HEAT-EXCHANGE DRIVEN AIRCRAFT FOR LOW ALTITUDE AND SURFACE EXPLORATION OF VENUS: PHASE 1 STUDY FINAL REPORT

(L. Lemke, D. Hall, E. Noe Dobrea, A. Brecht)

SCIENCE RATIONALE AND INSTRUMENT SELECTION

Eldar Noe Dobrea, Amanda Brecht

Exploration of Venus below its clouds has long been impaired by the planet's massive CO2 atmosphere, corrosive trace gases, and high opacity. Except for a few narrow spectral windows, the 20 km thick global cloud cover blocks light over the entirety of the ultraviolet through thermal infrared spectrum, impeding orbital studies of the surface and lower portion of the atmosphere. In addition to the cloud opacity, Rayleigh scattering through 80 bars of CO_2 in the lower 40 km of the atmosphere results in a Rayleigh scattering depth between 25 and 1 in the visible through near infrared portion of the spectrum. That optically thick scattering, combined with the uniform angular distribution of the illumination provided by the highly diffusing clouds overhead greatly reduces even very large surface albedo contrasts (Crisp, *pers. Comm.*). To date, the surface has only been mapped via synthetic aperture radar by the Magellan mission at scales of 120 to 300 m/pixel. In addition to contributing to the difficulty of performing studies of the surface from orbit, the extreme temperatures and pressures at the surface also create a challenging environment to utilize a lander for exploration. The longest-lived lander on Venus' surface survived for 2 hours. Hence, limited access to observe the surface and lower atmosphere of Venus arguably puts the surface and lower atmosphere of Venus as one of the least explored regions of the inner Solar System.

The Venus Exploration Analysis Group (VEXAG) publication "Goals, Objectives, and Investigations for Venus Exploration" provides a science traceability matrix describing the investigations necessary to address the gaps in our understanding of Venus (VEXAG, 2019). Prolonged flight beneath the cloud deck is seen as one of the more feasible methods of addressing key questions identified by VEXAG [1]. Prolonged flight in the atmosphere of Venus would enable an investigation to obtain key measurements identified by the Venus Exploration Analysis Group (VEXAG) including: accurate isotopic measurements; characterization of the composition, abundance, variability, and distribution of key reactive phases and aerosols; vertical and latitudinal structure and dynamics of the atmosphere beneath the cloud deck; surface composition, geology, tectonics, thermophysical properties, and volcanism.

A potential pathfinder mission capable of prolonged flight in the lower Venusian atmosphere (altitudes of 5-30 km) and carrying about 25 kg of scientific payload would enable groundbreaking studies. Table 1, which is informed by the Venus Aerial Platforms Study Team's report [2], lists the proposed instrument suite.

Priority (1-2)	ITEM	MASS (kg)	POWER	PURPOSE
1	Pressure/Temp sensor	0.15	0.3 W	Atmospheric structure
1	UV / Vis / NIR Spectrometer	3.00	3 W	Atmospheric species; surface mineralogy
1	Tunable Diode Laser Spectrometer	6.00	35 W	Trace species; isotopic abundances
1	Infrasound Microphone	0.10	0.2 W	Geophysics
1	VNIR Camera	1.00	3 W	Surface temperature and morphology
1	Chemical Sensors	0.50		Select chemical species
1	Ultra-Stable Oscillator	1.50	4.5	Atmospheric dynamics
1	Lightning detector	0.20		Atmospheric EM processes
2	GC Mass Spectrometer	9.00	50 W	Atmospheric species inc. noble gases
2	Nephelometer	1.50	0.2 W	Aerosol properties
2	Net Flux Radiometer (or microwave radiometer)	2.00	7 W	Atmospheric energy balance
	Total mass priority 1	12.45		
	Total mass priority 2	12.50		
	Total mass	24.95		

Table 1. Low Altitude Venus Aircraft Reference Instrument Suite

ENGINEERING CHALLENGES

Lawrence G. Lemke, Dave Hall

The starting point for the engineering part of this study is the realization that the heat energy (ΔQ) stored in low atomic weight materials (e.g., Lithium metal) moving through the temperature difference (ΔT) between low altitude and high altitude (ΔZ) on Venus can be larger than the gravitational potential energy (ΔE) required to lift that material between the low altitude and high altitude at the same location. For example, the outside air ΔT between 0 and 30 km ≈ 235 K. The heat energy stored in metallic Lithium over ΔQ = 235 K ≈ 846 KJ/kg. The gravitational potential energy stored in metallic Lithium when it is raised from 0 to 30 km on Venus, ΔE ≈ 263 KJ/kg—approximately 0.31 the magnitude of ΔQ. The first study goal was to determine whether the ΔQ stored in a low atomic weight material moving through a given ΔT could be converted into useful work with an efficiency (ε) sufficient to power a winged aircraft to climb through the ΔZ corresponding to that ΔT. If $\varepsilon \geq 0.31$ then it is possible that such an aircraft could be engineered to cycle continuously between high and low altitude using only, or mostly, the thermal gradient built into Venus' atmosphere. If this approach proved to be not feasible, the second study goal was to determine whether adding buoyancy to the aircraft would change the result.

The engineering problem can be divided into two main parts, the energy/thermodynamics problem, and the aerodynamic problem. The energy/thermodynamics problem consists of designing a heat engine system capable of efficiently turning ΔQ into useful work (shaft power and/or electricity) to power and propel the aircraft. The aerodynamic problem consists of designing an aircraft capable of being transported into the Venusian atmosphere inside an atmospheric entry vehicle, self-deploying into stable aerodynamic flight after entry and descent, and surviving Venus' harsh atmosphere over multiple flight cycles. The primary figure of merit for the aerodynamic efficiency of an aircraft is its lift-to-drag ratio (L/D). L/D measures how much of the available energy goes into working against gravity (lift) compared to how much energy is simply dissipated in the atmosphere (drag).

We began this study by studying the energy/thermodynamic problem and the aerodynamic problem in parallel. We assumed that the energy would be supplied by some type of heat engine system obeying the Carnot Cycle limitations, and that the aerodynamic design would provide the best feasible L/D consistent with fitting the aircraft inside an atmospheric entry vehicle. As we proceeded, we periodically compared results of these parallel study efforts to see if they could be made to converge to one or more solutions.

Design Iterations

Because a Venus aircraft would need to be delivered to the Venusian atmosphere inside an atmospheric entry vehicle of some kind, we began by defining a reference entry vehicle. We were guided by a 2013 study out of NASA Ames Research Center titled: "Atmospheric Entry Studies for Venus Missions: 45^0 Sphere-Cone Rigid Aeroshells and Ballistic Entries". This study was in support of an NRC Planetary Science Decadal Survey activity which anticipated large, New Frontiers and/or Flagship missions to Venus. It found that entry vehicles employing the basic Pioneer Venus probe geometry could be extended to 4.5-meter diameter and larger and accommodate entry masses up to 2750 kg. At the time of the study, it appeared that the SpaceX Falcon 9 would be a reasonable candidate launch vehicle on which to base a mission concept. Because the Falcon 9 can accommodate a 5.2-meter diameter launch shroud, we assumed the use of a nominal 5 meter outside diameter entry vehicle employing the 45^0 Sphere-Cone geometry as shown in figure 1a. Figure 1a shows a cross-section with a 45° half angle on the nose cone and a hemispherical back shell, also converging to a parting line between the two at an initial 45° initial angle. Skin thicknesses are shown to scale. The green dashed shape inside the walls denotes the limits of internal volume available for a payload with distance between the payload and shell ("rattle space.") to accommodate launch and entry vibration as well as structure to hold the payload in place.

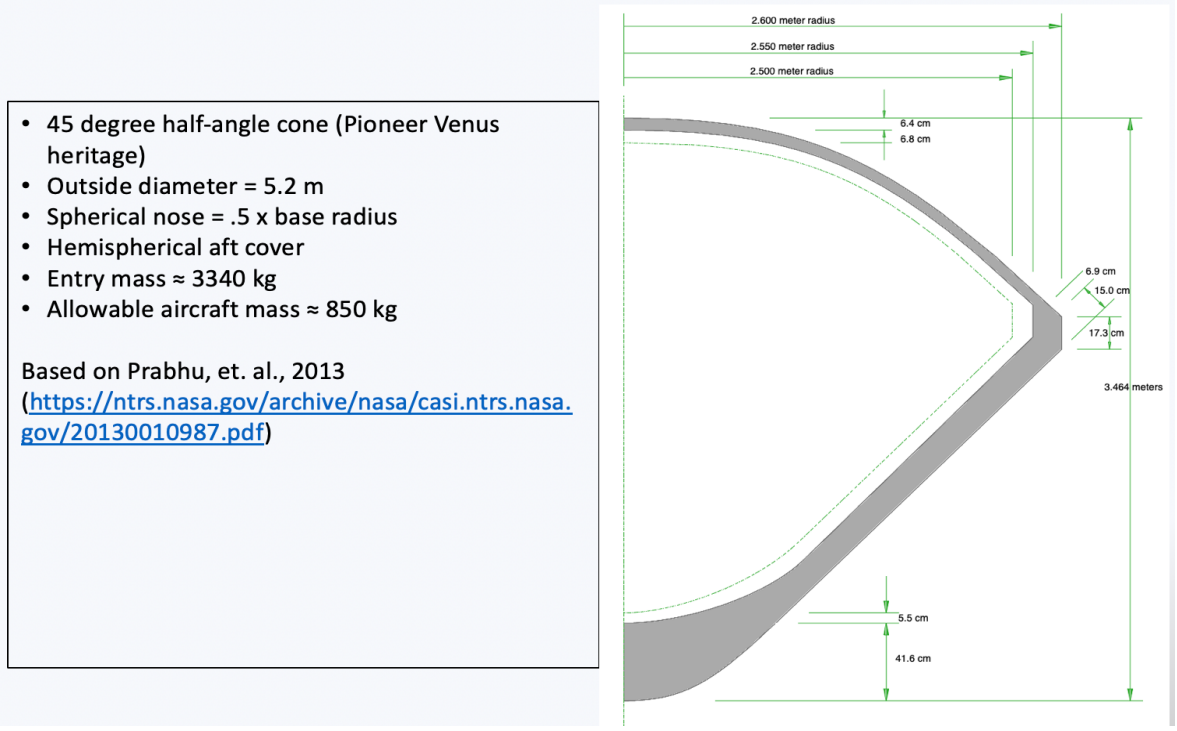


Figure 1a. Falcon 9-sized Aeroshell Geometry with Hemispherical Back Shell.

