations (9) and (10) can be used to estimate the required turn angle and corresponding flyby altitude for the candidate mission.
TABLE 1: Algorithm for Earth-Venus Transfer Definition.

<table>
<thead>
<tr>
<th>Calculation Step</th>
<th>Relevant Expression</th>
</tr>
</thead>
<tbody>
<tr>
<td>Transfer Period ((P_T))</td>
<td>(P_T = 2\pi \sqrt{\frac{a_p^3}{\mu_{sun}}})</td>
</tr>
<tr>
<td>Eccentricity ((e))</td>
<td>(e = \frac{a_E}{a_T} - 1)</td>
</tr>
<tr>
<td>True Anomaly ((f))</td>
<td>(\cos f = \frac{1}{e} \left( \frac{a_T (1 - e^2)}{a_T} - 1 \right))</td>
</tr>
<tr>
<td>Eccentric Anomaly ((E))</td>
<td>(\cos E = \frac{e + \cos f}{1 + e \cos f})</td>
</tr>
<tr>
<td>Mean Anomaly ((M))</td>
<td>(M = E + e \sin E)</td>
</tr>
<tr>
<td>Time from Perihelion ((TFP))</td>
<td>(TFP = M \sqrt{\frac{a_T^3}{\mu_{sun}}})</td>
</tr>
<tr>
<td>Outbound Time of Flight ((TOF_{outbound}))</td>
<td>(TOF_{outbound} = \frac{P_T}{2} - TFP)</td>
</tr>
<tr>
<td>Return Time of Flight ((TOF_{return}))</td>
<td>(TOF_{return} = P_T - TOF_{outbound})</td>
</tr>
</tbody>
</table>

Fig. 2 Ballistic Flyby Geometry (not to scale).

£.4 Candidate Trajectory and Periapse

Hyperbola Matching

Using Eq. (8), it can be shown that the shortest possible mission duration using the trajectory design presented herein requires a total time of two years, with \(k=5\) flyby opportunities in total, and has a semi-major axis \(a_p = 1.265 \times 10^8\) km. This corresponds to outbound and return Venus flight times of approximately 119 and 165 days, in either order.

However, a spacecraft following this trajectory would have an inbound hyperbolic velocity at Venus of approximately 5 km/s, which is too great in magnitude for a purely ballistic flyby manoeuvre without propulsive augmentation. Fortunately, this can be corrected with a small correction burn (i.e. deceleration) applied at periapse, which is generally the most efficient point in a trajectory for adding or subtracting energy. The required correction \(\Delta V\) can be determined by matching the inbound and outbound (i.e. desired) hyperbolae at the periapse of the second and determining the velocity difference between them, as shown in Fig. 3.

Choosing a minimum allowable flyby altitude of 300 km and solving Eqs. (9) and (10), the outbound trajectory is required to have the properties

\[ V_w^- = 4.363\text{ km/s}, \; \delta = 93.56^\circ \]

and

\[ V_r^- = \sqrt{2\mu_r / r + \left(V_w^-\right)^2} = 11.01\text{ km/s} \]

For the inbound trajectory, we find that

\[ V_w^+ = 4.95 \text{ km/s} \]

and

\[ V_r^+ = \sqrt{2\mu_r / r + \left(V_w^+\right)^2} = 11.26 \text{ km/s} \]

Thus, the correction \(\Delta V\) required to enter the loitering trajectory at this altitude will be:
The correction \( \Delta V \) is retrograde on approach, and prograde for Earth return. The total velocity change required for trajectory augmentation in this mission design will therefore be approximately 490 m/s. Figure 4 shows the propulsive \( \Delta V \) required for this manoeuvre as a function of minimum flyby altitude. A sample mission timetable for a 2015 departure is presented in Table 2.

