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Systems Analysis for a Venus Aerocapture Mission

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March 2006

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Abstract

Previous high level analysis has indicated that significant mass savings may be possible for planetary science missions if aerocapture is employed to place a spacecraft in orbit. In 2001 the In-Space Propulsion program identified aerocapture as one of the top three propulsion technologies for planetary exploration but that higher fidelity analysis was required to verify the favorable results and to determine if any supporting technology gaps exist that would enable or enhance aerocapture missions.

A series of three studies has been conducted to assess, from an overall system point of view, the merit of using aerocapture at Titan, Neptune and Venus. These were chosen as representative of a moon with an atmosphere, an outer giant gas planet and an inner planet. The Venus mission, based on desirable science from plans for Solar System Exploration and Principal Investigator proposals, to place a spacecraft in a 300km polar orbit was examined and the details of the study are presented in this paper.

Results indicate that a significant mass savings is possible when aerocapture is used instead of all propulsive or aerobraking missions. This equates into more mass for science and/or lower required launch vehicle performance. Based on this analysis there are no new technologies required to enable this mission. However, there are technologies in the areas of thermal protection and guidance, navigation and control which can significantly enhance the mission.

Introduction

In order to place a spacecraft in orbit around a planet or moon sufficient velocity must be removed such that the gravitational field of the target body will transform the approach hyperbolic trajectory into a closed elliptical orbit. Traditionally this has been accomplished using retro-rockets to provide deceleration forces to slow the spacecraft to the required velocity. For bodies that have an atmosphere, using atmospheric drag can be a viable method of delivering some or all of the deceleration forces. “Aerobraking,” where a vehicle makes numerous passes through the atmosphere has been demonstrated, most recently during the Mars Odyssey mission, to be operationally viable while delivering an increase in mass to orbit over that which would have occurred if an all rocket mission were conducted. The Odyssey mission used a retro-rocket to remove sufficient velocity to place the spacecraft in a highly elliptical orbit about Mars and then aerobraking to remove remaining excess velocity to circularize the final science orbit. This aerobraking is operationally intense as it required some 315 passes through the Martian atmosphere over 77 days.

In “aerocapture” sufficient velocity is removed, due to the drag on the vehicle, during a single pass through the atmosphere that the spacecraft is placed in a closed orbit from the original hyperbolic orbit. This aerocapture maneuver, while unproven operationally, has been shown in simulation to be feasible and to provide even more significant mass savings than aerobraking because the propulsion system mass and propellant is essentially eliminated. Performance projections for aerocapture show a vehicle mass savings of between 40 and 80-percent, depending on destination, for an aerocapture vehicle compared to an all-propulsive chemical vehicle. In addition, aerocapture is applicable to multiple planetary and satellite exploration destinations of interest to the NASA Science Mission Directorate. These results led to the identification of aerocapture as one of the top three propulsion technologies for solar system exploration missions during the 2001 NASA In-Space Propulsion Program (ISP) technology prioritization effort, led by Marshall Space Flight Center, to rank current ISP propulsion technologies. An additional finding was that aerocapture needed a better system definition and that supporting technology gaps needed to be identified.

An aerocapture systems analysis effort focused on aerocapture at Titan with a rigid aeroshell system was completed in September 2002[1]. Titan was selected as the initial destination for the study due to potential interest in a follow-on to the on-going Cassini/Huygens mission. The systems analysis was conducted by a multi-center NASA team led by Langley Research Center which included scientists and engineers from Ames Research Center, Jet Propulsion Laboratory, Johnson Space Center, Langley Research Center, and Marshall Space Flight Center. Because of the success of this original study, aerocapture systems analyses have been completed for Neptune[2], as representative of the giant gas planets, and Venus as representative of terrestrial planets with an atmosphere. This report will present a summary of the results for the Venus Aerocapture Performance Review as presented on September 29, 2004[3].

Background

An aerocapture flight profile schematic is shown in Figure 1. The vehicle approaches the planet/moon from an approach hyperbolic trajectory, shown at point 1, designed to achieve state conditions including flight path angle at atmospheric interface, point 2, within a predetermined range. Bank angle modulation, for rotation of the lift vector about the velocity vector, is initiated by the guidance system at point 3. The drag on the vehicle as it passes through the atmosphere provides reduction in velocity or negative "delta-V" required to capture the vehicle into the desired elliptical orbit. The amount of delta-V imparted to the vehicle is controlled by the on-board guidance by modulating bank angle, i.e., the direction of the vehicle's lift vector. A command of lift up during an atmospheric pass results in increased altitudes, nominally decreased atmospheric density, yielding reduced drag and reduced delta-V imparted. A command of lift down results in decreased altitudes, nominally increased atmospheric density, yielding increased drag and increased delta-V imparted. Bank angle modulation is commanded throughout the atmospheric pass from point 3 to point 5. By point 5, where the influence of aerodynamic forces is no longer significant, the energy depleted from the initial hyperbolic trajectory is that required to capture the vehicle into the desired elliptical orbit. At apoapsis, point 7, a small delta-V burn is performed to increase velocity to raise the periapsis out of the atmosphere.