In addition, the aeroshell will have a canister attached to it that contains spin thrusters and

supersonic and subsonic parachutes for atmospheric entry. This entry canister is shown in yellow atop the Falcon 9 aeroshell in Figure 1b.

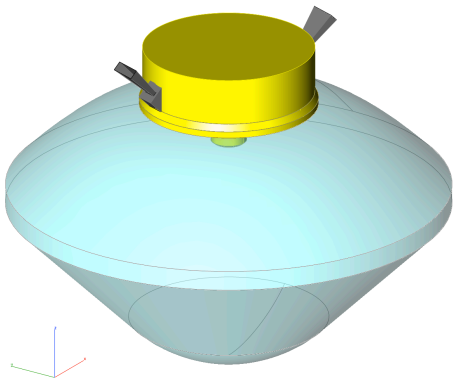


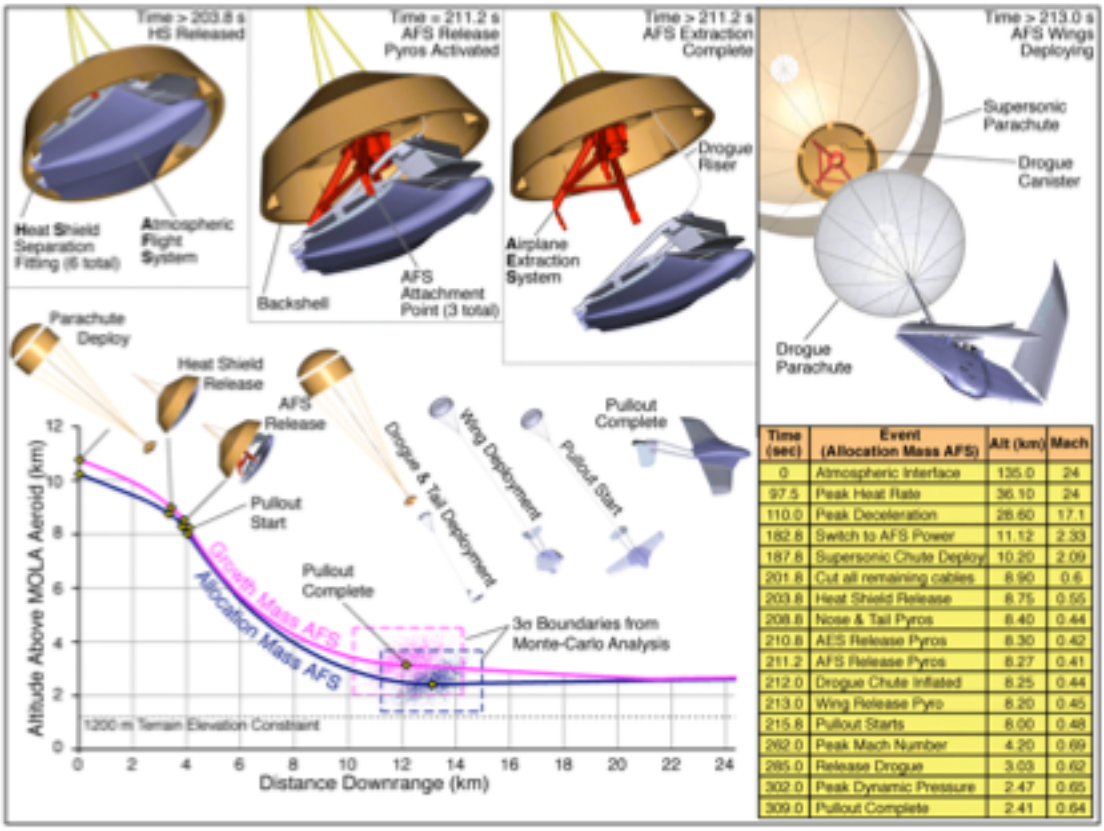
Figure 1b. Falcon 9 Aeroshell with Atmospheric Entry Cannister.

Low Altitude Venus Aircraft (LAVA) Configurations

The main effort in this study was directed at defining one or more candidate winged aircraft configurations capable of low altitude flight at Venus. One of the first design trades an engineer must face in designing an aircraft for deployment into the atmosphere of another planet is the strategy for folding the aircraft to fit into the entry vehicle and for unfolding it in the atmosphere once the entry vehicle has slowed to subsonic speed. In general, the more wing folds that are included in the design, the higher the aspect ratio of the wing and the higher the L/D of the aircraft. However, this comes at the cost of increased complexity and increased risk. This problem has received substantial study in the last ≈ 25 years—mostly in the context of Mars missions. Figure 2 illustrates one such wing folding and deployment approach that was adopted for the Aerial Regional-scale Environmental Survey (ARES) Mars Scout airplane mission. This mission was one of four finalists in the 2002 NASA Mars Scout mission opportunity and was ultimately rated as "selectable" by the NASA selection official, even though that mission opportunity was ultimately awarded to the Phoenix lander mission. The ARES mission adopted a conservative design using a single fold on each wing in part because this technology had been successfully demonstrated in a high-altitude balloon drop in 1999. Based on this history, we chose to adopt the same conservative approach of one-fold per wing for this study.

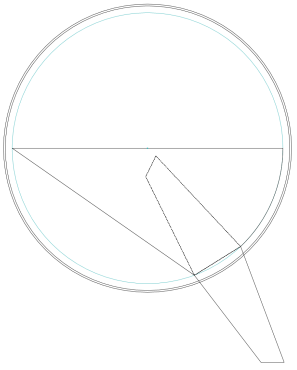
A blunt body aeroshell requires an aircraft configuration that allows as much volume as possible to be packaged near the aeroshell's maximum diameter. Considerable internal volume might be available for a large fuselage if its shape is tailored to the aeroshell geometry. Figure 3a (left) shows a candidate fuselage shape with a foldable wing, which is similar to blended wing body (BWB) aircraft being studied at Boeing, Airbus, and NASA. A planform view of Boeing's BWB 450 is shown in Figure 3b (right) for comparison. Enough test data have been collected in both wind tunnels and free flight to predict vehicle flight characteristics of these blended wing bodies.

The LAVA team also performed form, fit, and function studies of other shapes such as flying saucers and Burnelli lifting bodies before settling on a blended wing body. See Figure 4a (top) and Figure 4b (bottom). Neither shape provided an advantage over the BWB and, in both cases, added to parasite surface area which increased drag with no commensurate advantage with lift.



Time (sec)	Event (Allocation Mass AFS)	Alt (km)	Mach
0	Atmospheric Interface	135.0	24
97.5	Peak Heat Rate	36.10	24
110.0	Peak Deceleration	28.60	17.1
182.8	Switch to AFS Power	11.12	2.33
187.8	Supersonic Chute Deploy	10.20	2.09
201.8	Cut all remaining cables	8.90	0.6
203.8	Heat Shield Release	8.75	0.55
208.8	Nose & Tail Pyros	8.40	0.44
210.8	AES Release Pyros	8.30	0.42
211.2	AFS Release Pyros	8.27	0.41
212.0	Drogue Chute Inflated	8.25	0.44
213.0	Wing Release Pyro	8.20	0.45
215.8	Pullout Starts	8.00	0.48
262.0	Peak Mach Number	4.20	0.69
285.0	Release Drogue	3.03	0.62
302.0	Peak Dynamic Pressure	2.47	0.65
309.0	Pullout Complete	2.41	0.64

Figure 2. Aircraft Folding and Deployment Strategy for ARES Mars Airplane Mission. From: https://ntrs.nasa.gov/api/citations/20080015501/downloads/20080015501.pdf



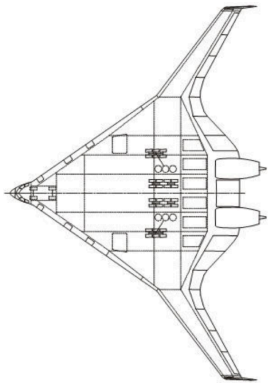


Figure 3a. Unconventional Fuselage Shape Aeroshell.

Figure 3b. Boeing BWB 450 Planform in

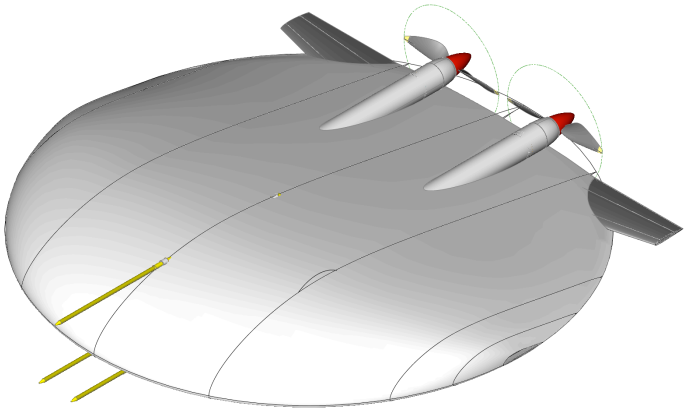


Figure 4a. Flying Saucer Configuration.

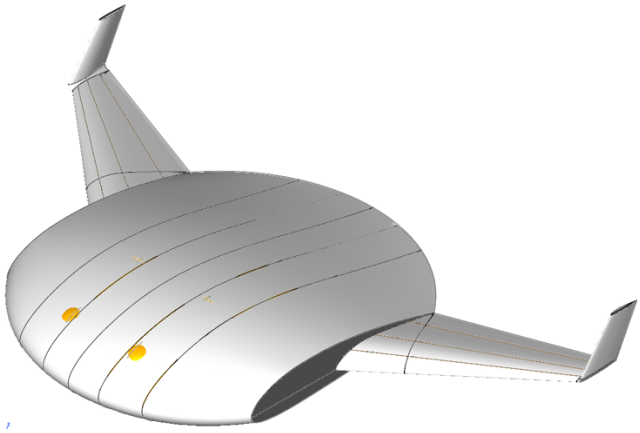


Figure 4b. Burnelli Center Lifting Body Configuration

The basic blended wing body (BWB) configuration scaled from Figure 3b included vertical surfaces (winglets) at the wingtips to make the original span-limited swept wing transonic 450 passenger airliner configuration more efficient. These winglets generate a horizontal force which is directed inward. The combination of this force and its trailing edge vortex sheet combine with the wing flow field to markedly improve wing efficiency. If a winglet were added to the winglet, such that the new surface generates a downward lift force like that found on conventional horizontal tails, then it can theoretically contribute to longitudinal stability as well as further reducing induced drag. Since these surfaces are small compared to a conventional horizontal and vertical tail and are located at the wingtips, they would be easy to stow in a blunt body aeroshell, as shown in Figure 5. This winglet modification is referred to as a C-Wing.