A trajectory similar to that proposed herein was previously investigated in [9] as a method of undertaking comparable piloted flybys of Mars; however, Mars—like Venus—is insufficiently massive to support a purely ballistic conversion from transfer to loitering orbit for spacecraft travelling at high approach velocities. Indeed, even for a minimum-energy Earth-Mars transfer, the red planet is insufficiently massive to support a purely ballistic loitering mission. A comparison of approach velocity and required flyby altitude for both Venus and Mars is presented for reference in Fig. 5. By far, Venus prevails over Mars as a candidate for this variety of mission in terms of propulsive requirements.

### 3. SPACECRAFT DESIGN AND SIZING

Spacecraft design for this mission plan is somewhat simpler compared to crewed vehicles intended for surface landings, resulting from the absence of manoeuvres such as orbital capture/aerobraking, descent, landing, or ascent, which generally dominate the design considerations for crewed spacecraft and drive mission mass upwards significantly.

In this mission design, after trans-Venus injection the proposed spacecraft can essentially coast for the duration of the mission, with only the relatively small flyby corrections and subsequent deterministic manoeuvres required after Earth departure. Therefore, the primary design concerns surround issues of radiation projection and thermal control, resulting from the extended period of time during which the spacecraft will be in relatively close proximity to the Sun. At Venus’ distance from the Sun, the solar constant is almost double that at Earth, depositing approximately 2600 W/m² on exposed surfaces. Astronauts will correspondingly require a greater amount of shielding to protect against increased solar radiation and the possibility of solar particle events. Fortunately, in the absence of landers, aeroshields and field equipment mass, much greater margin exists for increasing the mass of subsystems required to combat these issues.

#### 3.1 Crew Size

Crew size recommendations for this mission is primarily driven by the desire to utilize as much of the complete 24-hour coverage of Venus as possible, while still maintaining the smallest functional crew number possible. In the interests of balancing mission mass and the effectiveness of the expedition, a reference crew number of four is assumed in this mission, intended to work in two shifts of approximately 10 hours per day, with 14 hours dedicated to sleeping, eating, personal hygiene, recreation and exercise as recommended in [10]. There are, however, disadvantages to having astronauts work in shifts: with
different crew members performing differing tasks simultaneously, sleeping and recreation areas must be isolated from one another within the spacecraft, likely with sound-proof bulkheads and variable lighting levels, which will increase spacecraft mass. This is viewed as both necessary and acceptable in this mission design, in the interests of maximizing both crew comfort and productivity.

3.2 Habitat Design

Assuming a habitable volume per crew member of 30 m$^3$ (which is a generous 10 m$^3$ greater than that recommended in [10]) comprising 50% of the total spacecraft pressurized volume, the required internal space can be enclosed by a cylindrical module 5 m in internal diameter and 12.2 m in height. A spacecraft with this module can benefit within expanded payload fairings of the Boeing Delta IV-H, Lockheed Atlas V, or proposed NASA Ares I. However, if a different launch system were used, and the maximum exterior diameter was limited to 4.5 metres (i.e. to fit within fairings of the Russian Proton Breeze M or European Ariane V), then either a height increase to approximately 15 metres or a decrease in the total volume per astronaut to the more conventional value of 20 m$^3$ would be required. For the habitat module, work areas and recreation/sleeping areas could be easily separated by 3 deck—the first containing all command and control instruments and workstations, with the lower two containing galley/recreation areas and crew quarters, respectively.

3.3 Deterministic Electric Propulsion System (DPS)

The candidate spacecraft for this mission would use a ballistic flyby to alter its heliocentric Earth-Venus transfer into a circular orbit at Venus, but inclined to the plane of the planet’s orbit. If, however, the spacecraft is allowed to continue on its post-flyby course unaltered, it would naturally re-approach Venus approximately 112 days later as it neared its first nodal crossing. Without correction, this would result in a second flyby that would alter the spacecraft’s trajectory, throwing it somewhere towards the vicinity of Earth’s orbit, but nowhere near the Earth itself. In Table 2 it was shown that four nodal crossings would be required after the initial flyby, with the spacecraft targeted for an Earth return manoeuvre only at the fifth opportunity. Consequently, the candidate spacecraft in this mission must utilize an integral propulsion system to either regress or progress the line of nodes of its orbit to the extent of the Venusian gravitational sphere of influence (approximately 6.2 x 10$^5$ km) to avoid ill-timed flybys, and will also require such propulsion to mitigate perturbations to its trajectory throughout the course of the mission resulting from close proximity to Venus.