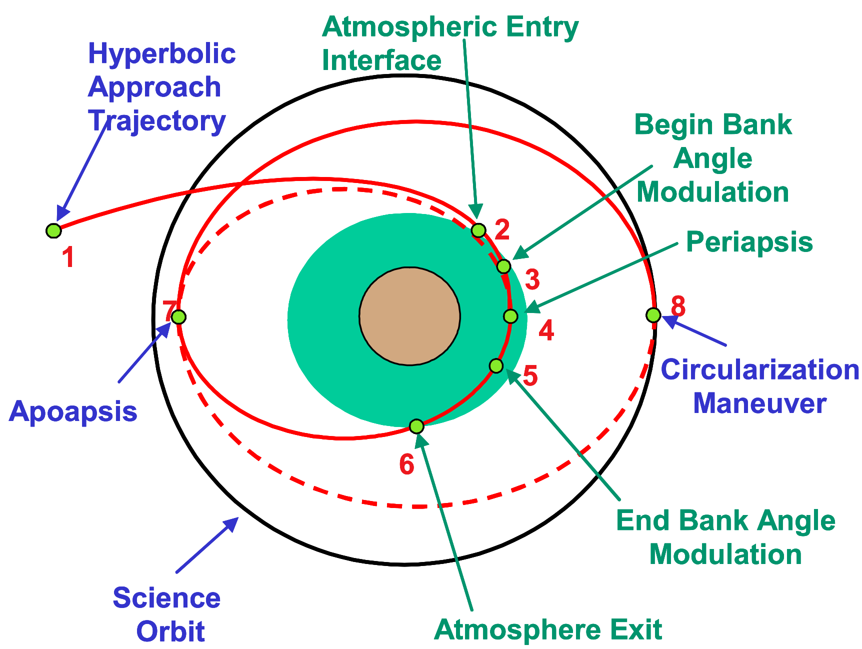


Figure 1 Aerocapture Flight Profile

The aerocapture theoretical corridor is bounded by the full lift up trajectory and the full lift down trajectory for a nominal atmosphere and vehicle aerodynamics. Thus, the theoretical corridor width is defined by the difference between the entry flight path angle corresponding to a full lift down trajectory and the entry flight path angle corresponding to a full lift up trajectory for a nominal atmosphere and aerodynamics. If the vehicle enters the atmosphere at a flight path angle steeper than defined by the full lift up trajectory the vehicle lands. If the vehicle enters the atmosphere at a flight path angle shallower than that defined by the full down trajectory, the vehicle is not captured. Figure 2 illustrates the theoretical corridor and the effects of navigation, atmosphere and aerodynamic uncertainties on the theoretical corridor. The plot, using Titan as an example, shows the entry flight path angles for a full lift up and full lift down trajectory as a function of the atmosphere variable Fminmax, where Fminmax=0 is the nominal atmosphere, Fminmax=-1 is the lowest density atmosphere, Fminmax =+1 is the maximum density atmosphere. The plot is shown for a given vehicle, entry velocity, and target orbit. To first order, the aerocapture corridor required to accommodate atmospheric dispersions, navigation errors (delivery flight path angle), and aerodynamic uncertainties can be root sum squared to determine the total corridor required. If the theoretical corridor is significantly greater than the corridor width required, then the vehicle control authority is estimated to be adequate. However, factors such as high frequency atmospheric density perturbations are not included in this approach and can affect the results. Monte Carlo simulation analyses must be completed to assess feasibility and robustness.

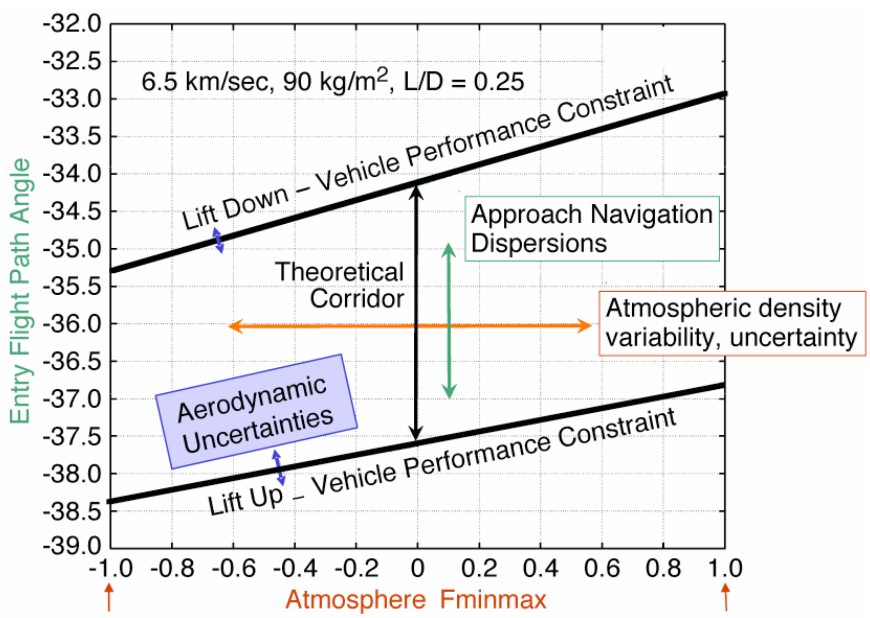


Figure 2 Theoretical Aerocapture Entry Corridor

Concerns expressed regarding the risk of aerocapture have largely been in three areas: 1) atmosphere variability and uncertainty; 2) approach navigation delivery errors; and 3) aerodynamics uncertainty utilized to control capture. To address these concerns, the following approach can be taken in the design of the aerocapture system:

1. Based on available atmospheric measurements, quantify and model the physical range of atmosphere variability and uncertainty.
2. Quantify the approach navigation delivery errors, and incorporate navigation systems into the vehicle design to reduce errors.
3. Quantify the aerodynamic uncertainties including margin.
4. Select a vehicle L/D to provide adequate control authority.
5. Provide adequate vehicle control responsiveness to accommodate perturbations, including atmospheric perturbations.
6. Develop a robust guidance algorithm and system.
7. Evaluate aerocapture robustness through Monte Carlo simulation incorporating all variability, uncertainty, errors and dispersions.
8. Design the aerocapture system to provide margin *above* 3-σ success, in particular for first time flights and for high value payloads.

The objectives for the aerocapture systems studies were to provide higher fidelity analyses for validation and updates to aerocapture assumptions made in earlier mission studies, including performance, environments, mass properties, etc. The results of the analysis were to be provided to scientists, mission planners, technology planners, technologists and future mission managers. The feasibility, benefit and risk of aeroshell aerocapture system and technologies for each destination were to be defined. Technology gaps were to be identified and performance goals of key technologies defined.

Approach

The multi-center aerocapture systems analysis team included NASA engineers and scientists from Ames Research Center (ARC), the Jet Propulsion Laboratory (JPL), Johnson Space Center (JSC), Langley Research Center (LaRC), and Marshall Space Flight Center (MSFC). The various centers brought the following skills and capabilities to Venus study: ARC, initial aeroheating and TPS sizing estimates; JPL, mission design, science requirements, spacecraft initial design and mass properties, and navigation; JSC, vehicle guidance and performance; LaRC, team leadership, project integration, trajectory simulation, aerodynamics, vehicle performance and initial aeroheating estimates; and MSFC, atmospheric modeling.

Venus Science Areas of Interest

Based on the Science Priorities from the Solar System Exploration Decadal Survey[4] and from the Principal Investigators' proposals for the Venus missions within the Discovery Program, four main science areas of interest have been identified.

1. A determination of the Venus atmosphere including the lower atmosphere composition, isotopic ratios, trace gasses and meteorology, measurement of thermal profiles, identification of the 4-dimensional structure, composition and aerosol size of the cloud layers, and the global circulation and super-rotation.
2. The determination of the surface composition and observation of active and recent volcanic processes including out gassing and heat loss mechanisms.
3. Determining the interactions between the surface and the atmosphere.
4. The interaction between the upper atmosphere and the solar wind including atmospheric mass loss rates and upper atmospheric heating.

These four areas will likely require instruments capable of measurements in the infrared (IR) window of the atmosphere for global imaging spectroscopy and in the microwave range for radiometry measurements to infer temperature, pressure and atmospheric opacity as a function of altitude. In addition, in order to determine the interaction between the upper atmosphere and the solar wind, measurements of low energy neutral and charged particles will be necessary.

A potential instrument suite, an IR Imaging Spectrometer, a Microwave Radiometer and a Low Energy Particle Detector, was chosen for this study to allow estimates of mass and volume for spacecraft packaging and sizing. The spacecraft must be placed in a near circular, polar orbit providing total planet coverage for an extended period of time. A polar orbit will not require orbit evolution as planetary rotation will provide the full longitudinal coverage and a 300km orbit, as will be discussed, will be stable with minimal need for station keeping for up to and beyond two full Earth years in duration. It is proposed that an aerocapture mission with an entry over the Venus north pole could deliver such a spacecraft to the desired science orbit at a significant mass savings as compared to other methods of orbital insertion at Venus.