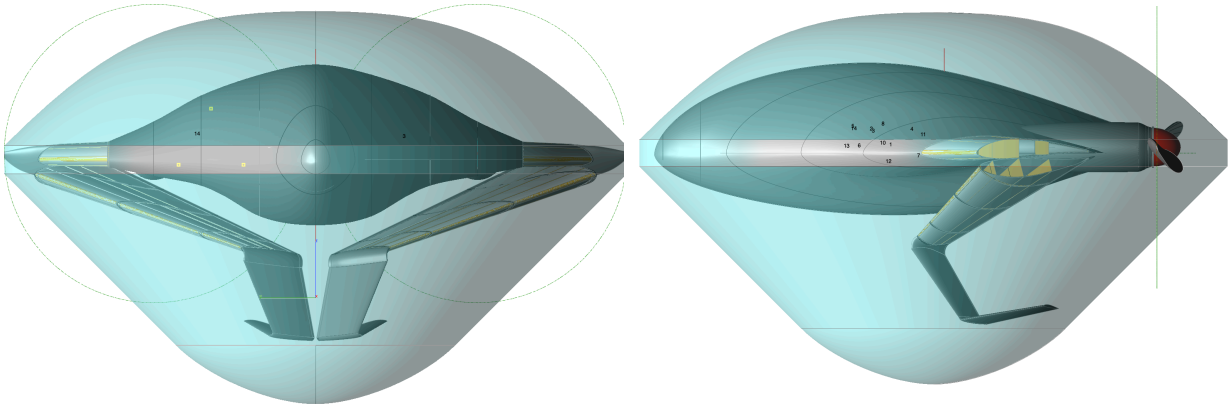


Figure 5. BWB C-Wing Stowed in Blunt Body Aeroshell Prior to Deployment; One Fold Per Wing.

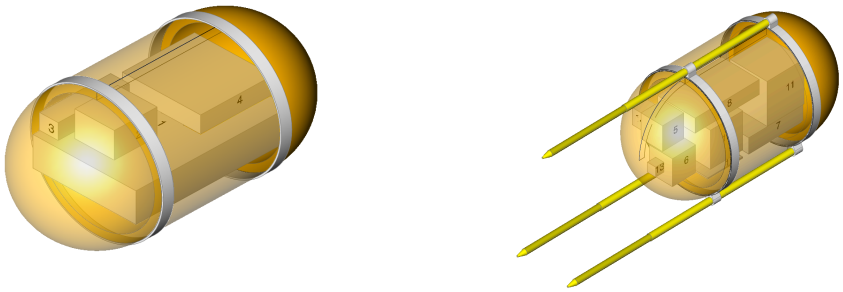
Missions and Payloads

Some engineering characteristics of the instrument suite discussed in the science section are summarized in Table 2.

	Item	Mass	length	width	height	power	
		kg	mm	mm	mm	W	
Fixed Equipment							
1	Integrated Avionics Unit	5.0	434.300	129.500	647.700	32	
2	Small Deep Space Transponder	3.2	305.000	330.000	73.700	20	
3	Traveling Wave Tube Amplifier (X band)	1.0	300.000	78.700	68.600	30-200	
4	Phased Array Antenna (X band)	5.5	305.000	330.000	73.700	20	
	Radar Altimeter						
	Avionics & Payload Pressure Vessel	29.4					
Payload							
5	Pressure/Temperature	0.2	22.890	22.890	22.890	0.30	sample port on skin (or probe in clean "air")
6	Nephelometer	0.1	23.510	23.510	23.510	0.20	oriented forward and down
7	Flux Radiometer	2.0	304.800	152.400	76.200	7	oriented up and down
8	Visual & Near Infrared Spectrometer	3.0	360.000	210.000	80.000	18-40	
9	Microwave Radiometer	3.0					
10	Tunable Laser Spectrometer	6.0	181.660	181.660	181.660	35	sample port on skin
11	GC Mass Spectrometer	9.0	207.950	207.950	207.950	50	sample port on skin
12	Radar Sounder	3.0	196.000	120.000	6.600	5-10	oriented down
13	Infrasound Microphone	0.1	46.410	46.410	46.410	0.20	any orientation
14	Near Infrared Camera	1.0	304.800	152.400	76.200	7	oriented up and down
	Chemical Sensors						
	Light Optical Atmospheric Counter						
	Imaging Microscope						
	Aerosol Mass Spectrometer						
	Ultra Stable Oscillator						
	Lightning Detector						
	Magnetometer						
	Electromagnetic Sounder						
	Gravimeter						

Table 2. Science Instrument Engineering Characteristics

The numbered components to be carried in avionics and payload pressure vessels are shown in Figures 6a (left) and 6b (right). The same avionics and payload suites are carried by all aircraft configurations we studied. Not shown in Figures 6a and 6b are shelving, electrical runs, and cooling systems required to serve the electronic components.



Figures 6a and 6b. Avionics (left) & Payload (right). Components in Thermally Controlled Pressure Vessels.

Mission assumptions focused on low altitude airborne flight with the aircraft cycling between ≤ 5 km above the surface and ≈ 30 km. The Venusian atmosphere is dense enough at low altitudes that fixed wing flight might be enhanced using buoyancy to offset vehicle mass and we ultimately included this option in our trade space. Aerodynamic design parameters such as lift and drag are both driven by dynamic pressure which is a product of atmospheric density and airspeed squared. The LAVA team initially looked at airborne fixed wing alternatives which would cycle between 0 and 30 km where the temperature declines from 735K at 0 km to 499K at 30km. Figure 7 is an inboard of the BWB C-Wing showing sizes and locations of powertrain-related components.

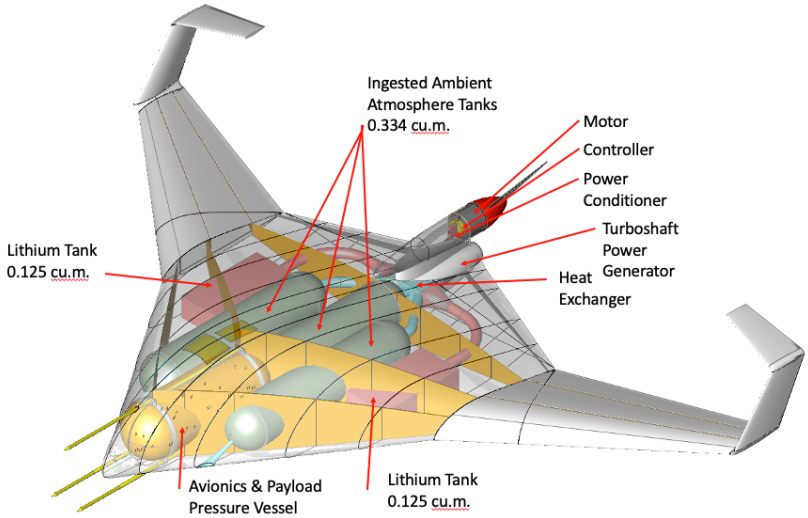


Figure 7. Heat Engine & Heat Sink Powertrain in BWB C-Wing

Because the low altitude Venusian atmosphere is extremely dense only very low flight speeds and wing areas are needed to generate lift. Figure 8 shows basic performance plots for the BWB

C-Wing with a hypothetical heat sink-driven heat engine and Table 3 identifies important point values. Data were run at 275 kg gross mass.

Significance of the curves:

- Light blue (lift-to-drag ratio) and dark red ($C_L^{3/2}/C_{DT}$) are indicators of aerodynamic efficiency;
- Yellow (thrust power required) is the power required to overcome drag;
- Purple (thrust power available) is the power generated by the propulsive system;
- Orange (climb rate) is an indication of how quickly an airplane could climb if all available power were used to change altitude;
- Dark Green (sink rate) shows how quickly an airplane would lose altitude if no climb power were available.

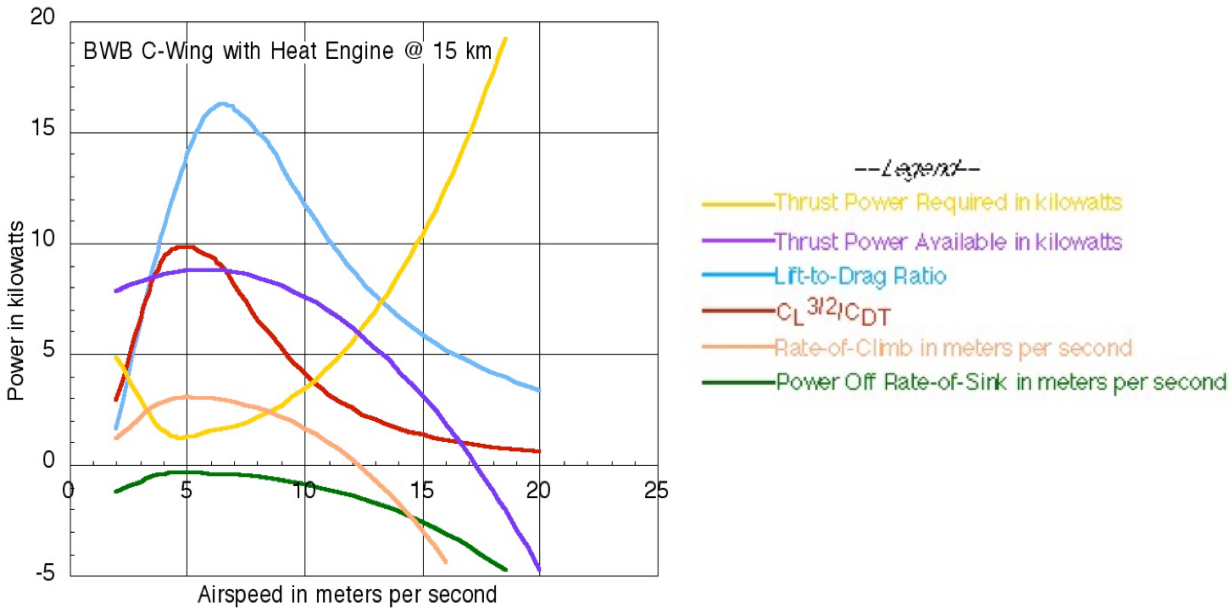


Figure 8. Basic Performance Curves for BWB C-Wing with a Heat Engine.