A solar electric propulsion system is assumed for deterministic manoeuvres in this mission design. Electric propulsion systems feature extremely high exhaust velocities, which significantly reduce the mass of propellant required for a given velocity change, and strongly prevail over chemical propulsion systems when low thrust, long duration burns are acceptable.

A solar electric DPS for this mission was sized using a power-free method discussed at length in [10] and [11]. For electric propulsion systems, the low-thrust rocket eq. Can be written as:

$$ R = \frac{m_{\text{payload}}}{m_{\text{initial}}} = e^{-\frac{\alpha}{r \eta \tau}} \left(1 - e^{-\frac{\Delta v}{\alpha \eta}}\right) \left(\frac{c^2}{2 \alpha n \tau_b}\right) \quad (11) $$

where $\alpha$ is the specific power mass of the propulsion system (W/kg), $\eta$ is the electric efficiency between the spacecraft power generation system and the engine’s exhaust, and $\tau_b$ the total engine burn time. Taking the engine’s maximum velocity change requirement as 1000 m/s (estimated from [8]), $\alpha = 100$ W/kg (using flexible solar panels [10]) and $\eta = 0.85$, then for burn durations varying from 60 to 90 days (even the latter of which gives the spacecraft a 20% time margin to perform the entire manoeuvre) we can generate performance curves for the candidate propulsion system as shown in Fig. 6. It can be seen that, for all burn time assumptions, the mass ratio of a spacecraft using electric propulsion reaches a maximum value, after which additional increases in exhaust velocity actually result in decreased effectiveness. The optimal exhaust velocity for the candidate DPS can be calculated as $c = 35,860$ m/s, simply by taking the derivative $dR/d\tau$ from Eq. 11, corresponding to a specific impulse of approximately 3655 s and a spacecraft mass ratio of 0.945. However, this value may prove
unrealistically high in terms of performance. A more modest and attainable choice of \( c = 20,000 \) m/s corresponds to a specific impulse of approximately 2040 s, which decreases the overall spacecraft mass ratio by only 1% from the optimal value—a negligible loss in overall performance.

After Venus approach and flyby, the deterministic propulsion system would be used to regress the node of the spacecraft’s orbit beyond the sphere of influence of Venus; then, after the fourth flyby opportunity, the system would again be used to restore the node to its original value, allowing the spacecraft to perform the appropriate gravity assist for Earth return.

Mass allocations for the candidate spacecraft and deterministic propulsion system are shown in Table 3. It is noteworthy that decreasing the mission crew size to three astronauts would correspond to an overall injection mass of 48.7 tonnes, as opposed to the 63.6 tonne value presented in Table 3.

4. SPACECRAFT LAUNCH AND DEPLOYMENT

Two possible methods of launching and dispatching the spacecraft for this mission were considered: the first using launch systems and spacecraft slated to be developed by NASA in support of the recent Vision for Space Exploration (VSE), and the second using a series of EELVs to deploy candidate spacecraft and injection propulsion in stages to be assembled on orbit.

Should NASA successfully develop its proposed Crew Exploration Vehicle (CEV), corresponding Ares I, and shuttle-derived Ares V, then the spacecraft proposed herein could be launched and injected to Venus using a sequence identical to that currently proposed for NASA expeditions to the moon. In such a scenario, a single launch of NASA’s Ares V would be used to deliver the proposed Venus Transfer Habitat (VTH) and DPS to low-Earth orbit (LEO), along with an Earth Departure Stage (EDS) similar to that currently outlined for trans-Lunar injections [4]. After successful deployment of the VTH/DPS/EDS, the 4-person crew would be launched to orbit in a stripped down CEV using the Ares I, which would rendezvous and dock to the VTH. The crew would then transfer to the VTH and power the CEV down to a quiescent operating mode for the duration of the mission. The CEV service module may then be discarded since its systems would no longer be necessary to supplement the VTH. Alternatively, the CEV propulsion system may be used instead of a dedicated cruise stage to perform flyby corrections at Venus. Once assembled, the spacecraft would be injected by the EDS onto the outlined near-minimum energy Venus transfer (\( C_3 = 8.55 \text{ km/s}^2 \)).