Navigation and Mission Design

The interplanetary trajectory delivers the spacecraft to the atmospheric interface point at a planet or moon for entry through the atmosphere. In this case flight through the atmosphere will yield an aerocapture into orbit rather than an entry, descent and landing on the surface. Accurate navigation, that is, determining exactly where the spacecraft is relative to the ideal trajectory, is critical as the uncertainty in initial conditions has a large effect on the margins for successful aerocapture. “Misses” can lead to surface impact or having the spacecraft skip off of the atmosphere as the guidance and control systems are fairly limited in capability because of low vehicle performance. Even without a catastrophic failure, poor navigation and initial conditions can leave the spacecraft in a less than useful science orbit.

For this analysis, a launch from Earth between 21 October and 11 November, 2013, on a Type I trajectory, figure 3, with C3= 8.3 km^2/sec^2, yields an entry for aerocapture at Venus on 7 April, 2014 over the north pole. Nominal entry parameters will be a velocity of 11.25 km/sec, with an entry interface altitude of 150 km, at an entry flight path angle of -6.12 degrees, at latitude of 86 degrees and inclination of 90 degrees. After the aerocapture pass, this should yield a 300km circular polar orbit following a periapsis raise maneuver at entry plus 45 minutes, at apoapsis.

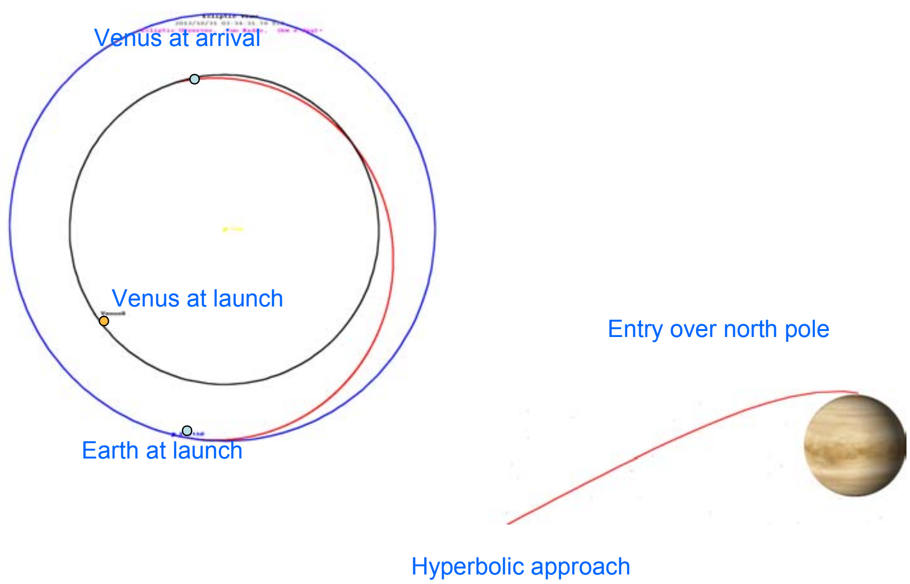


Figure 3 Type I Earth-Venus Trajectory With Entry Over The North Pole

As mentioned, uncertainty in the entry state can affect the resulting aerocapture pass. The current navigation model relies on doppler and range radiometric data only with 3-sigma measurement errors of 0.3mm/sec and 20m respectively. Navigation is limited to these because there is no Global Positioning System or "GPS" available away from the Earth and the obscuring atmosphere and lack of natural satellites at Venus render optical navigation imprecise at best and completely unusable at worst. The dominant sources of error in the navigation model are the Venus ephemeris and the knowledge of the vehicle state. However, because of the large numbers of Venus missions and observations, the ephemeris is quite well known.

Estimated errors in the vehicle's position are indicated in table 1 and result in flight path angle uncertainty of +/- 0.4 to 0.2 degrees depending on whether the last update to the spacecraft inertial system is at entry minus 48 hours or 5 hours. Clearly the later the final update can occur the less error there will be in the integrated inertial system calculation of the vehicle state.

Table 1 Estimated Errors on Vehicle Position

	Doppler & Range
Data Cutoff at E-48 hours	
Semi-major axis (km)	10.7
Semi-minor axis (km)	4.6
Ellipse angle (deg)	86
Entry time (s)	6.5
B magnitude (km)	10.7
Flight Path Angle (deg)	**±0.4**
Data Cutoff at E-30 hours	
Semi-major axis (km)	8.1
Semi-minor axis (km)	3.9
Ellipse angle (deg)	72
Entry time (s)	5.2
B magnitude (km)	7.8
Flight Path Angle (deg)	**±0.3**
Parameter Update	
Data Cutoff at E-5 hours	
Semi-major axis (km)	5.7
Semi-minor axis (km)	2.5
Ellipse angle (deg)	70
Entry time (s)	3.6
B magnitude (km)	5.4
Flight Path Angle (deg)	**±0.2**

Integration of inertial measurements does provide highly accurate state vectors as long as the systems can be properly updated. As indicated in table 2, all successful planetary entry probes have met or exceeded their required uncertainty in flight path angle. It is shown that the proposed Titan mission and Venus mission will exceed their requirements using updated doppler and range measurement only.

Table 2 Required Entry Uncertainty

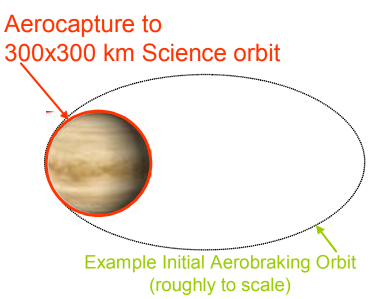
Mission	Entry FPA	Delivery Error	Delivery Time	Reqm't
Venus Aerocapture*	-6.12°	±0.4°	E-2 d	<±0.5 >
Titan Explorer*	-36.8°	±0.6°	E-2 d	<±1.0>
Mars Pathfinder	-14.2°	±0.4°	E-2 d	±1.0
MPL	-12.0°	±1.0°	E-2 d	~±0.5
MER	-11.5°	±0.2°	E-2 d	±0.25
Stardust	-8.2°	~±0.8°	E-2 d	±0.80
Galileo probe	-8.6°	±0.6°	E-140 d	±1.4
Huygens probe	-64.0°	±3.0°	E-21 d	±3.4

* proposed aerocapture mission
<-> indicates proposed requirement

Mission Analysis and Spacecraft Concept

Basic mission parameters, already presented, provide an aerocapture 300x300 km polar orbit based on a 1090 kg spacecraft with a ballistic coefficient of 138 kg/m^2. Orbit circularization requires a 99 m/sec increase in velocity at apoapsis after some 4300 m/sec of velocity is removed during the single aerocapture pass.