Maximum Speed	=	13.5	mps
Minimum Sink Rate Speed	=	5.0	mps
Minimum Sink Rate	=	40.1	mpm
Maximum Climb Rate	=	203.5	mpm
Maximum Climb Rate Speed	=	5.0	mps
L/Dmax	=	11.9 @ 9	mps
CL^1.5/CDTmax	=	8.8 @ 7	mps

Table 3. Performance Points of Interest for BWB C-Wing with Heat Engine at 15 km Altitude.

Low flight speeds at low altitudes also permit the use of thick airfoil sections with large internal volume that can accommodate bulky internal components such as avionics and payload pressure vessels and the powertrain. Internal volume is also available for powertrain components that could provide variable buoyancy.

Heat Engines

We began studying the thermodynamic problem by reviewing the state of technology for aerospace quality closed-cycle heat engines. Of note are Stirling cycle engines; small Stirling engines (≈ 100 W range) have been developed and flight qualified for deep space missions under NASA's Advanced Stirling Radioisotope Generator program. Larger versions (≈ 1000 W range) have been designed and tested for terrestrial applications. Figure 9 shows a 1 kW free-piston Stirling generator designed by Sunpower Inc. for terrestrial residential applications.



- Mass = 22 Kg
- Electrical power out = 1000 We
- Temperature ratio R = 2.7
- Carnot efficiency = .63
- Thermal/mechanical efficiency = .58 x Carnot efficiency
- Mechanical to electrical efficiency = .9
- Overall efficiency = .33

Figure 9. Representative kW-Class Free-Piston Stirling Generator

The theoretical maximum possible efficiency (the Carnot efficiency) of a closed-cycle heat engine is determined by the ratio, R, between the temperature of the hot gas and the cold gas ($R = T_h/T_c$). The efficiency is given mathematically by $\varepsilon = (1 - 1/R)$. For a Stirling engine operating on the heat supplied by a heat sink, the maximum hot side temperature is the outside air temperature at the lowest altitude of flight—in this case 735 K at Z = 0 km. The minimum cold side temperature is the outside air temperature at the highest point of the flight. The temperature ratio and therefore the Carnot efficiency increases as the total ΔZ increases. For example, for ΔZ = 0 to 20 km $R \approx 1.28$, but for ΔZ = 0 to 30 km $R \approx 1.49$. This leads to a Carnot efficiency $\varepsilon \approx 33\%$ for ΔZ = 0 to 30 km. But this level of efficiency is only achievable if the heat sink is heated by the outside air at the lowest point of the flight, carried to the highest point in an insulated container, and then used to power a heat engine whose cold side is maintained at the outside air temperature at that altitude—or vice versa.

However, this peak efficiency is not sustainable because an onboard heat sink is a finite heat sink. As heat is extracted, its temperature decreases, and R asymptotically approaches 1. This cuts the effective Carnot efficiency of a finite heat sink powered heat engine approximately in half, compared to what it would be if the heat sink did not change temperature. However, this process can also be operated in the opposite direction on the descent, effectively gaining back the loss of apparent Carnot efficiency due to a finite heat sink. Thus, a cycle from 0 to 30 km and back could yield an effective ideal Carnot efficiency of ≈ 33%. However, real Stirling engines can achieve only about 60% of ideal Carnot efficiency due to internal losses. Thus, the realizable efficiency of a heat sink powered Stirling engine operating between 0 to 30 km and back would be no more than ≈ 20%.

More efficient heat engine cycles, such as the Ericsson cycle were investigated. An Ericsson cycle engine with recuperators approaches ideal isothermal heat flow and is projected to be able to achieve ≈ 85% of Carnot efficiency. This level of efficiency would produce an overall cycle efficiency of ≈ 28% when cycling between 0 to 30 km and back.

Windmilling on Descent

To minimize the amount of energy required from a heat sink engine for the climb portion of the flight, a strategy was adopted in which energy is recovered via a windmilling propeller when the aircraft descends. The energy can be stored in rechargeable batteries for use during the ascent portion of the flight. Performance modeling showed that approximately 60% of the geopotential energy contained in the mass of the aircraft can be recovered and made available for this purpose.

Wind Power on the Surface

Having made the decision to harvest wind energy on descent we considered whether it might be possible to land on the surface and harvest additional wind energy from the surface winds while the aircraft remained stationary. Because this would entail vertical takeoff and landing, we conducted a supplementary design study to determine whether such a vehicle could fit within an atmospheric entry vehicle. See Figures 10a, 10b, and 10c.

Because atmospheric density at the surface at Venus is roughly one-fifth that of water, forward flight speeds of 3-5 m/s can result in sufficient lift to keep an aircraft airborne close to the surface. This raises the possibility of landings and take offs with negligible forward movement using vertical facing rotors. Forward flight efficiency could be enhanced by small wings which could serve a dual purpose of providing both lift and support structure on which to mount external actuators. This set of design considerations can be accommodated by tilt-rotor or tilt-wing configurations.

Figure 10a shows a nominal VTOL tilt-rotor configuration stowed in a blunt body aeroshell. Figure 10b shows the vehicle in VTOL/forward flight mode, and Figure 10c shows the vehicle in a

configuration which would use the rotors to generate power for landed science and housekeeping while on the surface.

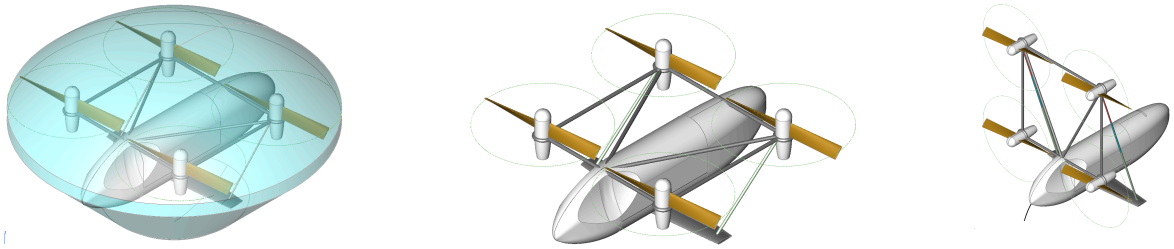


Figure 10a. Tiltrotor in Aeroshell; Figure 10b. Tiltrotor in Flight; Figure 10c Tiltrotor in Windmill Mode.

The surface wind at Venus averages approximately 1.5 m/s, which corresponds to a power flux of approximately 110 W/m^2. The windmill collection area would be approximately 15 m^2, leading to a power flux through the windmill assembly of ≈ 1,650 W. However, the rotors must be designed primarily for flight and vertical ascent. Such rotors have a conversion efficiency of ≈ 15% when used as windmills at low disc loading, leading to a harvested power ≈ 250 W. Although it proved feasible to package a quadrotor configuration into the reference aeroshell, the amount of wind power harvestable from the propulsive rotors while on the ground was estimated to be less than the amount of power required to actively cool the vehicle while on the ground. Thus, no net benefit was seen in this approach, and it was dropped from further consideration.

Preliminary Feasibility Assessment

By this point in the study, the flight cycle we envisioned is as follows:

1. The aircraft climbs from 0 to 30 km altitude using electrical energy stored in batteries powering an electric motor. This allows the motor to be sized to provide the most efficient rate of climb for the aircraft. The Lithium heat sink is maintained at 735K on the way up and is sealed from the outside air inside an insulated container.
2. At the apex of the flight, the aircraft begins a powered, level flight leg, during which it extracts energy from the temperature difference between the Lithium heat sink and the outside air. This energy is used for powering the flight leg, powering the science instruments (including active cooling), and communicating with Earth. It neither contributes to nor subtracts from the energy available for cycling between the altitude limits.
3. The aircraft then descends, using the windmilling propeller to recover some of the potential energy by charging batteries. During descent, The Lithium heat sink is maintained at 492K sealed from the outside air inside an insulated container.

4. At the bottom of the cycle, the aircraft flies in level, powered flight while again converting the temperature difference between the outside air and the Lithium heat sink into electrical power, which is stored in the batteries.

Based on this cycle, the following quantitative estimates emerge:

1. The potential energy to lift 1 kg of mass 30 km vertically through the Venusian gravity field is ≈ **263** KJ/kg.
2. With an average conversion efficiency (propulsive power to rate-of-climb) of approximately 62%, 424 kJ/kg of stored energy is required at the beginning of the climb in order to reach 30 km.
3. Approximately 60% of the potential energy of the vehicle (**158** kJ/kg) can be recovered via windmilling on descent. That would reduce the amount of energy required to be produced at the bottom of the cycle to (**424** – **158) = 266** kJ/kg.
5. A kg of Lithium moving through the ΔT from 30 to 0 km has a thermal energy content of ≈ **846** KJ/kg.
6. For thermodynamic conversion. efficiency $\varepsilon \approx .28$, approximately **237** kJ/kg is harvestable from the Lithium heat sink at the bottom of the cycle, leaving a net deficit of (**237** – **266) = 29** kJ/kg.
7. This analysis implies that essentially the entire mass of the aircraft would have to consist of Lithium for the cycle to close. This leaves no mass available for heat engine, batteries, payload, airframe, etc.

Buoyancy

At this point in the study, it was realized that a cycling, heat sink powered aircraft using only aerodynamic lift is probably not feasible given the low levels of propulsive and thermodynamic efficiencies that are practically available from mechanical heat engines. However, it was also realized that incorporating buoyancy into the problem could potentially improve the energy balance.

A cycling, hot air balloon will need to have variable volume and maintain a near-zero pressure between the inside and outside. Cycling between 0 and 30 km, for example, will require the volume at altitude to be approximately 6.6 times the volume at 0 km. With this realization, we examined possible candidate technologies for construction of balloon envelopes capable of functioning in the extreme heat of Venus' atmosphere at or near the planet's surface. The aerospace polymer Kapton (which has been proposed as a Venus balloon material) can survive at temperatures down to approximately 10 km altitude. The aerospace polymer Kevlar can theoretically survive at Venus surface temperatures, but it would be within ≈ 10 C of its melting point and its strength would be only a small fraction of its strength at room temperature. We could find no data in the publicly available literature in which the material properties of Kevlar were characterized at 735 K in a CO_2 atmosphere, thus we cannot assume the use of this material.