As an alternative to this approach, the candidate VTH spacecraft could also be launched and deployed by a series of six EELV launch systems with between 20 and 25 tonne ETO capacities, with the crew delivered on a seventh subsequent launch in an Earth return capsule such as NASA’s CEV or another comparable transfer vehicle such as an augmented Russian Soyuz capsule. In this scenario, the first launch would deliver the primary habitat module to LEO, after which a second launch would deliver a service module and the deterministic propulsion system, which would be mated to the pressurized module autonomously. Four supplementary launches would then be used to deliver four high energy, hydrogen/oxygen propulsion stages to the VTH, which would be mated aft like train cars. Upon complete assembly (assumed to require approximately six months, for the purposes of predicting propellant boiloff), the crew would then be delivered in a small transfer capsule, which would mate to the assembly and power down for the duration of the expedition. The propulsion stages would then be ignited at successive peripeces, widening the orbit of the VTH until the final burn imparts sufficient energy to initiate the required trans-Venus injection. Proposed vehicle designs and configurations are illustrated in Fig. 7.

In both scenarios, at the conclusion of the mission the crew would transfer back to the attached capsule (CEV or equivalent) and employ a direct Earth entry from a hyperbolic approach trajectory, subsequently landing on the surface using either airbags or an Apollo-style ocean splashdown. While both deployment schemes involve orbital assembly to some extent, it is noteworthy that NASA has been performing comparable manoeuvres since the 1960s, and that orbital assembly is one of the operations with which today’s space industry actually has considerable experience.

5. SURFACE EXPLORATION

The effectiveness of the proposed mission is principally limited by the capability and durability of the robotic explorers used in the Venusian environment. A mission predicated on lengthy periods of exclusively operating robotic explorers from a distance will be significantly improved by the ability to maximize the operator’s sense of presence and functionality. Therefore, telepresence can significantly contribute to the return from the proposed mission by offering additional mission flexibility. True telepresence—in which operators are able to view robotic explorers as physical extensions of themselves—has arguably not yet been realized in robotic space exploration, as a consequence of generally lengthy time delays experienced by unmanned round-trip communications. The time delay drawback also severely limits the extent to which remote operators can react quickly or intuitively to the unexpected.

The mission design proposed herein would eliminate the significant communications time lag that has, to date, characterized robotic space exploration.

By reducing the problem of signal delay, the remaining issues surrounding the establishment of Venusian telepresence are ones of survivability and human/machine interaction. The latter has been discussed at length in previous work [10], and surface exploration options for both surface and aerial robotic explorers have
TABLE 3: Spacecraft Mass Allocations. Initial Mass Estimate (based on statistical data). IME = 592(crew number x mission duration [d] x pressurized volume [m$^3$]$^{0.346}$ = 62359 kg.

