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Multiplying Mars Lander Opportunities with MARS$_{DROP}$ Microlanders

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ABSTRACT

From canyons to glaciers, from geology to astrobiology, the amount of exciting surface science awaiting us at Mars greatly outstrips available mission opportunities. Based on the thrice-flown Aerospace Corporation Earth Reentry Breakup Recorder (REBR), we present a method for accurate landing of small instrument payloads on Mars, utilizing excess cruise-stage mass on larger missions. One to a few such microlanders might add 1-5% to the cost of a primary mission with inconsequential risk. Using the REBR and JPL Deep Space 2 starting points for a passively stable entry vehicle provides a low mass and low ballistic coefficient, enabling subsonic deployment of a steerable parawing glider, capable of 10+ km of guided flight at a 3:1 glide ratio. Originally developed for the Gemini human space program, the parawing is attractive for a volume-limited microprobe, minimizing descent velocity, and providing sufficient remaining volume for a useful scientific payload. The ability to steer the parawing during descent opens unique opportunities, including terrain-relative navigation for landing within tens of meters of one of several specified targets within a given uncertainty ellipse. In addition to scientific value, some Mars human exploration Strategic Knowledge Gaps could be addressed with deployment of focused instruments at multiple locations.

INTRODUCTION

Building from the MARS$_{DROP}$ architectural concept,[1,2] we have derived a preliminary non-optimized existence-proof-level mission concept within the constraints and capabilities of the high-heritage REBR-based entry vehicle.[3] As the outcome of a Science Mini-Workshop held in April, we selected a scientific payload consisting of a miniaturized Tunable Laser Spectrometer (TLS) to provide geographic diversity of methane abundance measurement to complement the similar but larger instrument aboard *Curiosity*. We included a set of meteorology instruments that also serve to help interpret TLS results. Finally, the video camera used for descent guidance, and a similar camera that pops up after the lander deploys on the surface, can afford unparalleled views of the terrain geomorphology and appearance.

While landed mass is typically tightly-limited by lander size limitations, cruise stages on most lander missions have typically had hundreds of kilograms of excess capability for delivery to Mars. In some ways like CubeSats utilizing excess launch vehicle capability to low Earth orbit, we propose that excess Mars injection capability can be utilized to multiply the number of

