Design Considerations for Venus Microprobes

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Introduction

THE Venusian surface and lower atmosphere are interesting but inhospitable environments that to date have been explored by 50- to 350-kg probes. Sensitivity to environmental conditions is acute for smaller (and cheaper) vehicles, where the timescales for the vehicle to equilibrate thermally with its hostile surroundings are short and comparable to the free-fall descent time. I explore how small a Venus microprobe carrying a specified payload can be made and still use conventional low-temperature (< 373 K) electronics, with vehicle shape and size, insulation technology, and phase-change thermal ballast as free parameters.

The envisioned scenario has an autonomous balloon[1,2] (or "aerobot") floating at about 60-km altitude in the Venusian atmosphere, where ambient conditions are relatively benign (0.2 bar, 268 K). An aerobot with altitude control capability could descend to the surface for a short period, but a number of technical challenges must be overcome.[3] A lower-risk approach for obtaining data on high-priority surface sites is to release small, expendable descent probes as the aerobot passes overhead. Maximizing the number of sites that can be explored requires that these probes have a small mass.

A low-mass probe, however, first will have a small thermal inertia and so will warm up quickly. Further, a small probe will fall slower and hence can warm to unsurvivable temperatures before the surface is reached. Design options to combat these challenges are 1) heavy probes, i.e., either larger, more capable probes or added mass ballast; 2) slender probes, giving higher ballistic coefficients and hence shorter descent times; 3) added insulation to slow the rate of heat leak into the probe; 4) added thermal ballast, such as a phase-change material; 5) increased packaging density of the probe systems themselves to maximize the ballistic coefficient; and 6) use of systems that can tolerate higher temperatures.[4] Active cooling,[5] likely to be neither practicable nor worthwhile on a small system such as a microprobe, has not been considered.

Options 1 and 3 essentially lead to solutions similar to the Pioneer Venus probes. Option 2 has potential, although increasing the slenderness of the probe also increases the wetted surface area (through which heat can leak) and increases the mass of insulation required for a given insulation thickness. Option 4 has been used on Russian landers (in combination with 1 and 3). Note that, in this application, the common figure of merit for thermal ballast, namely heat absorbed per kilogram, may not be the most appropriate, inasmuch as the material also acts as mass ballast. Option 6 is not explored in this Note.

Mission and Payload

The microprobes are envisioned to take images of the surface during descent, like the Ranger lunar probes. As such, the payload consists of a charge-coupled device (CCD) camera (which rules out option 6), the power source, data-handling electronics, a radio transmitter, and the supporting structure. Once released, the probe would not be commanded but simply would acquire and transmit data. Typical descent times are a fraction of an hour, and it is assumed that the minimum feasible package would be 0.2 kg.

Modest enhancements to this basic probe could include temperature and pressure sensors, a simple nephelometer (laser plus photodiodes) and up-and-down radiometers to measure the particulate opacity and insolation profiles, accelerometers to record wind gusts and the surface impact, electrical-property sensors (the failure of external sensors[6] on all four Pioneer Venus spacecraft at 12-km altitude remains unresolved but has been attributed to atmospheric electricity), and solid-state (absorption spectroscopy?) composition measurements. To minimize heat leaks by penetrations, the optical sensors might be coupled to the outside by optical fibers. It is assumed that more sophisticated measurements, such as detailed gas composition by mass spectrometry, are more practicably performed from the aerobot itself. A Doppler tracking experiment between the probe and the aerobot might be a useful measurement, although whether a suitable oscillator can be packaged in the probe has not been evaluated. To accommodate these and other enhancements, I also have studied package masses of 0.6 and 2.0 kg. A sketch of a possible probe configuration is presented in Fig. 1. A modest slenderness ratio to allow a rapid descent is indicated, and penetrations of the insulation layer are limited to cables to interface with the aerobot (not shown), wires to external sensors and the antenna, and optical fibers. The camera focal plane is matched to the optics image plane by a fiber-optic bundle: This coupling maximizes the thermal path between the imager and the outside.