This contrasts with an alternative aerobraking mission, figure 4, which requires some 122 days and 670 orbits before the final 300x300 km science orbit can be achieved. After removing 2300 m/sec by retrorocket for initial orbit insertion, another 2011 m/sec would be removed through 670 aerobraking orbits reducing the orbital period from 4.4 to 1.6 hours.



- Aerobrake Mission
  - Orbit Insertion dV: 2300 m/s
  - A/B Duration: 122 days
  - A/B orbits: 670
  - Accumulated A/B dV: 2011 m/s
  - Initial orbit period: 4.4 hr
  - Final orbit period: 1.6 hr

Figure 4 Alternative Aerobraking Mission

There are no "spacecraft design" issues for the Venus study as the same shape and general packaging approach that was used for the Titan were used for this analysis. In addition the successful aeroshell plus cruise stage used for the Mars Exploration Rover missions give high confidence that the proposed Venus Aerocapture Discovery Mission spacecraft is reasonable. Initial spacecraft mass and power estimates are shown in table 3 and indicate that to deliver 65 kg of science instruments to Venus will require that some 1165 kg of spacecraft must be launched.

Table 3 Spacecraft Mass and Power

Alloc: Allocation
Cont: Contingency
CBE: Current Best Estimate
MEV: Max Expected Value
Cont = (MEV-CBE)/CBE
Margin = (Alloc-MEV)/MEV

		#	CBE	Cont	MEV	Alloc		JPL Margin
Launch Vehicle Capability						1165		
Propellant and Pressurant					**89.6**			
Hydrazine + Helium					**89.6**			
Cruise					27.3	50	m/s	
Aerocapture					50.9	99.2	m/s	
Orbit					8.8	10	m/s	
Residual & Pressurant Tank					2.6		**Margin**	
Launch Dry Mass			**680.0**	**22.2%**	**831.2**	**1075.4**	**29.4%**	36.8%
Aerocapture System Dry Mass			**242.7**	**20.0%**	**291.3**	**349.5**	**20.0%**	30.6%
Spacecraft Dry Mass			**437.3**	**23.5%**	**540.0**	**725.9**	**34.4%**	39.8%
Instruments			**50.0**	**30.0%**	**65.0**			
Bus		121	**387.3**	**22.6%**	**475.0**			
Attitude Control		8	19.5	2.2%	19.9			
Command & Data		26	37.3	17.3%	43.8			
Power		5	46.5	17.0%	54.4			
Hydrazine Propulsion System		39	32.4	6.9%	34.6			
Structures & Mechan		1	140.0	30.0%	182.0			
Harness	31.00	1	31.0	30.0%	40.3			
X-Band Telecomm		40	20.6	6.5%	22.0			
Thermal		1	60.0	30.0%	78.0			

	Phase	Cruise	Aerocap	Science
Solar Range	AU	1.000		0.723
Battery Capability	**A-h**		**35**	
Array Capability (EOL)	**W**	**505**		**635**
Instruments	W	0	0	100
Attitude Control	W	26	31	31
Command & Data	W	35	45	45
Propulsion	W	0	46	0
Power	W	15	15	15
Telecomm	W	187	17	187
Thermal	W	60	0	30
28V CBE Loads	**W**	**323**	**154**	**408**
28V Load Contingency			**30%**	
28V MEV Loads	**W**	**420**	**200**	**530**
28V Load Margin		**20%**	**20%**	**20%**
Load	**A**		**8.6**	
Battery Life	**hr**		**4.1**	
Max Sun offpoint Time	**hr**		**1.0**	
Depth of Discharge	**hr**		**25%**	
JPL Margin		36%	39%	36%

Analysis of methods of orbital insertion shown in table 4, ranging from chemical propulsion with and without aerobraking to single pass aerocapture, indicate that significantly greater mass can be delivered through the use of aerocapture because more propulsion system and propellant mass is required than aeroshell thermal protection system, TPS, mass to produce the required final orbit.

Table 4 Comparison of Aerobraking, Aerocapture, and Chemical Propulsion Missions

User Input	**CBE**	**Contingency %**	**Contingency Total**	**Margin**	**Allocation**
LV Capability (Delta 2925H-10, C3=8.3)					**1165**
CHEMICAL WITH AEROBRAKING					
Cruise Stage Isp	**326.0**				
Cruise Stage dV (Aerobrake 300x23000km)	**2350.0**				
Cruise Stage Propellant	659.9	**3.0%**	679.7	**0.0%**	679.7
Propulsion Dry Mass	**101.0**	**10.0%**	111.1	**0.0%**	111.1
Venus Orbit Delivered Capability (kg)					**374.2**
CHEMICAL WITHOUT AEROBRAKING					
Cruise Stage Isp	**326.0**				
Cruise Stage dV (**300x300 km**)	**4300.0**				
Cruise Stage Propellant	914.9	**3.0%**	942.4	**0.0%**	942.4
Propulsion Dry Mass	**101.0**	**10.0%**	111.1	**0.0%**	111.1
Venus Orbit Delivered Capability (kg)					**111.5**
CHEMICAL WITHOUT AEROBRAKING					
Cruise Stage Isp	**326.0**				
Cruise Stage dV (**300x8500km**)	**3000.0**				
Cruise Stage Propellant	763.1	**3.0%**	785.9	**0.0%**	785.9
Propulsion Dry Mass	**101.0**	**10.0%**	111.1	**0.0%**	111.1
Venus Orbit Delivered Capability (kg)					**268.0**
AEROCAPTURE					
Cruise Propellant					27.3
Pre-aerocapture separated mass			**50.0**		50.0
Aerocapture Entry Mass					**1087.7**
Venus Aeroshell Mass (30% of launch mass)					349.5
Aerocapture propellant					50.9
Venus Orbit Delivered Capability (kg)					**687.3**

These results compare well with a recent aerocapture cost/benefit analysis: **Jeffery L. Hall, Muriel A. Noca and Robert W. Bailey (2003). "Cost-Benefit Analysis of the Aerocapture Mission Set", AIAA Paper 2003-4658, presented at the 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Huntsville, AB, July 20-23, 2003.**

Atmospheric Model

The Venus International Reference Atmosphere (VIRA)[5] provides dependence of the Venus-GRAM mean atmospheric parameters on height, latitude and time-of-day or solar zenith angle. Basically, the VIRA characteristics are functions of altitude with low (0-100km) being a function of latitude only, middle (100-150 km) being a function of time-of-day only and high (150-250 km) being a function of solar zenith angle only. An example of density versus altitude is given in figure 5. From this information it can be seen that Venus has the most rapid changes in density with altitude of any of the planets which equates to very low scale heights, HS. For example, Mars, HS = 6.4-8.2 km, Earth, HS=8.4 km and Venus, HS= 3.5-4.7 km. This means that small changes in altitude can result in large changes in atmospheric density which, with all else being equal, will result in very narrow entry corridor widths.