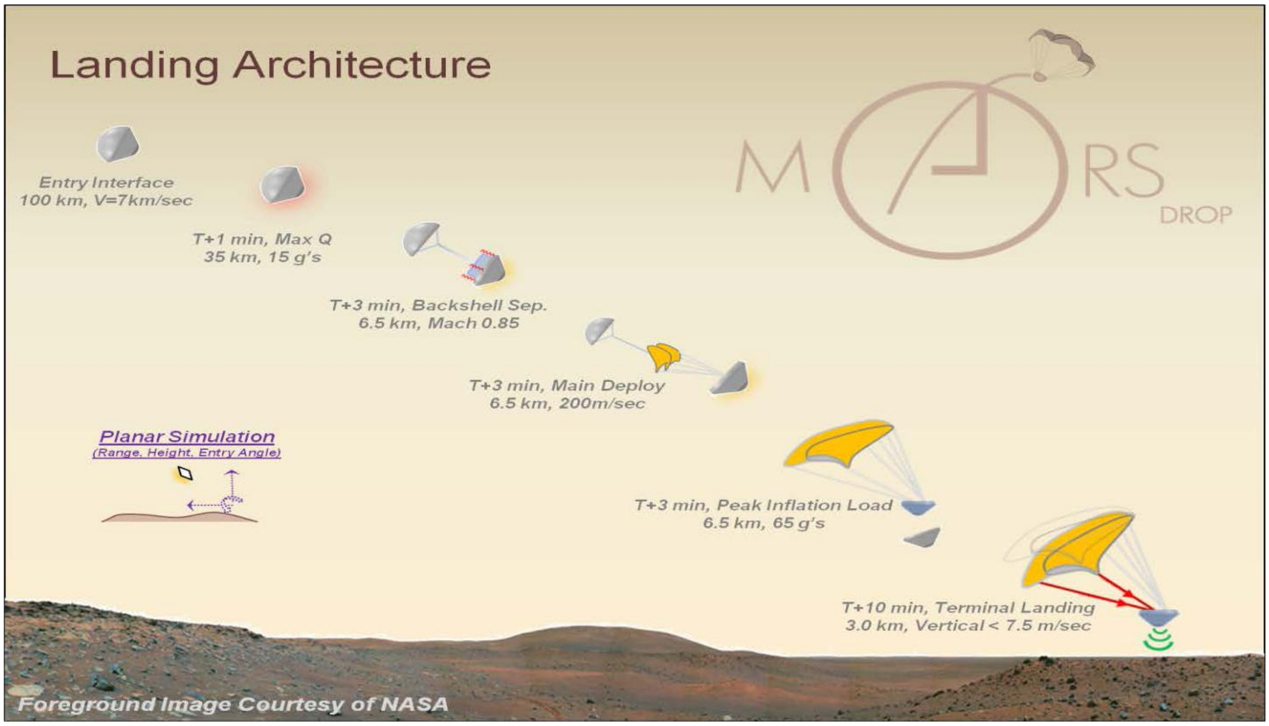

Figure 1: MARSDROP events from entry to landing.

landers launched with each Mars mission. While orbiter missions may have less excess capability than landed cruise stages, MARS$_{\text{DROP}}$ lander variants may be dropped off before Mars orbit insertion for their own atmospheric entry (Figure 1), or carried aboard an orbiter for a modest mass of 10-20 kg including their own de-orbit propulsion.

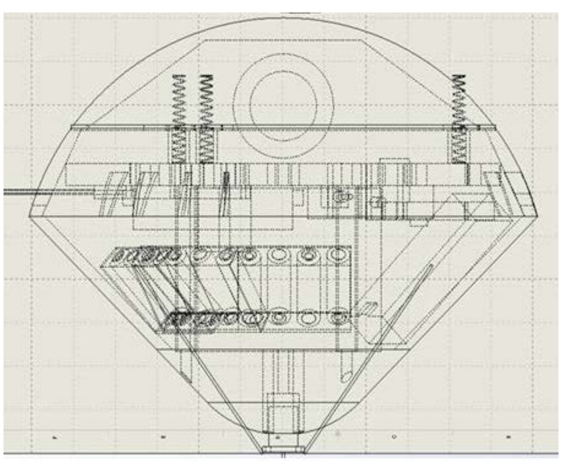

Figure 2: REBR Aeroshell shown with direction of entry downward. Payload is in the bottom; parawing under deployable backshell on top.

The entry vehicle we envisage for initial missions is identical to the Aerospace Corporation's Reentry Breakup Recorder (REBR; Figure 2), with a 30 cm diameter, and ~3 kg entry mass. Having entered Earth's atmosphere from orbit at ~9 km/sec, we believe it is qualified to survive Mars atmospheric entry from hyperbolic trajectories at ~7 km/sec.

The entry vehicle enters with its conical primary heat shield forward, and hemispheric backshell containing the parawing and its deployment mechanism. When carried as a secondary payload, MARS$_{\text{DROP}}$ landers will of course not be dictating the trajectory of approach to Mars, and may only be given a specific deployment time and kickoff velocity to ensure separation from the primary. In our architecture, we plan a *Backpack* that mounts on the aft end of the backshell. This Backpack contains attitude determination and control components from the CubeSat repertoire, enabling it to null tipoff rates resulting from jettison from the primary, seek and acquire spatial reference, re-orient to optimal entry interface orientation, spin up to potential ~2 rpm rotation for enhanced hypersonic stability, and then separate itself upon actuation of a g-switch that senses sustained atmospheric slowing. Optionally, a thrusting capability may be added to the Backpack enabling a few to 100's of m/sec propulsion from release to target entry conditions.

We also envisage options where several, even up to a few dozen, MARS$_{DROP}$ microlanders could go to Mars on a dedicated carrier whose trajectory would be specific to the multi-microlander mission.