However, in 2005 JPL proposed a balloon system for use in a Venus sample return mission. Their Low Altitude Balloon, operating from 0 to 15 km was an accordion-like bellows constructed of thin stainless steel foil sections. In terrestrial applications bellows of this type can withstand 1000s of expansion/compression cycles and can be constructed of metals other than stainless steel. We chose as the baseline for our concept a bellows constructed of Titanium foil sections edge-welded with lasers or electron beams.

Hot Air Balloons

Our initial approach to incorporating buoyancy into our aircraft was to consider the use of a Lithium heat sink as a heater in a hot air balloon. If such a balloon was initially near neutral buoyancy at 0 km altitude, a small heat addition would make it positively buoyant and maintain that positive buoyancy to altitude. One reason for this is that the heat transfer from a heat sink to a working gas can be accomplished nearly isobarically. In this case, the ratio between heat input and P-V work done is determined only by the ratio of the heat capacity of the working gas at constant pressure, c_p. to the heat capacity at constant volume c_v. Monatomic gases in general have the largest c_p/ c_v and Helium has the highest absolute value of c_p because it is the lightest monatomic gas. The practical effect of this is that if Helium is used as the lift gas, the heat sink will convert its heat energy into P-V work with an efficiency of ≈ 40%. Not only is this value above the 31% required for theoretical breakeven, it stays essentially constant with altitude. A second reason buoyancy is attractive in this application is that it requires almost no additional onboard mass for machinery, except the balloon envelope itself. The surrounding atmosphere of the balloon essentially is the heat engine and does not contribute to dead mass.

We modeled the performance of such a bellows, filled with Helium as the lift gas, operating between the surface and altitudes up to 30 km. In order to produce a net gain in energy as such a device cycle between the surface and altitude, there must be a difference between the volume-versus-altitude profile on ascent, as compared to descent. We found that the optimum approach is to allow the bellows to equilibrate with the outside air temperature and pressure on ascent and to use a Lithium heat sink starting at the temperature of the outside air at the highest altitude to extract heat from the lift gas (i.e., supercool it) on descent. If the bellows is filled to provide slight positive buoyancy at the surface, it will provide steadily increasing lift on ascent because of the different gas constants of Helium and CO_2. When heat is extracted on descent into a cooler heat sink (supercooled), the bellows will have consistently less buoyancy than the surrounding atmosphere. Therefore, it will fall under its own weight and the kinetic energy of that fall can be extracted by a windmilling propeller and stored in batteries to power the ascent and onboard energy needs.

To explore this option, the fuselage of the basic Blended Wing Body design was enlarged to the maximum extent possible to accommodate a variable volume, thin-wall Titanium bellows. It was found that the largest bellows that could be fit within the outer mold line of the fuselage had an expansion ratio of ≈ 2.6 (fig. 11). Because this would have limited the altitude range to about 15 km, the LAVA team examined ways to increase the expansion ratio; one is shown in figure 12.

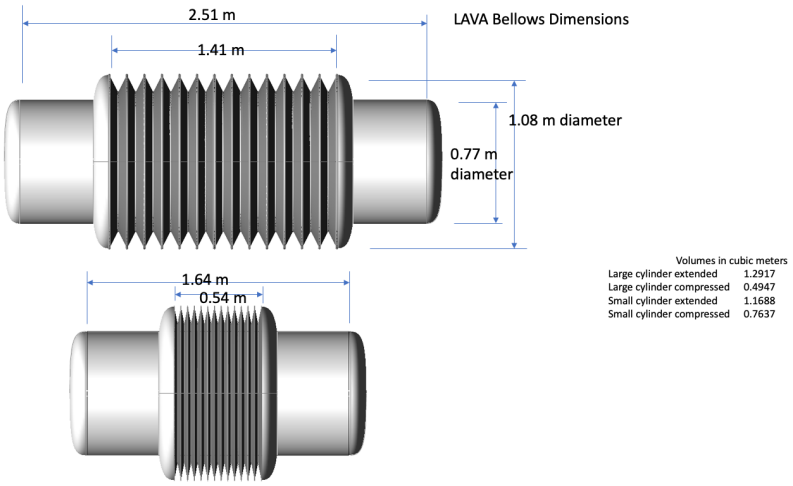


Figure 11. Horizontal Bellows in Expanded and Compressed Positions.

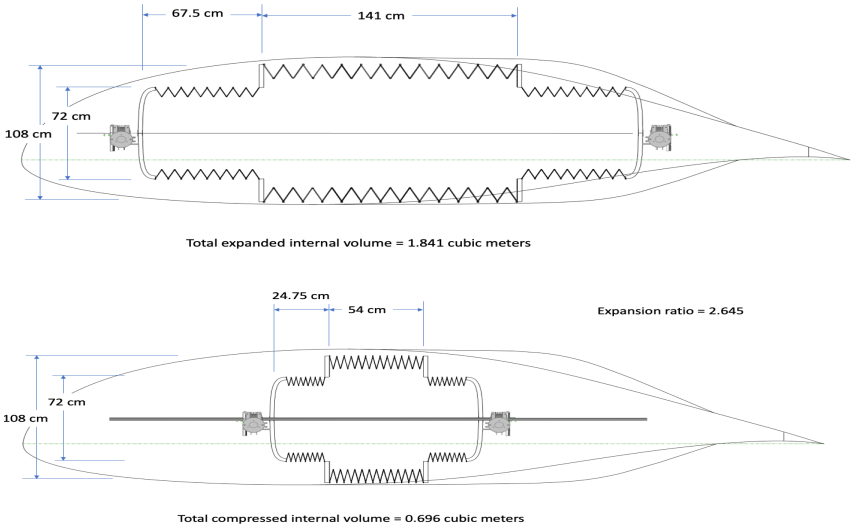


Figure 12. Horizontal Bellows in Expanded and Compressed Positions Inside a Candidate Airfoil.

Either of these bellows designs can fit into a BWB C-Wing, as shown in Figure 13, thus providing an expansion ratio of approximately 2.6. A larger expansion ratio can be attained with a more complicated arrangement by allowing the nose and tailcone to move along the centerline as shown in Figure 14.

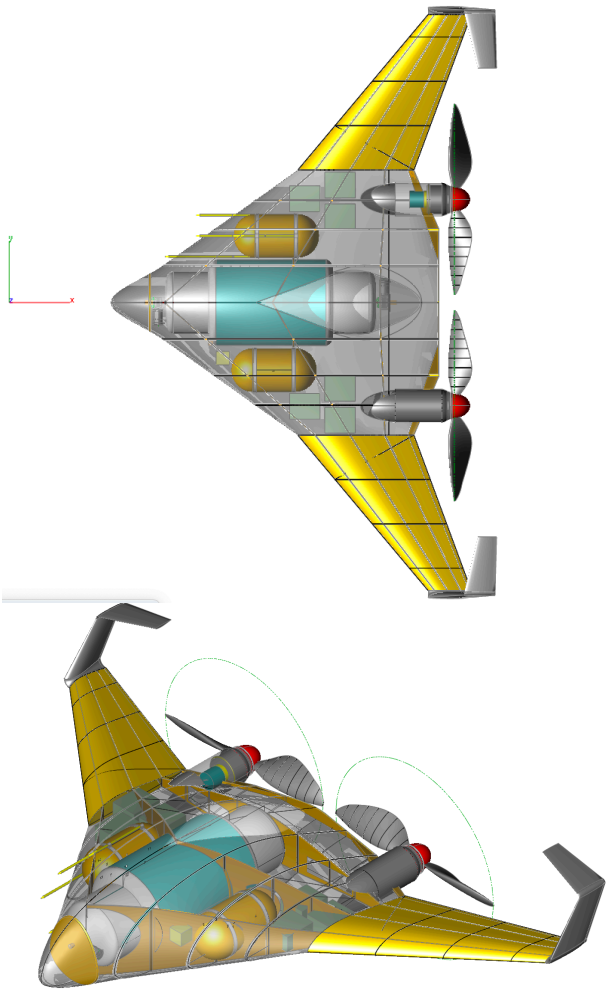


Figure 13. Horizontal Bellows Fitted into a BWB C-Wing.

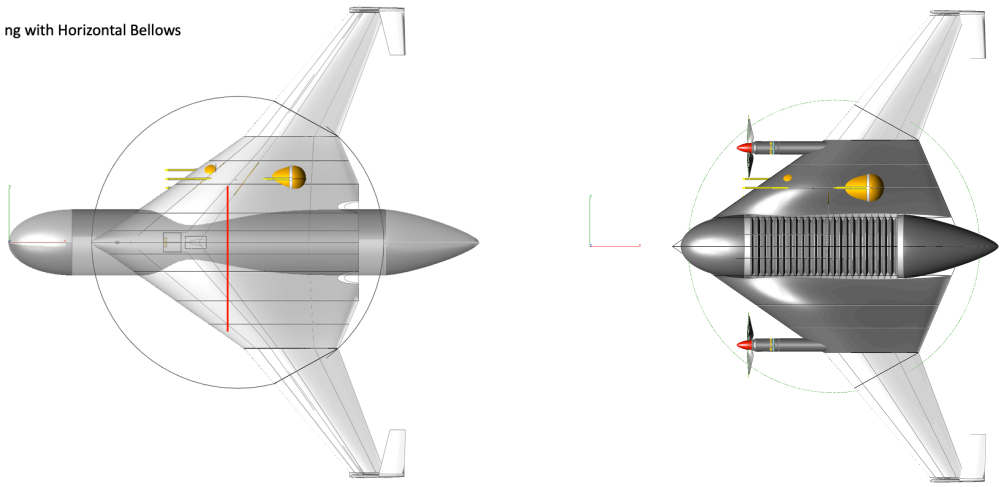


Figure 14. Horizontal Bellows with Movable Nose and Tailcones.

Carrying this line of thought to an extreme, the bellows could be reoriented vertically to provide considerably more buoyant lift and with an expansion ratio of 3.0. See Figure 15.