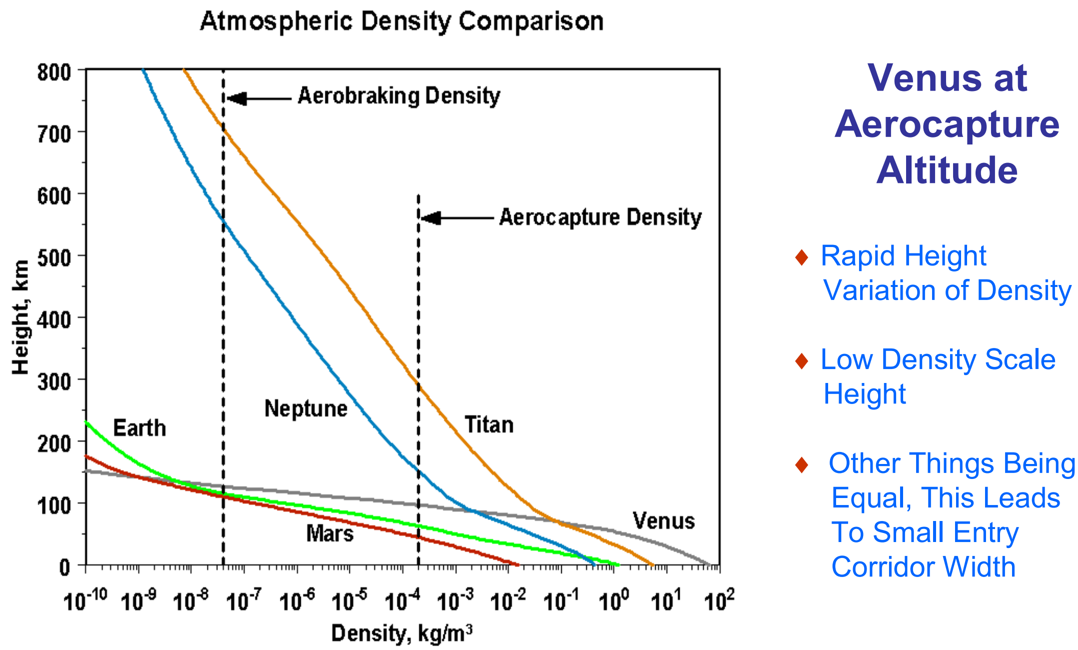


Figure 5 Comparison of Density Versus Altitude for Several Planets and Satellites

The information for the VIRA is based on some 23 successful spacecraft, both orbiters and entry probes, that have been sent to Venus.

Aerocapture Guidance

The guidance system proposed for a Venus aerocapture mission is the Hybrid Predictor-corrector Aerocapture Scheme (HYPAS) which was developed in about 1989 for the now cancelled Aeroassist Flight Experiment (AFE). HYPAS was tested, compared and evaluated against other guidance algorithms in both three- and six-degree-of-freedom computer based simulation and selected for the AFE flight. Development of the actual flight code was on schedule when AFE was cancelled. Since then HYPAS has been used in numerous human and robotic mission studies performed at the Johnson Space Center for both Earth and Mars. HYPAS has thus been evaluated over a wide range of lift-to-drag ratios, ballistic coefficients, atmospheres, entry conditions and target orbits. It was the leading contender for the guidance system for the 2000 Mars Surveyor Program until aerocapture was eliminated from the mission plan. Likewise, it was considered for the CNES Mars 2005 Sample Return Orbiter and the 2007

Premier Missions but once again aerocapture was eliminated from the plans because it was not a proven technology. Prior to this study,[6] it was the basis for the Titan[7] and Neptune[8] aerocapture mission analyses and significant improvements in performance and robustness have been achieved in the last 10 years.

The basic HYPAS algorithm guides a lifting vehicle through an atmosphere to a target exit orbit apoapsis and inclination. Given a set of initialization constants (L/D, Ballistic coefficient, atmospheric characteristics, etc.) and the target exit orbit HYPAS starts with the vehicle updated initial state and through sensed acceleration computes position and velocity. This is compared with values required to achieve the desired exit orbit and HYPAS commands the vehicle to bank to roll the lift vector in the proper direction to fly the spacecraft on the targeted trajectory through the atmosphere.

Specifically, HYPAS consists of two phases during the aerocapture pass shown in figure 6. The capture phase is intended to remove sufficient velocity to ensure vehicle capture. Bank angle commands are given to stabilize the trajectory and drive the vehicle toward equilibrium glide conditions with a desired vertical lift. The exit phase sets the vehicle up for the correct exit velocity to ensure the proper capture orbit apoapsis. By analytically predicting the velocity vector at atmospheric exit, at a constant altitude rate, bank angle is commanded such that the required velocity vector is achieved that will produce the correct orbit. The lift vector is adjusted through bank angle to null the error between the predicted and the target apoapsis. In addition, lateral logic can command a "roll reversal" maneuver as required to change the vehicle heading without affecting aerocapture to ensure that the final orbital inclination is achieved.

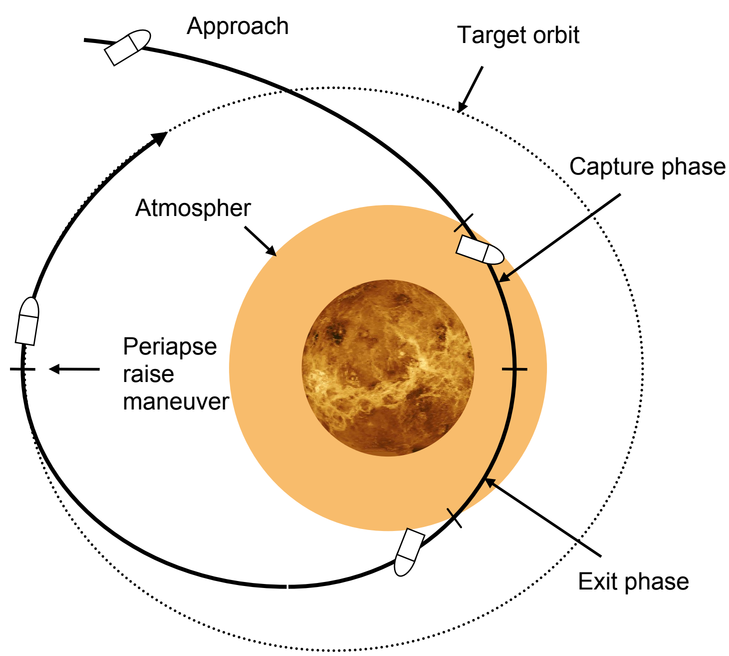


Figure 6 HYPAS Aerocapture Phases

The core of HYPAS is an analytically derived control algorithm based on deceleration due to drag and an altitude rate error feedback.

$$\left(\frac{L}{D}\right)\cos\phi_{cmd} = \frac{C_L}{C_D}\left[\cos\phi_{eq.gl.} - G_{\dot{h}}\left(\frac{\dot{h}-\dot{h}_{ref}}{\bar{q}}\right) + G_D\left(\frac{D-D_{ref}}{\bar{q}}\right)\right]$$

All references are computed and updated during flight. This analytic, non-iterative, on-the-fly approach leads to efficient code (320 source lines of FORTRAN code), minimal storage requirements and fast, consistent execution times.