With or without Backpack propulsion, landing dispersion ellipses will be several times the gliding range of a parawing deployed at 5 – 10 km above Mars' surface. Our concept of operations accounts for this by having a detailed image library stored aboard each microlander of its possible target area on Mars. Within this library, a prioritized number of desired landing sites are specified. Within less than a minute of parawing deployment, the highest-priority location within achievable glide distance at a 3:1 glide ratio can be identified autonomously onboard, and the terrain relative navigation (TRN) software onboard actuates control lines on the parawing to steer to target. Increasingly finer resolution image maps in the onboard library are georeferenced and stepped through as the gliding vehicle approaches its target, enabling a landing within 10s of meters where high-resolution orbital imagery is available, such as from the HiRISE camera aboard Mars Reconnaissance Orbiter.

Figure 3: Scaled parawing and microprobe. (Upper left photo: NASA)

Depending on parawing deployment altitude, glide times on the order of 10 – 20 minutes are expected. Since the TRN software is using video camera imagery to steer, a recording of this video can be easily taken on the fly into a bulk memory. With temporal and spatial sampling, this video can be "played back" and reassembled on the ground with varying degrees of fidelity based on available downlink volume capacity through orbiting relay assets. Descent video, TLS, and other data can be ranked for downlink priority to create an information-rich data stream based on the instrumentation and measurement priorities of the MARS$_{DROP}$ mission.

Use of a parawing (Figure 3) enables significant lift from flight through even Mars' thin atmosphere to be used to slow descent. While landing can be "soft" compared to that of a penetrator or parachute-only descent, the 7 m/sec descent velocity component of a 3:1 glide ratio at ~20 m/sec still represents a major impact event for the equipment involved. As anyone knows who has dropped a cellphone, or had a GoPro^{tm} video camera drop off a fast-moving all-terrain vehicle, or crashed a model aircraft, scaling laws enable small electronic packages to withstand 100s of g deceleration loads. We have designed the heat shield to also limit deceleration loads from an oblique, bouncing tumble to a halt across the typical surface of Mars.

From a tumble of course, the microlander could come to rest either on its cone or its flat "topside." We have therefore incorporated spring-driven panels, released sequentially, to ensure self-righting for solar panel, telecom, and instrument visibility after landing.

Once safely landed, telecom capability is provided by a UHF link through Mars-orbiting assets.

Once on the surface, one of the biggest challenges is surviving the large day/night temperature excursions. Our small package can be well-insulated using aerogel and multi-layer insulation, such that ~1-2 W is capable of keeping all sensitive components above -40 C minimum operating temperature. Batteries are sized to accept excess power from solar panels during the day, and meter out heater power at night within an acceptable depth-of-dischange. This battery mass is mounted near the apex of the inside of the entry heat shield to move the center of mass as far forward as possible for entry stability. If a CG can be achieved <25% of the axial length of the entry body, this could eliminate the need for a spinning entry, but this is a very challenging design parameter, so we have assumed a spinning entry at a spin rate to be determined from future analysis.

Using the kinds of inexpensive componentry we envisage, limited analysis, and iterative ground testing cycles of components and system, we have a target surface lifetime of 90 sols. However, nothing in the landed system has any design attribute that locks in failure at this duration. Dust on the solar panels, radiation exposure, and disconnection resulting from thermal cycle-induced fractures represent the most likely failure modes considered so far. Thus we expect weeks to a year of operation on Mars' surface.

Another advantage of our small system is the ability to apply dry heat microbial reduction and other techniques up to and including the level of sterilization. If achievable, this enables consideration of $\text{MARS}_{\text{DROP}}$ landers as a vehicle for carrying out investigations in sensitive locales on Mars under the most stringent planetary protection requirements to prevent forward contamination. Once sterilized (and verified sterile), a microlander may be placed into a sterile, tight-fitting plastic bag for handling on the ground up to and including attachment to its carrier. Upon Mars entry, the plastic bag will simply burn off, leaving the sterile lander inside to go to places like the polar fringes, potential wet sites like recurring slope lineae, and potentially even martian caves.

An informal survey of members of the Mars science community indicates that a variety of qualified instrumentation can be built within the ~1 kg science payload allocation available for the REBR-heritage entry vehicle.[2] Modest increases in entry vehicle diameter enable higher science payload mass, perhaps up to 2+ kg, while still maintaining a ballistic coefficient such that subsonic parawing deployment is possible sufficiently above Mars' surface.