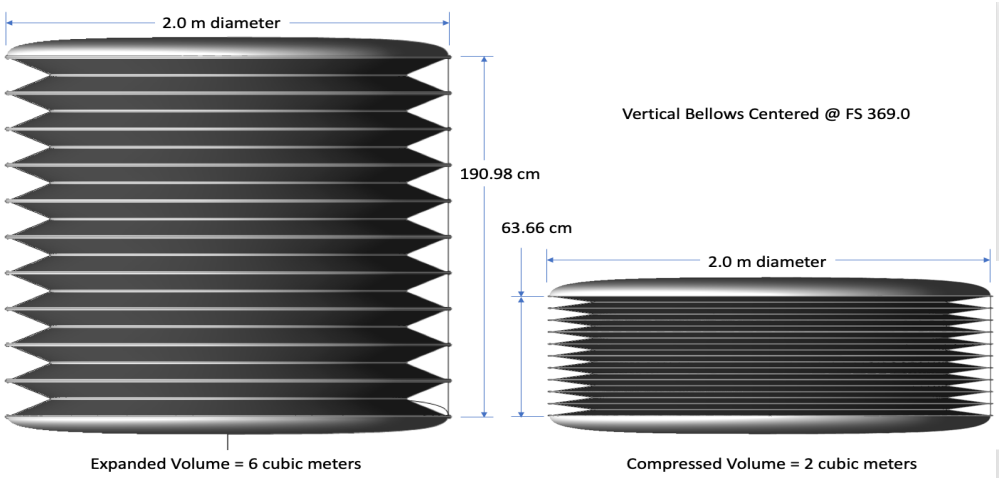


Figure 15. Vertical Bellows Arrangement

Figure 16 shows how this would be fitted into a BWB C-wing configuration. In theory, the bellows would only expand to its full height when lift is entirely buoyant and there's little or no forward motion.

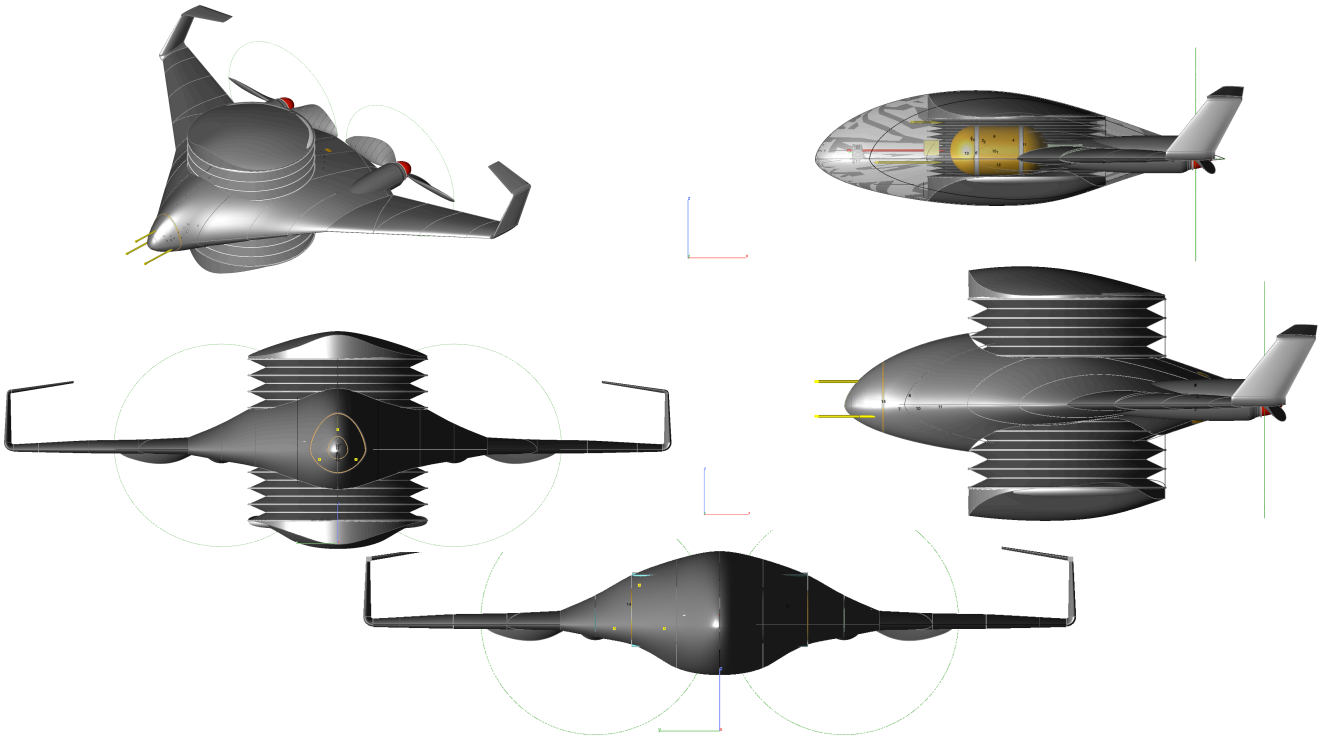


Figure 16. Vertical Bellows Fitted into a BWB C-Wing.

Dirigibles

Because a buoyant vehicle cycling between 0 and 30 km would require an expansion ratio of ≈ 6.6, it was realized that the only way to achieve an acceptable altitude range utilizing buoyancy was to dispense with the blended wing body configuration altogether and adopt a more traditional airship design.

One or more flexible bladders inside a rigid shell can accommodate expansion ratios up to about 10 and would allow an altitude range up to ≈ 36 km. However, this would require a larger air vehicle and therefore a different entry vehicle. To accommodate a traditional airship configuration, we assumed the use of a rigid, mid L/D, symmetric, aeroentry vehicle with the same dimensions as SpaceX's stretched Falcon 9 launch shroud. When used as a combination launch shroud/entry vehicle, its aerodynamic performance is expected to be similar to that of entry vehicles studied in e.g., NASA's DRA5 Mars Mission Architecture. The geometry is shown in figure 17.

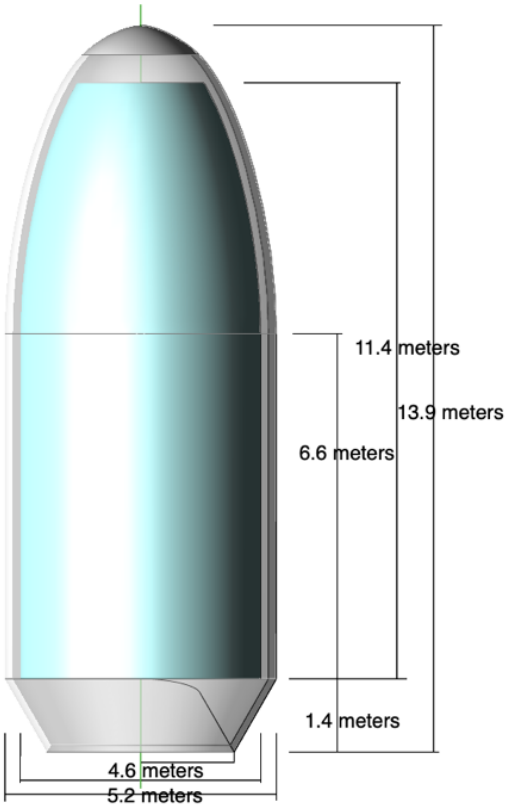


Figure 17. Five Meter Diameter Mid L/D Aeroshell

Solar Energy

Although the useful solar flux at or near Venus' surface is negligible, the total flux (upwelling and downwelling) at altitudes of 15 km is a few hundred W/m^2 per steradian. A collection area of a few square meters is sufficient to provide kW levels of useful solar power when harvested

by specially designed solar cells tailored to the solar spectrum in the Venusian atmosphere and the high temperature. Significant work has gone into the development of spacecraft subsystems and components capable of surviving and operating at the conditions of the Venusian near-surface under the NASA-Glenn HOTTECH program. For example, preliminary calculations indicate that Low Intensity High Temperature (LIHT) Solar Cells being developed by HOTTECH may be sufficient to power an airship. We assumed the use of HOTTECH Solar Cell technology in our designs.

The resulting aircraft concept is a near-neutral buoyancy vehicle incorporating Titanium foil bellows together with a solar array consisting of specialized solar cells, covering a large portion of the wings and fuselage. Solar-electric energy provides the small amounts of climb energy required as well as parasitic loads for thermal management, instrument operation, telecommunications, etc.

Figures 18a and 18b show a design iteration of a rigid airship with small lifting surfaces sized to fit within the 5.2-meter diameter mid L/D aeroshell.

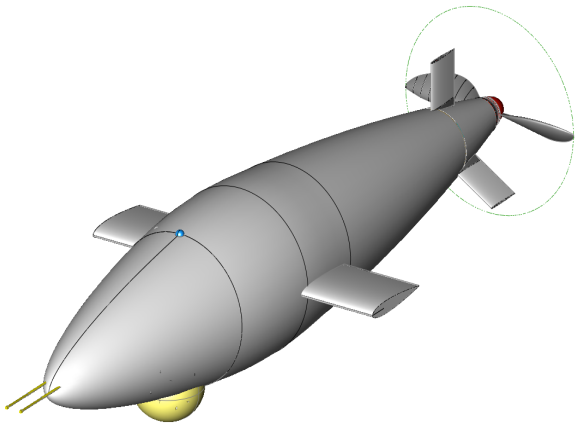


Figure 18a. Variation of a Rigid Airship with Lifting Wings.

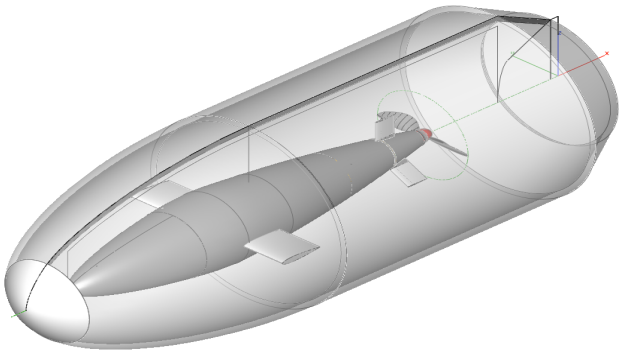


Figure 18b. Rigid Airship Stowed in Mid L/D Aeroshell.

This configuration, which is as much rigid airship (dirigible) as airplane, has minimal lifting surfaces, yet its aerodynamic performance doesn't differ much from the BWB C-Wing configuration, as shown in Figure 19 and Table 4. These data were run at 220 kg gross mass assuming neutral buoyancy.