To develop the final trajectory requires several steps. The theoretical maximum entry corridor is determined by examining the full-lift-up and full-lift-down trajectories which optimum guidance performance can maintain and yield a captured exit orbit. Sensitivity analysis is performed by varying density, flight path angle and C_L/C_D to determine what trajectories are actually captured by the guidance system resulting in a nominal guided trajectory through the entry corridor. Monte Carlo simulations are then conducted to determine guidance robustness by varying all parameters to +/- 3-sigma in a random sense to determine guidance failure rates. Guidance parameters can be fine tuned to reduce/eliminate failures and the Monte Carlo simulations can be re-run. This determines the final trajectory through the atmosphere which the day-of-flight guidance, navigation, and control will attempt to follow to aerocapture into the desired final orbit.

In developing the actual aerocapture trajectory the following initial assumptions were made: vehicle mass=900 kg, reference area=5.515 m^2, C_D=1.43, C_L=0.3575, C_L/C_D=0.25, entry velocity=11.25 km/sec, azimuth=0 deg, altitude=150 km, time, 7 April 2004, 12:00:00:00 VEN, target orbit, 300 km circular polar. Analysis shown in figure 7 indicates that for a 300 km orbit the entry flight path will be -6.12 degrees with a corridor width of +/- 0.76 degrees.

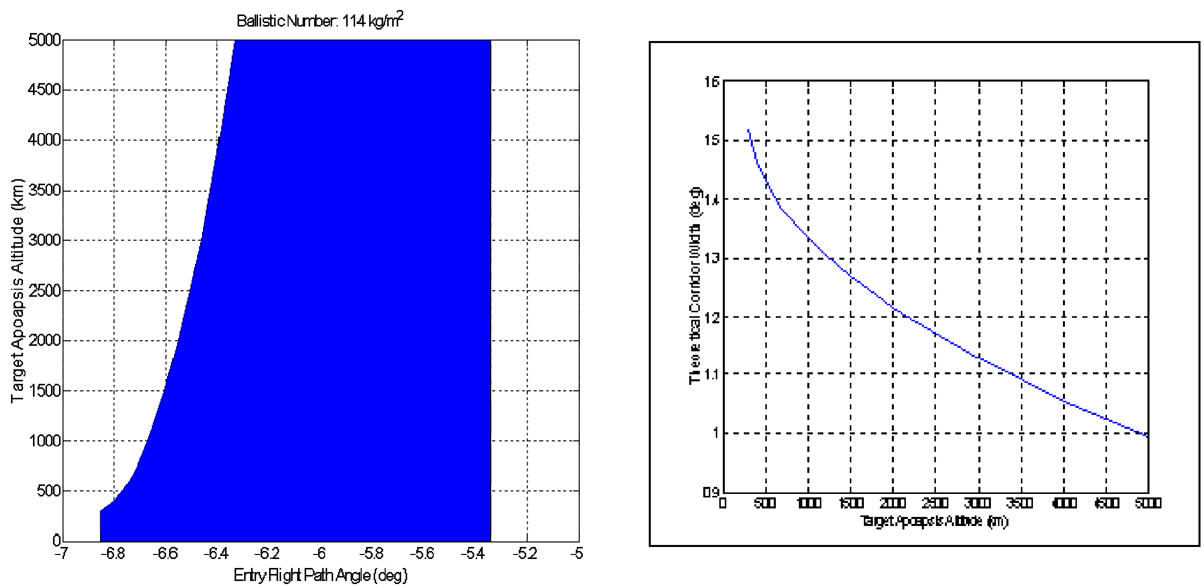


Figure 7 Aerocapture Corridor Width Versus Target Apoapsis

This rather narrow corridor width is caused by the small scale height mentioned previously. Very minor variations in altitude yield large variations in density which significantly affect the vehicle performance. This is exacerbated by the relatively low roll rate available to this class of vehicles which limits the ability of the spacecraft to respond to changes in performance by reorienting the lift vector to fly up, down, right or left as required by guidance. This can lead to conditions where the spacecraft enters the exit phase of the aerocapture pass before commanded roll maneuvers have been completed. Solutions are to provide increased roll control authority and/or force the algorithm to allow time for the completion of desired maneuvers.

A nominal aerocapture profile has been developed. Initial assessments were conducted using SORT with the final analysis performed directly in POST. The guidance has been tuned for roll rates of 20 and 30 deg/sec along with ballistic coefficients of 114 and 228 kg/m^2 ensuring that the final vehicle will fall somewhere between those values. The low atmospheric scale height does present some challenges and while the current guidance algorithm captures 96-percent of the theoretical entry corridor, it may be desirable to push the roll rate to the 30 deg/sec value.

Monte Carlo Simulations

A complete simulation is modeled in POST. Beginning with final navigation and state vectors at 60 seconds prior to atmospheric interface, defined as 150km altitude, the simulation flies an initial -6.12 degree flight path through the capture phase, exit phase to initial apoapsis and the periapsis raise maneuver and finally the orbit circularization maneuver at what would have been the periapsis. The models required for the POST simulation are the initial state provided by the JPL navigation model, the Venus atmosphere model, 70 degree sphere cone trimmed aerodynamic and aeroheating databases, HYPAS guidance and the bank angle controller.

The simulation assumes a 70 degree sphere cone with $C_L/C_D = 0.25$. The Venus atmosphere model, discussed earlier, assumes random high frequency perturbations to approximate uncertainties in the atmosphere. HYPAS guidance is tuned for the nominal entry with a target apoapsis of 300 km. The vehicle is trimmed using nominal aerodynamics (C_N=+/- 3-percent and C_D=+/- 5-percent) with uncertainties in center of gravity location (X_{CG}/L=+/- 0.012-percent and Z_{CG}/D=+/-0.026-percent) applied and variations in trim angle of attack (α=+/-2.0deg) simulating uncertainties in pitching moment. All variations applied are to 3-sigma in a normal distribution. The bank angle controller had a maximum roll rate of 30 deg/sec and a maximum roll acceleration of 5 deg/sec^2.

As indicated in figure 8, 2001 test cases for roll rates of 20 deg/sec and 30 deg/sec were captured with a relatively small variation in initial apoapsis. While the variation in results is greater for the 20 deg/sec cases, all of the initial periapsis projections are all within the on board delta-V capability of the vehicle to perform the periapsis raise maneuver, thus all cases are considered to be successfully captured.