Performance Model

The performance model is derived from one used in a previous study of deep Jupiter probes.[7] It is assumed that the probe is released from an altitude of 60 km and free falls to the surface. The terminal velocity is computed on the basis of a fixed drag coefficient C_D (assumed conservatively to be 0.3, the value depending on the slenderness of the vehicle as well as the use of turbulating strips and/or a streamlining cowl), the mass/area ratio, constant gravity of 8.98 m/s^2, and a density profile for the Venus atmosphere from Seiff et al.[8] The time spent in each 1-km-altitude interval is computed, and the heat flow is determined from the wetted area and insulation conductivity and thickness, assuming steady-state conduction through the insulation. The corresponding internal temperature rise is calculated on the assumption that the payload has a specific heat c_p of 800 J/kg K and taking into account internal heat dissipation and buffering by any phase-change material. The specific-heat capacity of the insulation is lumped with that of the payload; this simplification introduces a slight favorable bias in the temperature-rise rate. The bias is less than 10% for the cases of interest, although in poor cases in which the insulation mass is large, the error is significant. The effects of buoyancy in the dense atmosphere and the first 20 s of descent (in which the probe accelerates to terminal velocity) are taken into account, although their effects are small.

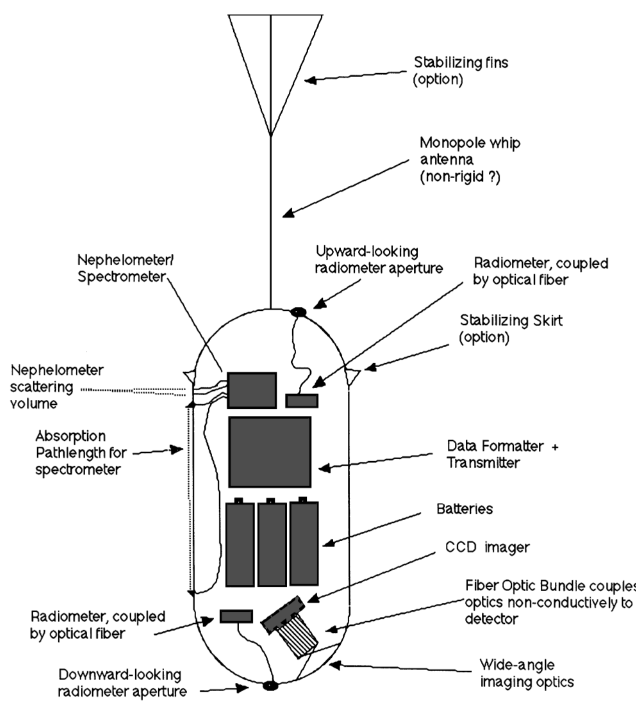

Fig. 1 Cross-sectional sketch of candidate microprobe configuration. The fins and/or skirt provide stabilization and spin control. The probe can be considered as a block of insulating material populated by electronic boxes.

Received June 12, 1997; revision received Sept. 30, 1997; accepted for publication Nov. 15, 1997.

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The zonal (east–west) winds vary from near zero at the surface to about 80 m/s at 60 km (Ref. 9). During a 0.4-h probe descent, the aerobot would be displaced by some 115 km, whereas the probe (convolving its altitude history with the wind profile) would move only 25 km. Thus, the aerobot would move from directly overhead at release to about 50 deg from vertical at surface impact. Such a geometry is compatible with a simple monopole antenna with a toroidal radiation pattern: The receiver on the aerobot then will be in the center of the lobe at the end of descent when the communication range is longest. However, the slant range to the aerobot and its aspect angle increase significantly for longer descent times (Fig. 2).

The probe geometry is defined by a radius and a slenderness ratio r with the length/radius equal to $1 + r$, i.e., a sphere has $r = 1$, whereas a bomb shape 2 diameters long has $r = 3$. It is assumed that the payload (the scientific instruments plus all subsystems required for data transmission to the aerobot, including structure) has maximum allowable densities of 560 and 1000 kg/m^3. (The Huygens probe has a density of about 300 kg/m^3; Galileo and Pioneer Venus, both with pressure hulls, are around 1000 kg/m^3.) Among phase-change materials considered are lithium nitrate trihydrate ($\rho = 1550$ kg/m^3, phase transition temperature 303 K, $c_p = 3000$ J/kg · K; $L = 296$ kJ/kg) and gallium ($\rho = 6000$ kg/m^3, phase transition temperature 302 K, $c_p = 374$ J/kg · K; $L = 80$ kJ/kg). Between 273 and 373 K, these materials absorb 924 and 705 kJ/liter, respectively. Two insulation options are considered (note that conventional porous insulation and multilayer insulations will not work in the high-pressure, high-temperature atmosphere): unfoamed polystyrene ($\rho = 1050$ kg/m^3, $k = 0.08$ W/m · K) and a Dewar with fumed silica filler (to provide structural support and block radiative heat transfer[10]) with assumed properties ($\rho = 200$ kg/m^3, $k = 0.01$ W/m · K). Note that the heat leak through electrical connectors, lenses, and other penetrations have not been considered; their effects are assumed to be lumped with these modest performance parameters.