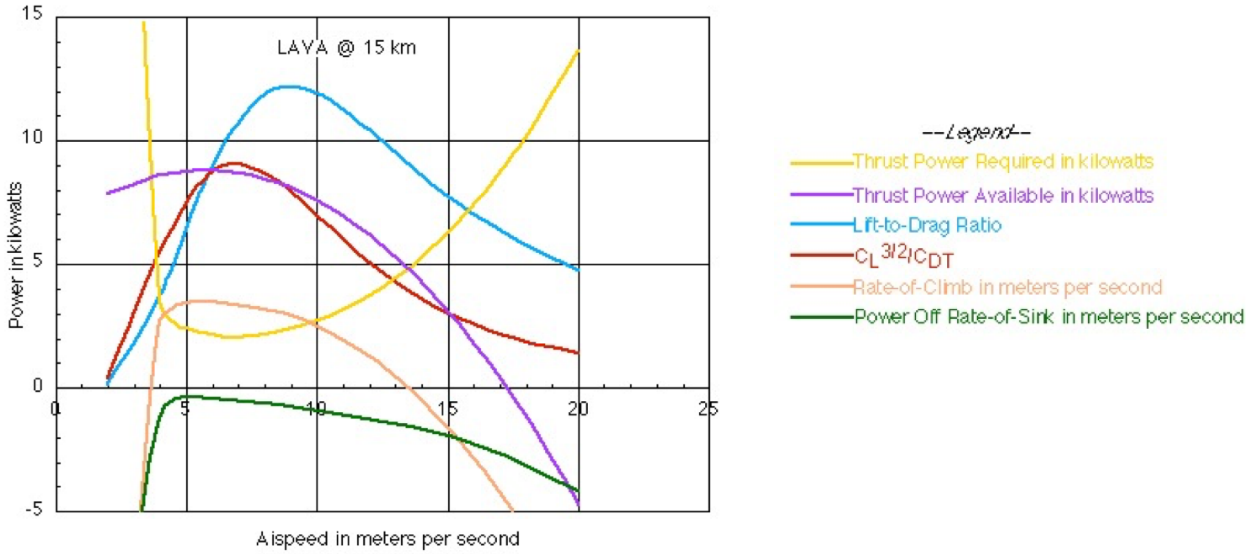


Figure 19. Buoyant Hybrid Rigid Airship Performance at Neutral Buoyancy.

Maximum Speed	=	13.5	mps
Minimum Sink Rate Speed	=	5.0	mps
Minimum Sink Rate	=	40.1	mpm
Maximum Climb Rate	=	203.5	mpm
Maximum Climb Rate Speed	=	5.0	mps
L/Dmax	=	11.9 @ 9	mps
CL^1.5/CDTmax	=	8.8 @ 7	mps

Table 4. Selected Performance Points for Buoyant Hybrid Rigid Airship at Neutral Buoyancy.

The Starship Dirigible

Although the neutral buoyancy approach yielded a design capable of repeatedly cycling between 0 and 30 km altitude, its gross mass is limited by the amount of buoyancy that can be supplied by the lifting gas volume. The rigid airship depicted in figures 18a and 18b had approximately 16.5 m^3 of lifting gas at 30 km altitude. The resulting gross mass (220 kg) was not enough to accommodate the reference science instrumentation package.

Because of square-cube law mass relationships, it is well-known that the performance of airships improves monotonically with size. This is opposite to square-cube scaling laws for winged aircraft which generally have an upper practical limit on size. This suggested that we design an airship

for the largest feasible size. Two technical developments on the foreseeable horizon which may be critical for the feasibility of this concept are the SpaceX Starship—capable of delivering ≈ 100 tons and 1,100 cubic meters of payload to Venus orbit and the Adaptive Deployable Entry Placement Technology (ADEPT) entry system being developed by NASA Ames Research Center. The ADEPT deployment approach allows very large planetary atmosphere entry vehicles to be folded and stowed inside launch vehicle shrouds and then be unfurled to provide low ballistic coefficient, lifting entries into planetary atmospheres. ADEPT was specifically designed to accommodate Venus atmosphere entries.

Our Starship dirigible mission concept is to use positive buoyancy to rise to a maximum float altitude and use negative buoyancy to descend to a minimum float altitude. This mission profile requires that we actively adjust the bellows volume at both the minimum and maximum altitudes to achieve neutral buoyancy at those points in the cycle. In our design this is accomplished by compressing lift gas from the buoyancy cells into high pressure storage tanks at the maximum float altitude to achieve neutral buoyancy and then compressing additional gas into the high-pressure tanks to achieve negative buoyancy when it is desired to descend. The power for this comes from solar cells, which produce their highest output at the maximum float altitude. The high-pressure storage tanks are pressurized to a level that is slightly higher than the outside pressure at the minimum float altitude. When it is desired to float at the minimum altitude or ascend, gas is simply released from the storage tanks back into the bellows to achieve neutral or positive buoyancy, with no need to consume power in the process. In this process, the airship alternates between being heavier than air, lighter than air, and neutrally buoyant.

The SpaceX Starship payload bay dimensions are shown in figure 20 in comparison to the dimensions of the blunt body entry vehicle and the 5.2-meter diameter mid L/D entry vehicle:

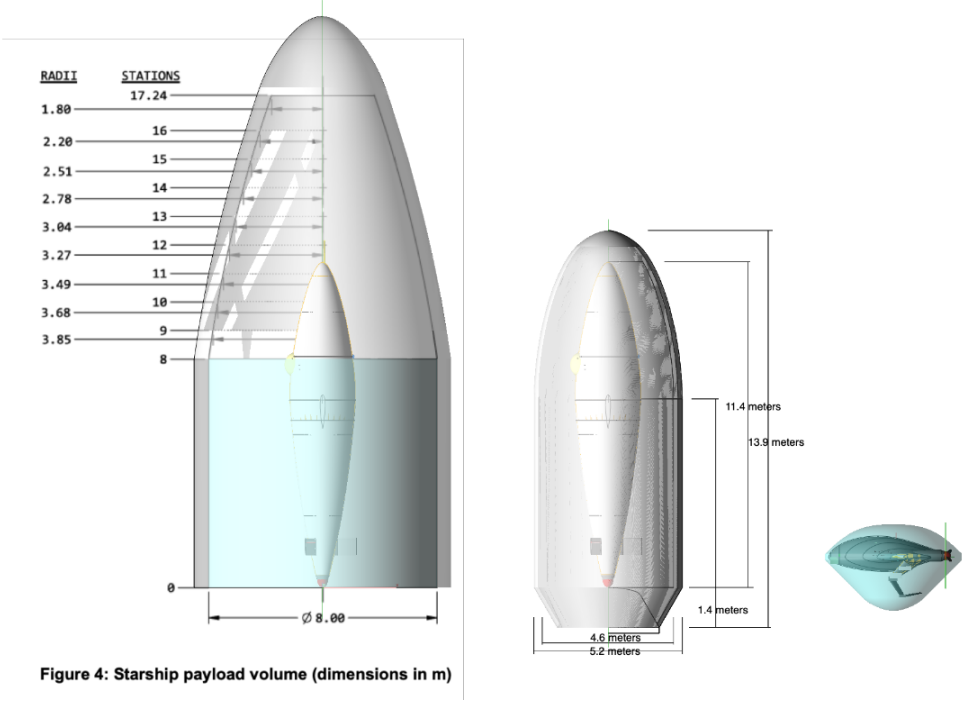


Figure 4: Starship payload volume (dimensions in m)

Figure 20. Comparison of Entry Vehicles Used in Study

We designed a dirigible with proportions similar to the Hindenburg to make optimal use of the internal dimensions of the Starship payload bay. Side and top views are shown in figure 21 and selected engineering parameters are shown in Table 5.

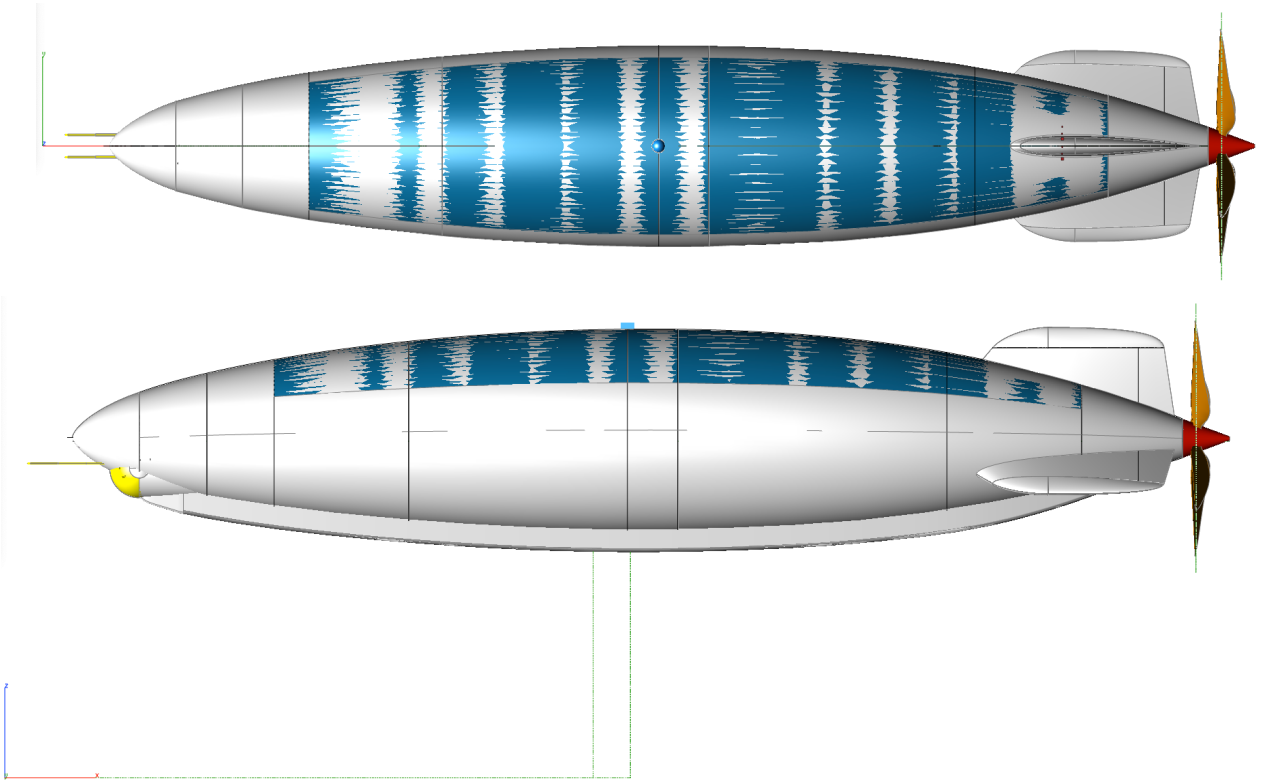


Figure 21. Hindenburg-Style Dirigible Sized to fit in SpaceX Starship Payload Bay

Overall Length	17.20 meters
Overall Height	3.318 meters
Overall Width	3.000 meters
Empty Mass	396.7 kilograms
Gross Mass	529.1 kilograms
Payload & Fixed Equipment	95.4 kilograms

Table 5. Selected Engineering Parameters for Starship Dirigible

SUMMARY AND CONCLUSION

Winged Flight

We studied the concept of utilizing the heat energy stored in a high specific heat capacity material to power a winged aircraft capable of flying on Venus from low altitude to high altitude and back by extracting that heat energy via an onboard heat-exchange engine.