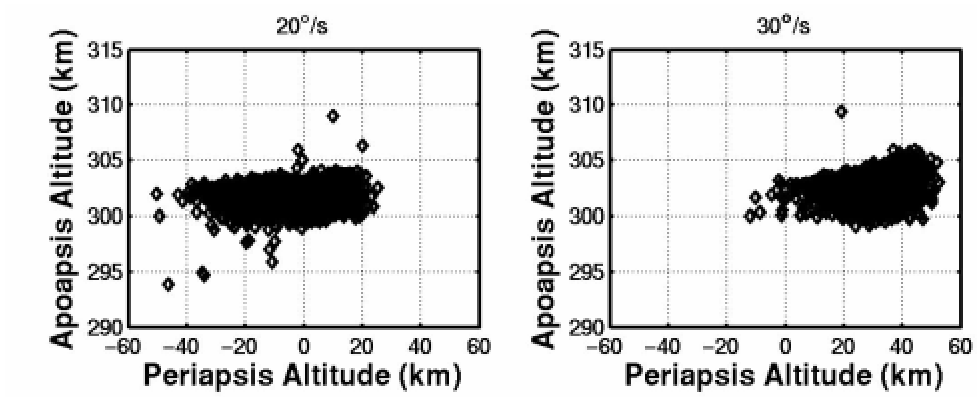


Figure 8 Comparison of Apoapsis vs. Periapsis Dispersions, Decreased Max Bank Rate vs. Reference Max Bank Rate

Given that the aerocapture successfully placed the spacecraft into the proper 300km circular, polar orbit, station keeping requirements to maintain the orbit for a two year mission were examined. Various orbits from 200 to

300km were analyzed to determine the total on-board delta-V in m/sec required to maintain the orbit. The proposed spacecraft will be similar to the Magellan spacecraft and the following assumptions are based on that spacecraft: 18.1 kg/m^2 ballistic coefficient, 630 kg on orbit mass, 17.41 m^2 frontal area continuously normal to the molecular flowfield with a C_D=2.0.

Even at 300 km altitude there is enough residual atmospheric drag to slowly degrade the orbit. As shown in figure 9, the orbit would be allowed to drop to 295 km at which time the spacecraft would re-boost itself back to 300 km. At the end of the two year mission the orbit would be allowed to reach 295 km.

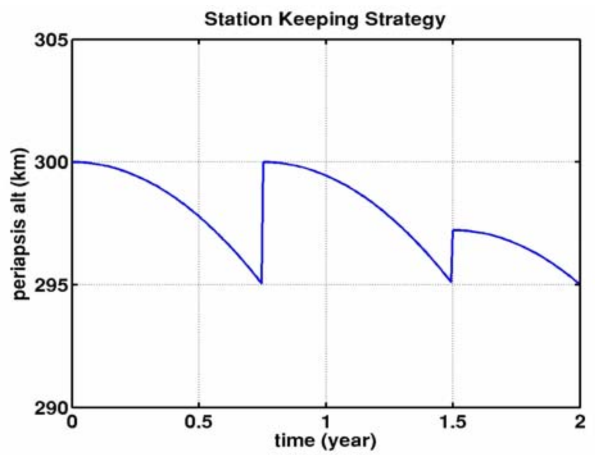


Figure 9 Station Keeping Strategy

The total delta-V required to re-boost the spacecraft for orbits from 200 to 350 km is shown in figure 10. Clearly the lower the orbit, the higher the drag and the more re-boosting that would be required. At the planned 300 km orbit this amounts to only about 1 m/sec total delta-V over the two year mission.

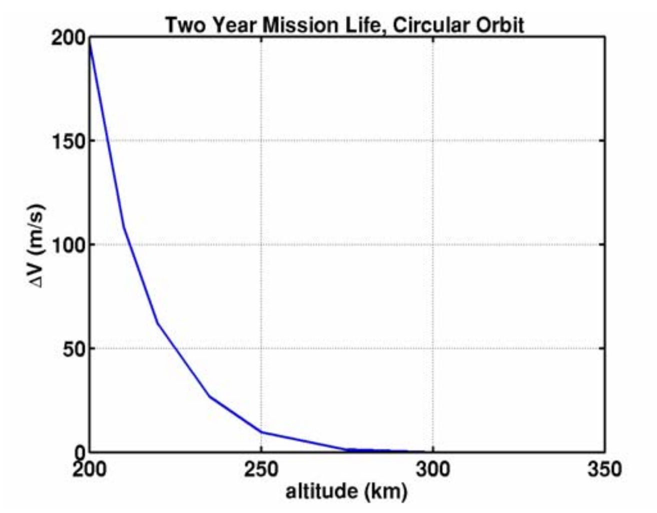
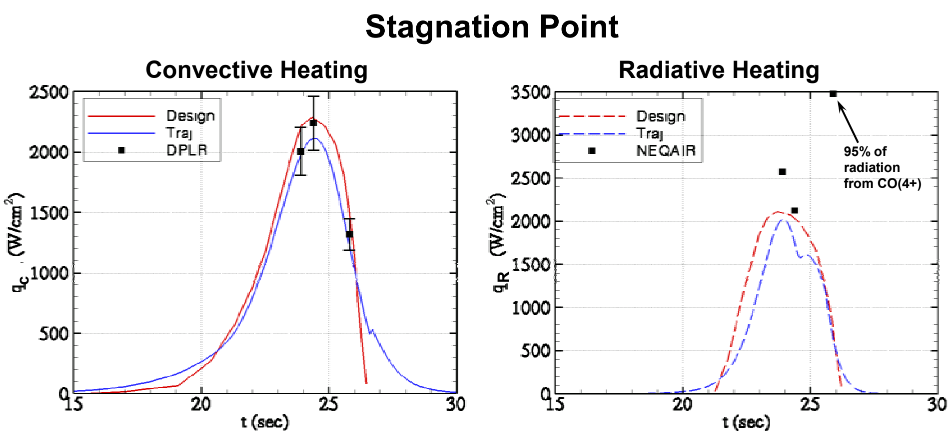


Figure 10 Delta-V Required for Station Keeping on a Two Year Mission

Aeroheating and TPS Selection and Sizing

In aerocapture, when the spacecraft passes deep enough into the atmosphere to remove enough velocity for orbit capture, it will undergo significant aeroheating. Since approach hyperbolic velocities are high compared to orbital velocities, heating during the atmospheric entry can well be the most significant design parameter. As such, attention to the thermal protection system can be of extreme importance. Since the objective of the initial Venus Study was performance, only a quick look at aeroheating environments was undertaken. Flowfield calculations were performed at LaRC with the LAURA code and at ARC with the DPLR code. These codes are similar in approaches and capabilities. Modeling assumptions for Venus entry included thermal and chemical non-equilibrium, radiative equilibrium wall boundary conditions with emissivity of 0.9, super catalytic wall boundary layer condition, laminar and turbulent (Balwin-Lomax) boundary layer and multi-component diffusion model with a 15 specie Venus kinetic model[9]. In addition, heating due to shock-layer radiation is computed in an uncoupled manner using NEQAIR at ARC and RADEQUIL at LaRC. The effects of coupling between the radiation and the flowfield are accounted for via engineering relations[10]. Radiation effects include both line and continuum radiation and assume a Boltzmann distribution of excited states.