Results and Conclusions

Viable configurations (those that reach the surface without their internal temperatures exceeding 373 K) are presented in Table 1. The payload mass, slenderness ratio, and insulation type were specified, and the vehicle radius and insulation thickness was varied by trial and error (by increments of 1 and 0.5 mm, respectively) to achieve a required payload density (±2%) and an acceptable temperature at impact.

It is seen first that the payload, i.e., noninsulation, mass for a viable microprobe is a small portion of the total mass for small (0.2-kg) payloads. The mass fraction increases significantly for more massive payloads. It is essentially impossible to achieve a vehicle that will reach the surface of Venus with a tolerable internal temperature for less than 0.35 kg, regardless of descent duration.

A second, perhaps suprising, conclusion is that a slenderness ratio of 1 (a sphere) appears to be the best (most mass-efficient) geometry for a Venus microprobe. It seems that the additional insulation thickness required to compensate for the larger surface area of a slender probe outweighs the effect of the shorter descent time. It is notable, however, that the vehicle mass is fairly insensitive to increases in slenderness ratio, and so the shape may be tuned to fit the desired descent time with little mass penalty. This is fortunate because the ballistic coefficient required for a rapid descent (< 0.5 h)

Table 1 Selected configuration cases

Case[a]	Payload mass, kg	Slenderness ratio	Vehicle radius, cm	Insulation thickness, mm	Ballistic coefficient, kg/m^2	Descent time, h	Internal temperature at impact, K	Vehicle mass, kg
A1	0.2	1	6.4	20	182	0.76	368	1.28
A2	0.2	3	5.5	18.5	293	0.57	362	1.56
A3	0.2	5	4.5	17	409	0.49	360	1.56
A4	0.2	10	3.7	15	655	0.39	361	1.69
A5	0.6	1	8.3	19	179	0.74	367	2.23
A6	0.6	3	6.3	16	304	0.56	373	2.27
A7	0.6	5	5.5	15	420	0.48	369	2.39
A8	0.6	10	4.6	14	689	0.38	362	2.75
A9	2.0	1	11.2	17	203	0.69	364	4.80
A10	2.0	3	8.5	15	356	0.52	362	4.86
A11	2.0	5	7.4	14	487	0.45	362	5.03
A12	2.0	10	6.0	12.5	771	0.38	364	5.32
B1	0.2	1	6.5	20	51	1.40	374	0.40
B3	0.2	5	4.4	16	118	0.93	368	0.44
B4	0.2	10	3.7	15	187	0.75	366	0.48
B5	0.6	1	7.7	13	371	1.19	371	0.79
B8	0.6	10	4.2	10	253	0.64	369	0.84
B9	2.0	1	10.2	7	111	0.94	367	2.18
B12	2.0	10	5.3	5	414	0.49	370	2.19
AP1	0.2	1	5.6	20	173	0.75	366	1.02
AP4	0.2	10	3.3	15	676	0.38	357	1.38
AP5	0.6	1	7.0	18	195	0.71	366	1.76
AP8	0.6	10	3.9	13	710	0.37	369	2.03
AP12	2.0	10	4.2	11	848	0.34	367	4.00
BP1	0.2	1	5.4	18	60	1.28	372	0.33
BP4	0.2	10	3.1	13	207	0.70	371	0.38
BP5	0.6	1	6.2	10	96	1.01	360	0.70
BP8	0.6	10	3.4	8	335	0.54	358	0.73
BP12	2.0	10	5.0	11	624	0.39	362	2.08
AG5	0.6	1	8.1	16	209	0.68	370	2.58
AG8	0.6	10	4.5	12	776	0.35	368	2.96
BG5	0.6	1	7.0	5	136	0.85	376	1.26
BG8	0.6	10	3.7	4	495	0.44	370	1.28
AL5	0.6	1	8.4	14	188	0.71	356	2.50
AL8	0.6	10	4.8	12	738	0.36	339	3.20
BL5	0.6	1	7.4	3	120	0.90	338	1.24
BL8	0.6	10	3.8	3	462	0.46	302	1.26