While a detailed grass-roots cost estimate remains to be performed, some existing metrics appear applicable. We envision this system as initially being of NASA Technology Demonstration mission or Class D-type reliability, where failure presents no threat to the primary mission. Some of the componentry envisaged in the proof-of-concept design undertaken so far is of CubeSat or similar heritage, such as the UHF transceiver, batteries, solar panels, cameras, computing and memory hardware, and ADCS for the Backpack, for which no single item appears likely to cost >$100K. To cover engineering costs a conservative metric used in planetary mission science instrumentation, is that it costs <~$1 M/kg. While based on instrument experience, this metric is probably more broadly applicable because the $\text{MARS}_{\text{DROP}}$ system is much like an instrument taken together. JPL's INSPIRE CubeSat, with all non-recurring engineering, test, similar parts count and somewhat less complexity cost <FY14$6M.

Deployment hardware needing the carried aboard Class B host missions will of course require greater analysis and testing, so this hardware might cost as much as $2M/kg and weigh up to 5 kg.

Thus we estimate that the total cost for a first $\text{MARS}_{\text{DROP}}$ technology/science demonstration mission, including costs borne for carriage aboard a high-reliability host mission, can be contained within FY15$20M. Subsequent missions could cost much less, probably <$10M for even two or three copies of the $\text{MARS}_{\text{DROP}}$ hardware and reasonable science instrumentation.

SCIENCE & HUMAN EXPLORATION

This microlander capability enables a new class of Mars science investigations by being able to deliver multiple miniaturized instruments to the most desirable locations for network science, and even the search for biosignatures at different locations.[4] The relatively low cost of each probe, combined with an ability to send multiple, redundant probes enables access to regions of Mars deemed scientifically interesting, but too risky for a large, expensive lander.

The ability to steer to targets of interest during the gliding phase opens up a wide variety of enticing locations to specialized instruments, including: a) within the canyons of Valles Marineris, b) lava flows in volcanic regions such as Tharsis, c) water-transported sediment deposits in alluvial fans and deltas[5] (such as Eberswalde[6]), d) proposed glaciers[7] and ice-rich terrains, e) the subliming swiss cheese terrain of the southern polar cap,[8] f) water carved terrain from catastrophic floods (e.g., circum-Chryse outflow channels[9]), g) melt water along the margins of the north polar cap,[10] h) potential geysers that create spider-terrain in high southern latitudes,[11, 12] i) bottoms of fresh impact crater sites[13,14] with high organic preservation potential, j) surface "windows" that serve as skylights opening to subsurface 'caves',[15] k) gullies,[16] and l) other surface changes[17,18] which may be signs of seasonal subsurface water running down crater walls.

Beyond science targets, piggyback microprobes, deployed several at a time, offer a new tool for human precursor missions to address several recommendations stemming from the Mars Exploration Program Advisory Group ("Findings and Strategic Knowledge Gaps"):[19]

a. *Finding #5: "Several landed measurements need to be made simultaneously with orbital measurements"*
b. *SKG Group A.2: "The atmospheric models for Mars have not been well validated due to a lack of sufficient observational data, and thus*

our confidence in them (for use in mission engineering) is significantly limited."

c. *SKG Group B.1: "Lower Atmosphere: We do not have sufficient Martian atmospheric observations to confidently model winds, which significantly affect EDL design…"*

We first examine the scientific motivation for a Mars microprobe development, followed by plausible capabilities and development gaps that need to be filled to enable this architecture. Then, as a proof of concept mission, we provide a conceptual outline of a first science demonstration that could be ready by 2019 (or later Mars opportunities) as a daughter spacecraft carried with a primary Mars mission.

We examine the useful science payload capability relative to very specialized instruments to assess the relative microprobe merits in tackling the most stimulating questions about Mars; e.g., what are the organic constituents, if any, dissolved in briny meltwater, or, what is the sediment history of flow down the wall of Newton or Horowitz crater where changing flow features have been identified in Mars Reconnaissance Orbiter HiRISE images taken months apart.[20] The following provides a representative sampling of compelling missions that could be undertaken with miniature imagers, weather instruments, and seismic probes.