<table>
<thead>
<tr>
<th>System</th>
<th>Sub-System</th>
<th>Description</th>
<th>Mass Allocation [kg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Structure</td>
<td>Primary</td>
<td>12% of IME</td>
<td>7,483</td>
</tr>
<tr>
<td></td>
<td>Airlock</td>
<td>2 crew, dual egress</td>
<td>1,000</td>
</tr>
<tr>
<td></td>
<td>Mechanisms</td>
<td>8% of IME</td>
<td>4,989</td>
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<tr>
<td>ADCS/RCS</td>
<td>Air</td>
<td>2% of IME</td>
<td>1,247</td>
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<tr>
<td>ECLSS</td>
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<td>VCD x2 Redundant</td>
<td>200</td>
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<tr>
<td>Thermal</td>
<td>Active</td>
<td>4BMS, TCCS, OGA, x3 Redundant</td>
<td>1,020</td>
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<tr>
<td></td>
<td>Radiator</td>
<td>2.6% of IME</td>
<td>1,621</td>
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<tr>
<td></td>
<td>TPS</td>
<td>1.4% of IME</td>
<td>873</td>
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<tr>
<td>Crew Systems</td>
<td></td>
<td>See Reference 10</td>
<td>13,556</td>
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<tr>
<td>EVA</td>
<td></td>
<td>1 suit per crew member</td>
<td>400</td>
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<tr>
<td>Logistics</td>
<td></td>
<td>5.0% of IME</td>
<td>3,118</td>
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<tr>
<td>Power</td>
<td>Battery</td>
<td>50 kW-h Li-Ion Batteries, 80% DOD</td>
<td>300</td>
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<td></td>
<td>PVA</td>
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<td>Dry Mass Total</td>
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<td>Command Module</td>
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<td>Service Module</td>
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</tr>
<tr>
<td>DPS</td>
<td>Structural Mass</td>
<td></td>
<td>1,120</td>
</tr>
<tr>
<td></td>
<td>Propellant Mass</td>
<td></td>
<td>300</td>
</tr>
<tr>
<td>Injection Mass Total</td>
<td></td>
<td></td>
<td>63,590 kg</td>
</tr>
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</table>

Key: 4BMS - Four-Bed Molecular Sieve; ADCS - Attitude Determination and Control System; DOD - Depth of Discharge; DPS - Deterministic Propulsion System; ECLSS - Environmental Control and Life Support System; EVA - Extra Vehicular Activity; IME - Initial Mass Estimate; OGA - Oxygen Generation Assembly; PVA - Photovoltaic Array; RCS - Reaction Control System; TCCS - Trace Contaminant Control System; TPS - Thermal Protection System; VCD - Vapour Compression Distillation.

similarly been widely discussed in past work, of which [1,2,12] represent particularly promising concepts. Further discussion of specific options, however, is beyond the scope of this work.

5.1 Communications Jitter

While round-trip signal delays for exploration rovers are significantly reduced in this mission design, a small, time-varying signal delay (communications jitter) will nevertheless be experienced throughout the duration, varying periodically with minimum and maximum values occurring at spacecraft equinox and solstice, respectively (Fig. 8).

The significance of communications jitter varies depending on the control system used for teleoperated vehicles, because certain control techniques require time stamping on control data packets. For the proposed mission, the round-trip signal delay can be determined as a function of the minimum flyby altitude $h$ and the mission time $t$ as:

$$\rho(t) = \frac{c}{2} \left( \frac{2a_i^2(1 - \cos(i))}{c^2 - h} + \sin \left( \frac{2\pi t}{P} + h \right) \right)$$  \hspace{1cm} (13)

Where $c$ in this equation is the speed of light, $i$ is the inclination of the spacecraft’s heliocentric orbit, and $P$ is its period. The round-trip signal delay for the first half-orbit of the sample mission is shown in Fig. 8, and reaches its peak value of approximately 90 seconds on November 28, 2015, repeating every $P$ afterwards.
Both ground and aerial vehicles in this mission design would likely communicate via S-band transmissions to and from the orbiting VTH spacecraft to minimize atmospheric attenuation. This has been previously utilized by several Earth-orbiting satellites for ground communications.

6. COST ESTIMATES

First order cost estimates for the proposed mission are presented in Table 4. All figures assume spacecraft development costs of approximately $25,000 per kilogram, and launch vehicle costs not cited are estimates using averaged fixed and variable costs for each vehicle available at the time this work was prepared. It is noteworthy that the mission configuration using a smaller number of EELVs is predicted to be approximately equal in cost to that using NASA’s proposed heavy-lift launch system. Regardless, the proposed expedition to Venus may be undertaken using a wide variety of different launch system combinations, and within the technological and fiscal capabilities of today’s major space agencies.