We began by defining the scientific rationale for such a mission, the instrument suite such a vehicle would carry, and the type of scientific measurements to be made. This instrument suite was guided by NASA's Venus Exploration Analysis Group (VEXAG) publication "Goals, Objectives, and Investigations for Venus Exploration".

We next defined a reference Venus atmospheric entry vehicle based on the NASA Ames Research Center study: "Atmospheric Entry Studies for Venus Missions:45° Sphere-Cone Rigid Aeroshells and Ballistic Entries". This entry vehicle was sized to fit within the launch shroud of a SpaceX Falcon 9 launch vehicle.

We next designed a representative aircraft that could be folded to fit inside the nominal 5-meter diameter blunt body atmospheric entry vehicle and then be deployed mid-air on Venus. This representative aircraft allowed us to determine how aerodynamically efficient such an aircraft could be. The requirement to fold and aircraft to fit inside a blunt body entry vehicle imposes constraints on the maximum aspect ratio such an aircraft can achieve and therefore also on the maximum L/D. By adopting a blended wing-body design with C-wingtip extensions, we were able to achieve a maximum L/D of ≈ 11.9.

In climbing from low altitude to high altitude, a fraction of the available propulsion energy goes into overcoming aerodynamic drag and the remainder goes into overcoming gravity (i.e., gaining altitude). The efficiency with which an aircraft can convert onboard stored energy to altitude is a function of both the L/D of the airframe and the design of the propulsion system. The total energy expended will be minimum when the best rate of climb speed and angle is maintained at each altitude. The best rate of climb speed is the speed (and therefore climb angle) that balances aerodynamic drag of the airframe and efficiency of the propulsion system. Climb efficiency favors a propulsion system with a high ratio of propulsive power available to the propulsive power required for horizontal flight. Our reference design achieved best rate of climb when the ratio of power available to power required for horizontal flight is ≥ 10.

Because planetary atmospheres become more rarified with altitude, minimum horizontal flight speed also increases with altitude, with a corresponding increase in minimum power required. This leaves less power available for climb, so the most efficient rate of climb will occur at the

minimum flight altitude and fall off monotonically with altitude. For our reference blended-wing-body C-wing design, we found that the efficiency with which the aircraft could convert stored energy into altitude was ≈ 80% at 0 km altitude, declining to ≈ 43% at 30 km, with an average value over the cycle of ≈ 62%.

Similarly, when descending from high altitude to low altitude, we found that such an aircraft can recover ≈ 60% of the geopotential energy in the form of electricity stored in batteries via a windmilling propeller.

We surveyed the component and materials technology available to support the design and construction of airframes capable of cycling between low altitude-high temperature and high altitude-low temperature. There appear to be design solutions for all major aircraft subsystems except the heat engine. For example, airframes and pressure vessels constructed of Titanium would withstand the temperature cycling and corrosive atmosphere of Venus' atmosphere. Electronics, actuators, batteries, and solar cells capable of functioning at or near the surface are being developed under NASA's HOTTech technology development program and would greatly reduce the need for active cooling on a cycling aircraft. We found that we could construct any of our airframes with technology already or soon to be available.

We surveyed the technology available for converting heat energy into usable power onboard the aircraft and could find no technical approach with sufficient power-to-weight ratio and thermodynamic efficiency to allow a winged aircraft design to close. This is mainly due to the low temperature initial difference between the hot side and cold side, and the fact that the temperature difference in a finite heat sink declines as energy is extracted.

Buoyancy

We next examined the use of thermal driven buoyancy (i.e., a hot air balloon) as a possible solution to the lack of a suitable conventional heat engine. In principle, the gas inside an expandable envelope (balloon or bellows) can be heated and/or cooled with an onboard heat sink and thereby do pressure-volume work directly on the surrounding atmosphere to produce lift or downforce. Other than the mass of the balloon or bellows and the gas, no additional mass is required for energy conversion apparatus. We found that the minimum mass solution for this approach occurs when the heat sink thermal capacity is equal to the thermal capacity of the lift gas (assumed to be Helium) at constant volume (C_v). To extract net useful energy out of this system, it is necessary that the integral of lift times distance on ascent be larger than the integral on descent. This can be accomplished by either superheating the lift gas on ascent to an average temperature greater than its otherwise isothermal value, or supercooling it below that value on descent, but not both. We found that the preferred approach is to supercool the lift gas on descent. An onboard heat engine used to supply shaft power is double acting; it can produce power at both the top and the bottom of the altitude cycle. Although a hot-air balloon approach produces a small net energy gain over one ascent-descent cycle considering only the potential energy budget, the gain is not enough to make the design close when other energy requirements are considered, such as science instrument power, communications, and thermal control. This

implies that such a heat sink driven buoyant craft would still need an auxiliary power source, with solar power being the obvious candidate.

Although the Venusian atmosphere is dense and opaque, modeling showed that useful amounts of solar power is harvestable with special high temperature solar cells between altitudes of approximately 15 to 30 km. If solar power is to be used on a Venus aircraft, then it can supplement or—in some cases—replace heat-sink derived power.

Therefore, our next approach was to examine passive buoyancy (i.e., no use of superheating or supercooling of the lift gas) coupled with solar power. This approach would minimize both the energy required to gain altitude and the energy available to be recovered on descent. We designed a winged aircraft containing a bellows constructed of thin-wall, edge-welded Titanium foil sections with a dynamic volume range (maximum volume divided by minimum volume) of ≈ 6.5. The bellows would be contained within a thin-wall rigid shell of volume ≈ $10m^3$, to maintain an aerodynamically efficient shape. Such a bellows can maintain a nearly constant buoyancy from 0 to 30 km altitude on Venus, with no pressure difference between the inside and outside of the bellows. If the total buoyancy is near the total system weight, such a vehicle is essentially a dirigible requiring minimal propulsive power. Although the insolation available at the surface of Venus is negligible, the insolation at 30 km is enough to provide kW levels of power. Moreover, because such a dirigible could remain at high altitude with minimal expenditure of power, it could stay at high altitude long enough to charge batteries and then venture down to the surface, whenever science observations are desired, rather than being constrained to a fixed schedule. However, such a dirigible can probably not fly faster than the super-rotating atmosphere, so it would have to be able to survive ≈ 50 hours on the night side of Venus in order to allow a long duration mission.

Given that this approach appeared feasible, it became apparent that high aspect ratio wings on a vehicle of this type served no important function and that a more conventional dirigible shape would probably be more efficient. However, a conventional dirigible would be restricted to a length of no more than about 5 meters if it were constrained to fit into a blunt body entry vehicle. A vehicle of this size would have a useful payload mass far too small to accommodate the reference instrument payload. We therefore designed a dirigible around a mid L/D lifting body shape consistent with the dimensions of a SpaceX Falcon 9 launch vehicle. This resulted in a dirigible with small stubby wings (to enhance the solar array area) with a total mass of ≈ 220 kg and aerodynamic performance not significantly different from the buoyant blended wing body aircraft. Although such a vehicle appears feasible, its payload capacity would not be sufficient to accommodate the entire science instrument complement, and we decided to study the performance characteristics of a larger Venus dirigible.

For purposes of this last element of our study we assumed the existence at some time in the future of the SpaceX Starship—capable of delivering ≈ 100 tons and 1,100 cubic meters of payload to Venus orbit and of the Adaptive Deployable Entry Placement Technology (ADEPT) entry system—capable of being stowed in the Starship payload bay and unfurled to deliver a

large payload into the Venusian atmosphere. Therefore, our final design was of a solar powered dirigible sized to fit inside the payload bay of the SpaceX Starship.

The resulting dirigible is ≈ 17.25 meters in length, 3 meters in diameter, and has a gross mass of ≈ 530 kg. Because of its size, there is room on the external skin for solar arrays producing ≈ 35 kW of electrical power at 30 km altitude and enough lift to accommodate a ≈ 190 kg pressurized, thermally controlled science instrument pod that includes all the instruments identified in the science section of this study. This design would allow a mission profile in which the dirigible would float at ≈ 30 km altitude while charging batteries and communicating with Earth, while floating with the wind into and through the night. As the vehicle nears the dawn terminator, it would make itself negatively buoyant and descend as rapidly as possible to the near surface (≈ 0 km altitude)—arriving near local sunrise. It would then float at that altitude taking scientific measurements and images as long as possible consistent with the battery lifetime. It would then make itself positively buoyant and rise again to the float altitude, where the cycle could be repeated indefinitely.

Conclusions

- We conclude that a cycling, heavier than air winged Venus aircraft powered solely by an onboard heat sink powered heat engine is not feasible with existing or foreseeable technology due to the poor Carnot efficiency associated with a finite heat sink and due to the poor power density of such devices.
- A cycling, partially buoyant winged Venus aircraft design powered by an onboard heat sink supercooling the lift gas on descent and supplemented with solar arrays comes closer to being feasible but still does not close.
- A dirigible containing zero-pressure buoyancy cells and powered by high temperature solar cells capable of cycling between 0 and 30 km is probably feasible with a combination of existing and foreseeable technology.
- Because of the square-cube scaling law associated with lighter than air vehicles, the largest feasible Venus dirigible will have the largest payload mass fraction and the most conservative performance margins. The advent of the SpaceX interplanetary Starship and the development of the lightweight, deployable ADEPT atmospheric entry vehicle technology will enable the delivery of a dirigible to the Venusian atmosphere with a length of ≈ 17 m, a diameter of ≈ 3 m, and a gross mass of ≈ 590 kg. Such a vehicle could accommodate the full complement of science instruments identified in the science section of this study and cycle between 30 km and 0 km indefinitely.

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