Some of the more intriguing recent observations on Mars suggest the potential presence of liquid water, especially given the significance of water for habitable environments. Specific sites include gullies and recurring slope lineae (RSLs) on crater walls (Figure 4) and potential geysers in the south polar region. Surface-based, time-lapse images can monitor terrain change and provide critical information on timing, surface temperature, flow properties, and liquid composition to evaluate mechanisms.

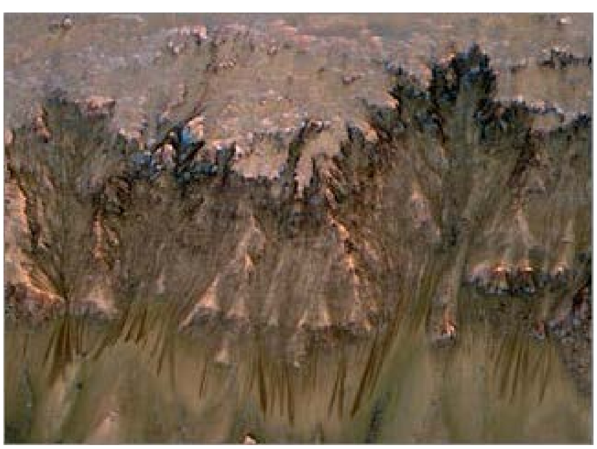

Figure 4: Seasonal features, Newton Crater. (NASA)

In the polar regions, there are a number of scientifically compelling targets. The polar caps preserve a record of past environments in the ice layers.[21,22] Additionally, the polar regions are actively undergoing change, from frost-dust avalanches,[23] and subliming carbon dioxide ice.[24] Here again, high resolution time-lapse images can be informative in evaluating the mechanisms.

The Valles Marineris canyon system spans ~5000 km (~3000 miles) and is four times deeper than the Grand Canyon. Within the canyons are a variety of distinct rock layers.[25] In addition to being an exciting and visually stunning site, lakes may have persisted within the canyon, particularly at southwestern Melas Chasma where layered beds are inferred paleolacustrine deposits with high biosignature potential.[26] Delivery of a flagship mission to Valles Marineris is often problematic due to the lack of large, flat surfaces to land and concerns over model-derived winds; however, it is a rich series of targets that is accessible with $\text{MARS}_{\text{DROP}}$. Stratigraphic relationships and compositional data can be determined with spectral images, enhancing our understanding of former environments.

To date, no missions have landed in volcanic terrains on Mars, despite the fact that the northern hemisphere is dominated by the Tharsis and Elysium volcanic provinces. Flow rheology is imprinted in the surface morphology, and ground-based observations would be important to constrain lava models.[27]

Furthermore, the view from the base of one of the massive Martian shield volcanoes would be phenomenal!

Beyond these sites, there are a number of locations on Mars that have unusual landforms or terrains that would benefit from higher resolution, multispectral images and surface-based investigation. Interesting terrain types include: a) dune gullies[25,28] and b) lobate aprons[29] and lineated valley fill potentially associated with glaciers.[30] With a descent imager, intriguing targets would include peering into 'caves' or fresh impact craters (some of which may have exposed ice).

Atmospheric phenomena can also be investigated with a $\text{MARS}_{\text{DROP}}$ payload, depending on the time of arrival. For example, there are well-known periods of weather phenomena (spiral storms) in the north polar region[31] and dust devil activity in the southern mid-latitudes.[32] Surface-based observations of these phenomena, especially with greater temporal coverage of these events, would create a more continuous record than orbital assets can provide.

With multiple MARS$_{DROP}$ payloads, there is the potential to create a distributed seismic network. Taking advantage of periodic impact events (several occur annually), the interior structure of Mars can be probed. This data would complement forthcoming data from the InSight mission and the ground penetrating radar instrument (RIMFAX) on the Mars 2020 rover.