[a] Ax denotes polystyrene insulation, no phase-change material, and payload density of 560 kg/m^3; Bx denotes advanced (Dewar) insulation, no phase-change material, and payload density of 560 kg/m^3; APx, BPx are the same as Ax, Bx but with payload densities of 1000 kg/m^3; ALx, BLx are the same as Ax, Bx but with lithium nitrate trihydrate equal to the payload mass added; AGx, BGx are the same as Ax, Bx but with gallium equal to the payload mass added.

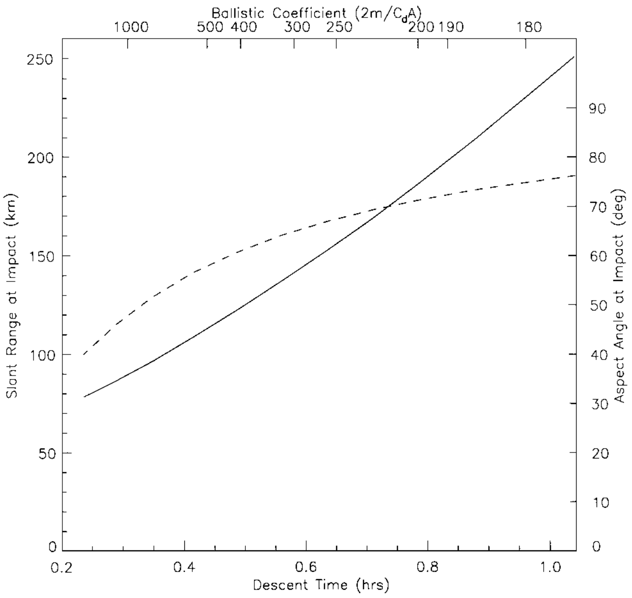

Fig. 2 Slant range (——) and aspect angle (– – –) to balloon at microprobe impact as a function of descent time and equivalent ballistic coefficient. Link geometry degrades as descent time increases.

is not compatible with a spherical geometry for these small probes. Practicable descent durations require vehicle masses of ~1 kg.

Significant mass savings are obtained by the use of advanced insulation. However, a detailed thermal analysis needs to be made to determine heat leaks through connectors, etc., because these may compromise the performance of good insulation. Similarly, if the internal power dissipation (assumed here to be 2 W, compared with typical 10- to 100-W heat leak through the insulation at impact) were to be more than a few watts, high-performance insulation would be somewhat less useful.

Phase-change materials are not mass efficient but can augment thermal margin to make a robust design (although they are no better in this respect than increasing the insulation thickness, if that is a viable option subject to the internal-power-dissipation caveat). The use of gallium (and, by implication, any mass ballast) as a phase-change material offers neither mass savings nor significant decrease in descent time because its poorer heat absorption characteristics compared with those of lithium nitrate require thicker insulation.

An increase in the packaging density of the payload itself offers substantial mass savings. Because of the mass savings (typically 25%), the denser payload does not significantly reduce the descent time.

In summary, if high-performance insulation is available and the relay link to the aerobot can tolerate ranges of 150 km and aspect angles of 70 deg, i.e., descent times of ~0.5 h, ~l-kg Venus microprobes are viable with payloads of 0.6 kg or more. Tighter relay link constraints would push to more massive probes.

Further study into Venus microprobes would require careful evaluation of the aerodynamic performance of the selected configuration (the $C_D = 0.3$ assumed here is somewhat arbitrary, and appropriate shaping of the vehicle might reduce this considerably), the relay link performance (which couples with the antenna design and the transmit power and data rate), and the thermal performance of the insulation with realistic penetrations, taking into account its nonsteady warming, i.e., its nonzero thickness. A further mission element that could be optimized is the release altitude: If the relay link is a strong driver, the aerobot might cruise at a lower altitude (where the winds are weaker and thus the range at impact will be smaller). This would permit the use of a smaller balloon because the ambient air density is higher, although the ambient temperature is also higher.