Comparisons of computed results with the design and flight results for the Pioneer Venus (PV) spacecraft are shown in figure 11. Understanding this comparison is essential to calibrate predicted results to quantify the uncertainty and risk in the current design. Convective heating rates are in good agreement among design, flight and DPLR results. The current NEQAIR results are higher than the PV design or flight radiative heating.



- ♦ **Design Data** : **scanned from PV CDR report**
- ♦ **Traj Results** : **3DoF reconstruction of design trajectory (engineering estimate)**
- ♦ **DPLR Results** : **fully catalytic ("error" bars show +/- 10% modeling variability only)**
- ♦ **NEQAIR Results** : **tangent slab corr. = 0.75; non-adiabatic corr. ~ 0.75**

Figure 11 Comparisons of Predictions and Pioneer Venus Flight Heating Results

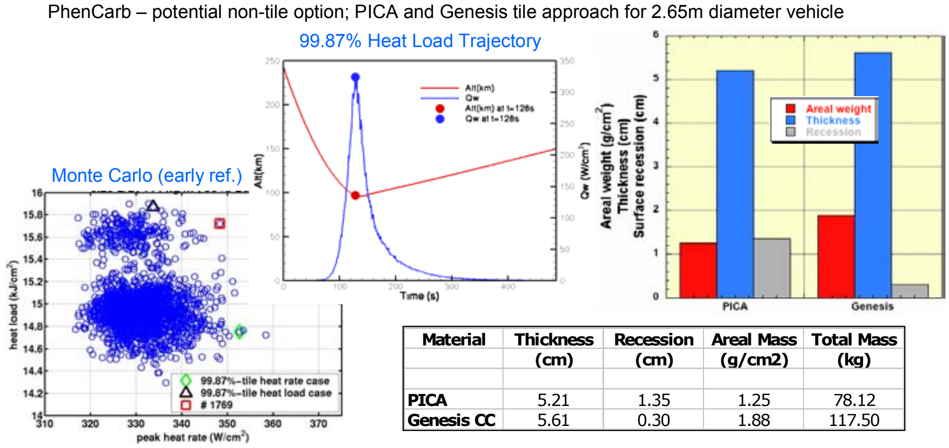
Tangent slab corrections, emission of CO and CN, as well as possible non-Boltzmann distribution of excited states must be examined to improve predictions. In addition there are data timing issues with the PV flight results which make data analysis difficult. The codes predict different levels of turbulent heating as well as radiative heating, and these differences must be examined as well.

The CFD codes employed (LAURA and DPLR) are in reasonable agreement for laminar flows, but predict different levels of turbulent heating. The radiation codes (NEQAIR and RADEQUIL) also differ significantly in their baseline heating predictions in this environment. However, the short duration of the system study did not allow for these differences to be investigated in any detail. These code-to-code differences must be examined to reduce the uncertainty in the aeroheating environment and associated TPS sizing estimates.

That said, estimates of heating for the aerocapture mission were made along the design trajectories discussed earlier and are shown in Figure 12. Peak zero-margin turbulent conductive heating rates were on the order of 500-600W/cm^2, and peak radiative heating rates were about 500-700W/cm^2. Net heat loads on the overshoot trajectory were about 55kJ/cm^2. No analysis of afterbody heating was performed during this study.

Based on these results a TPS selection was undertaken. The very large heat loads make fully dense carbon phenolic (the material on Pioneer Venus) unattractive since at these conditions it is an inefficient ablator leading to large system mass. Two other candidates with flight heritage exist that could likely handle this environment; PICA, which is flying on Stardust and carbon-carbon multi-layer, which was flown on Genesis. TPS sizings were carried out for each of there materials and the results show that PICA provides a somewhat lower (33%) mass of 78kg. However, it is likely that both of these materials would need to be fabricated as tiles for a vehicle of this size. No tile system has ever been subjected to the heating levels predicted for this mission and any forebody tiles would require robust gap filler and seal materials. In contrast, materials like the Applied Research Associates (ARA) PhenCarb family, if fabricated by the "Strip Collar Bonding Approach" (SCBA), would yield a continuous forebody TPS. At the least these results indicate that serious TPS development and qualification will be needed, as a strongly enhancing technology, for this mission.

- Initial convective and radiative aeroheating results computed with LAURA/RADEQUAL and DPLR/NEQAIR at pk heating pt on 99.87% pk heat load M.C. trajectory; highest heating location on vehicle for radiative and convective; coupling estimate included; no additional uncertainty included
- Results utilized to scale stagnation convective aeroheating indicator for initial aeroheating loads estimate
- TPS analysis completed for 2 materials, PICA and Genesis CC
- Future work: aeroheating methods- reduce uncertainty in mass and shape change; TPS sizing of ARA PhenCarb – potential non-tile option; PICA and Genesis tile approach for 2.65m diameter vehicle



Material	Thickness (cm)	Recession (cm)	Areal Mass (g/cm2)	Total Mass (kg)
PICA	5.21	1.35	1.25	78.12
Genesis CC	5.61	0.30	1.88	117.50

Figure 12 Initial Aeroheating and TPS Sizing

Conclusions

This systems analysis study of a polar orbit mission to Venus has demonstrated that aerocapture, from a performance perspective, is feasible and robust with 100-percent of the analysis cases resulting in useful captured science orbits. Top level analysis indicates that aerocapture will provide significant mass margins for launches on Delta 2925H and may allow launch opportunities for the standard Delta 2925. Performance sensitivities to both ballistic coefficient and vehicle roll rate were explored.

Based on this study there are no enabling technologies required to accomplish this mission. There are, however, strongly enhancing technologies that will significantly help such missions. Aeroheating analysis tools which will improve calculations for mass and shape changes caused by TPS ablation providing improved estimation and analysis of uncertainties. If TPS were to use PICA , then manufacturing, testing and flight certification of PICA and Genesis materials will be required. If ARA PhenCarb TPS were to be used, then a strip collar bonding approach to provide continuous forebody TPS must be developed and flight certified. Improvements in the guidance system to further vehicle performance robustness to accommodate the small atmospheric scale height at Venus was identified as an enhancing technology.

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14. ABSTRACT
Previous high level analysis has indicated that significant mass savings may be possible for planetary science missions if aerocapture is employed to place a spacecraft in orbit. In 2001 the In-Space Propulsion program identified aerocapture as one of the top three propulsion technologies for planetary exploration but that higher fidelity analysis was required to verify the favorable results and to determine if any supporting technology gaps exist that would enable or enhance aerocapture missions. A series of three studies has been conducted to assess, from an overall system point of view, the merit of using aerocapture at Titan, Neptune and Venus. These were chosen as representative of a moon with an atmosphere, an outer giant gas planet and an inner planet. The Venus mission, based on desirable science from plans for Solar System Exploration and Principal Investigator proposals, to place a spacecraft in a 300km polar orbit was examined and the details of the study are presented in this paper.

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