By piggybacking on a primary payload, MARS$_{DROP}$ expands the scientific return of the mission, enabling data collection at companion or complementary landing sites. For example, MARS$_{DROP}$ could be delivered to the source region (e.g., drainage basin), permitting surface-based observations at locations the rover would never explore, providing the capability of directly establishing provenance of sediments at the rover landing site. Alternatively, the payload could provide reconnaissance data of the ultimate rover destination, enhancing the science team's understanding of initial observations by broadening the geologic context of the region. If MARS$_{DROP}$ were sent to alternative locations, it could provide critical data (e. g., rock abundance, surface roughness, traversibility, etc.) for future landing site certification not well constrained in existing orbital attainable data.

In summary, a microprobe approach could deliver scientific payloads to sites of scientific interest that are not available or reachable by larger rover payloads, either because of the high risk of the landing location, or because of the specialized/limited nature of the investigation, or because of the need to deliver multiple, spatially dispersed probes.

SYSTEM & OPERATIONS CONCEPT

Entry, Descent, and Landing

The MARS$_{DROP}$ lander is expected to be deployed from the primary mission's cruise stage at an altitude >100 km from the Mars surface at a Mars-relative velocity of approximately ~7 - 7.5 km/sec. The probe will initially have its small Backpack attached that provides attitude control, using its Blue Canyon Technologies' XACT unit (star tracker, reaction wheels, IMU, etc.) with batteries and flight computer. Over a period of a few minutes, this backpack will first provide the correct orientation, and then impart the required spin (~2 rpm) to ensure stability throughout the reentry phases, as in Figure 5. Note that the Backpack is currently required to impart spin because in the current design the center of mass is not sufficiently close to the nose (<25% of the probe axial length) to ensure a passively stable entry. A slightly larger Backpack could be made to carry propulsion to enable orbital changes prior to entry. Prior to entry, the Backpack is jettisoned.

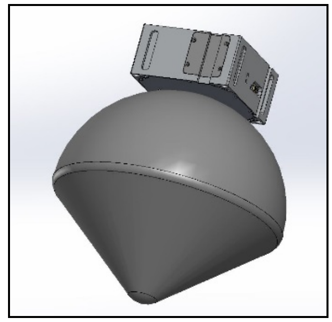

Figure 5: Post-deployment configuration with Backpack

During entry, the probe experiences maximum deceleration of ~12 g and heating ~150 W/cm^2 at an altitude of approximately 40 km.[33] Following entry, the parawing is deployed to enable gliding and a controlled descent, as in Figure 6. The navigation camera is also deployed at this point, centered along the glide path, which enables imaging the ground and horizon during descent, as in Fig. 7. The camera combined with onboard micro-IMU data and navigation algorithms helps determine the probe's position and altitude to 1% of relative altitude during the descent.

Figure 6: Parachute deployment controlled by probe for the descent to a desired target.

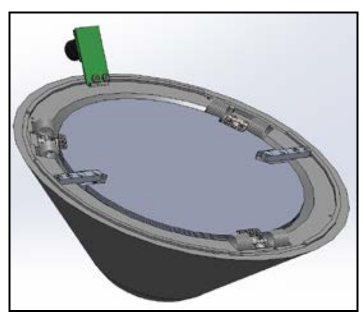

Figure 7: MARS$_{DROP}$ with descent navigation camera deployed to guide the descent.

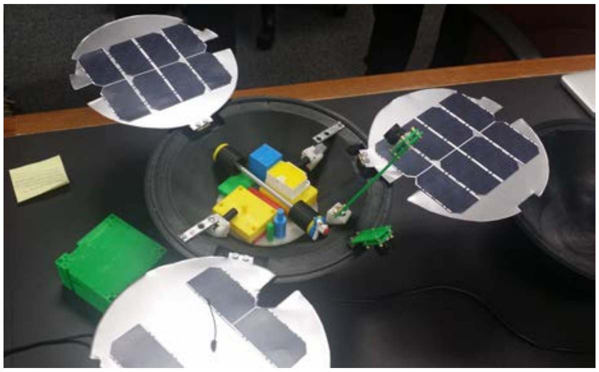

Figure 9: MARS$_{DROP}$ full-size 3D print of landed configuration.