Acknowledgments

This work was supported by a contract from the Jet Propulsion Laboratory. The author acknowledges useful discussions with Jack Jones, Kerry Nock, and Jeff Hall.

References

[1]Cutts, J. A., Nock, K. T., Jones, J. A., Rodriguez, G., and Balaram, J., "Planetary Exploration by Robotic Aerovehicles," *Journal of Autonomous Robots*, Vol. 2, No. 4, 1995, pp. 261–282.

[2]Nock, K. T., Jones, J. A., and Rodriguez, G., "Planetary Aerobots: A Program for Robotic Balloon Exploration," AIAA Paper 96-0355, Jan. 1996.

[3]Heun, M. K., and Jones, J. A., "Gondola Design for Venus Deep-Atmosphere Aerobot Operations," AIAA Thermophysics Conf. (submitted).

[4]Jurgens, R. F., "High-Temperature Electronics Applications in Space Exploration," *IEEE Transactions on Industrial Electronics*, Vol. IE-29, No. 2, 1982, pp. 107–111.

[5]Schock, A., "Integration of Radioisotope Heat Source with Stirling Engine and Cooler for Venus Internal-Structure Mission," 44th Congress of the International Astronautical Federation, Paper IAF-93-R.1.426, Graz, Austria, Oct. 1993.

[6]Borucki, W. J., "Estimate of the Probability of a Lightning Strike to the Galileo Probe," *Journal of Spacecraft and Rockets*, Vol. 22, No. 2, 1985, pp. 220, 221.

[7]Lorenz, R. D., Gautier, D., and Lebreton, J.-P., "A Deep Jupiter Probe," 44th Congress of the International Astronautical Federation, Paper IAF-93-Q.5.412, Graz, Austria, Oct. 1993.

[8]Seiff, A., Schofield, J. T., Kliore, A. J., Taylor, F. W., Limaye, S. S., Revercomb, H. E., Sromovsky, L. A., Kerzhanovich, V. V., Moroz, V. I., and Marov, M. Ya., "Models of the Structure of the Atmosphere of Venus from the Surface to 100 Kilometers Altitude," *Advances in Space Research*, Vol. 5, No. 11, 1985, pp. 3–58.

[9]Kerzhanovich, V. V., and Limaye, S. S., "Circulation of the Atmosphere from the Surface to 100 km," *Advances in Space Research*, Vol. 5, No. 11, 1985, pp. 59–83.

[10]Reiss, H., "Evacuated, Load-Bearing Powder Insulation for High Temperature Applications," *Journal of Energy*, Vol. 7, No. 2, 1982, pp. 152–159.

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Numerical Predictions of Hypersonic Flow over a Two-Dimensional Compression Ramp

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Introduction

HYPERSONIC flows over compression ramps, for example, as used for control purposes on re-entry vehicles, feature a complex structure of interacting shock waves and include shock-wave/boundary-layer interactions, which are known to induce flow separation. Subsequent flow reattachment on the ramp causes high heat fluxes, which must be predicted accurately to preserve the thermal and structural integrity of the vehicle.

Heat transfer rates from laminar, two-dimensional numerical calculations of a Mach 6.85, perfect-gas flow past a typical compression ramp are presented. Numerical results are compared with experimental data obtained from the light-piston, isentropic-compression hypersonic wind-tunnel facility at the University of Southampton.[1,2] The experimental model consisted of a flat-plate section 155 mm long followed by a ramp 51 mm long. The tunnel uses nitrogen and operates at a working-section Mach number of 6.85, a stagnation temperature ≈ 600 K, and a freestream unit $Re \approx 2.45 \times 10^6$ m^{-1}.

Presented as Paper 96-4597 at the AIAA 7th International Space Planes and Hypersonic Systems and Technologies Conference, Norfolk, VA, Nov. 18–22, 1996; received Jan. 10, 1997; revision received Nov. 29, 1997; accepted for publication Dec. 1, 1997.

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