7. DISCUSSION

The mission design discussed herein represents a means of exploring perhaps the most scientifically mysterious world in our solar system using either current or near-term technologies and launch vehicles, with the principal technological limitations being those imposed by the design of robotic explorers rather than piloted spacecraft. While the creation of a robotic exploration architecture capable of surviving the complete 15-month duration of the proposed mission represents a significant challenge, technologies are increasingly emerging that offer great promise for long-lived endurance and operation in the Venussian environment. Furthermore, if a series of candidate Venus exploration payloads were delivered to orbit prior to the arrival of the crewed mission, and their surface deployment staggered over the course of the mission, then engineers would be able to avoid what may otherwise represent unrealistic design life goals for robotic explorers.

A human expedition to Venus—while scientifically interesting—is not, however, likely to garner significant consideration as a near-term destination in human spaceflight: nor is it the intention of this paper to advocate that it should. However, as our capabilities in both crewed and robotic space exploration continue to mature, consideration and scientific interest may yet turn back towards Earth’s sister world. By undertaking an early human expedition to Venus using the methods discussed here, astronauts may be able to explore and characterize a new planet while simultaneously avoiding many of the key risks associated with lunar or Mars expeditions (such as those involving surface landings and direct exploration).

Should NASA continue to develop the resources proposed for the Vision for Space Exploration—and should early return missions to the moon meet with success—a two-year trip to Venus using lunar exploration assets may come to be viewed as a logical intermediate step to Mars. This would allow experimentation with both long-duration spaceflight and telepresence in a way not yet attempted, but in a far less risk-intensive, remote, or long duration venture.

8. CONCLUSIONS

A human expedition to near-Venus space was proposed, which would use a double Venussian flyby to execute a complete round trip using only one low-energy trans-Venus injection manoeuvre departing Earth, and several small correction manoeuvres (total $\Delta V < 1500 \text{ m/s}$) thereafter. A lengthy period in proximity to Venus would be used for conducting extensive telescience in order to characterize a world with mysteries of great significance to our understanding of climatology, weather forecasting, geology and planetary evolution.

Candidate spacecraft designs and deployment scenarios

![Graph](image)

**Fig. 8** Signal delay as a function of mission time between k=1 and k=2.

### TABLE 4: Mission Cost Estimates [FY 2007].

<table>
<thead>
<tr>
<th>Item</th>
<th>Comments</th>
<th>Cost Estimate</th>
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<tbody>
<tr>
<td>VTH Development Costs</td>
<td>$25 million/tonne x 45 tonnes</td>
<td>$1125 million</td>
</tr>
<tr>
<td>Crew Launch</td>
<td>Use Ares I/CEV</td>
<td>$200 million</td>
</tr>
<tr>
<td><strong>Robotic Exploration Architecture</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Surface Rovers</td>
<td>4 x $250 million</td>
<td>$1000 million</td>
</tr>
<tr>
<td>Solar Powered Aircraft</td>
<td>3 x $75 million</td>
<td>$225 million</td>
</tr>
<tr>
<td><strong>Deployment Scenario</strong></td>
<td></td>
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</tr>
<tr>
<td>5 EELV launches</td>
<td>5 x $200 million</td>
<td>$4438 million</td>
</tr>
<tr>
<td>5 Proton/Breeze M Launches</td>
<td>5 x $70 million*</td>
<td>$3625 million</td>
</tr>
<tr>
<td>1 NASA Ares V</td>
<td>1 x $1000 million</td>
<td>$4438 million</td>
</tr>
</tbody>
</table>

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were presented, and it was argued that an expedition as proposed herein can be accomplished using relatively simple vehicle designs and existing or near-term launch systems. And first order cost estimates were found to be modest with respect to other current and planned space exploration investments. As both crewed and robotic exploration capabilities of today’s space agencies continue to evolve over the next two decades, it is recommended that conducting a human expedition to Venus as proposed herein be investigated in earnest as a possible extension of expanding human spaceflight capabilities.

ACKNOWLEDGEMENTS

The trajectory illustrations in Figs. 2 and 3 were created by Ryan Hewer. Tim Hocken provided valuable editing, and Geoffrey Landis gave valuable engineering insights and suggestions.

REFERENCES


(Received 28 June)

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