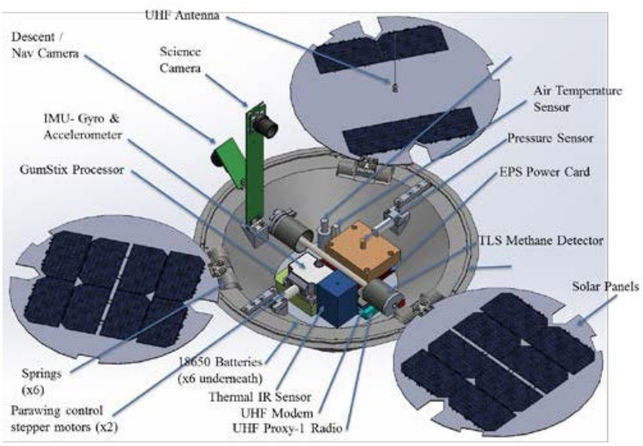

Figure 8: Landed MARS$_{DROP}$ Configuration.

During descent, on-board navigation algorithms control actuators that pull on wingtips to turn (one wingtip) or change glide angle (both wingtips). The navigation system enables the MARS$_{DROP}$ probe to glide to pre-selected landing sites. Normally a ~3:1 glide ratio is achieved, however a lower glide ratio may be employed, along with turning, to target a desired landing locations.

The MARS$_{DROP}$ lander is expected to reach the Mars surface approximately 10 minutes after parawing deployment. The expected speeds are approximately 20 m/sec total, ~7 m/sec vertical, 18.7 m/sec horizontal. While a flare may be challenging to achieve, we are currently exploring the potential and have included the required hardware in our baseline design, based on the assumption that sufficient accuracy may be achievable from successive images forming stereo pairs taken along a baseline estimated from flight parameters. Without a flare, we expect the probe to experience approximately 300-500 g maximum deceleration on landing, and to roll and bounce prior to reaching its final resting position. If the vehicle strikes a flat boulder face-on, design loads will be exceeded, but under most scenarios we envision at scientifically desirable landing sites, a skid-and-tumble across <100 m has a high probability of being survivable for our ruggedized payload.

Because of uncertainty in the landing orientation, we've designed the deployable petals with springs having sufficient stored strain energy such that they will "right" the lander regardless of landing orientation and expose the "petals" to sky, see Figure 9 and Figure 8. To reduce the chance of failure, solar cells and an alternate antenna can also be placed on the "under" side of the petals just in case the "righting" scheme fails.

The descent camera is expected to break off during landing, before the circular petals deploy.

Post- landing, the MARS$_{DROP}$ lander will have normal lander-type operations. The lander will largely operate autonomously on the Mars surface. It will collect science and engineering data according to scientific requirements and available energy resources and collect solar power and store energy in on-board batteries, particularly to support heaters for nighttime survival and low-powered operations. The lander will also transmit science, navigation, and engineering data to an orbiter when it is in view and available**.**

Configuration

The MARS$_{DROP}$ lander is designed with a small spacecraft (i.e., CubeSat/U-class) design philosophy. The architecture is designed to be lean, low-cost, and multi-functional. For example, the batteries are used both to lower the center of mass (CM) and store energy for nighttime thermal needs, the cameras are used for navigation as well as geological science, the aero shell acts both as a heat shield and crushable impact absorber to reduce the forces experienced at impact, and the pressure sensor provides both altitude information for the descent and weather data on the surface. We describe the representative configuration in Figure 9 with the set of science instruments selected for the first demonstration. Future variations may include a diverse suite of instruments but will likely use similar spacecraft components.

Table 1: Master Equipment List (MEL) for components that enter. Suppliers shown only for proof-of-concept; no selection is represented. Entry mass (<3.5 kg) consistent w/ mass from Aerospace Corp. REBR flights from Earth orbit.

Subsystem	Components	Mass	Power	Heritage / Supplier
Entry & Descent	Aeroshield (1,200 g), Parawing (400 g), Stepper motors (2 x 10 g)	1,620 g	-	REBR/Aerospace Corp.
Payload	Methane Detector (TLS)	150 g	0.67 W	MSL/ JPL
	Pressure, Air Temperature, and Humidity Sensors	113 g	0.43 W	MSL/ JPL, various
Payload/Navigation	Descent/Geology Camera (2 x 40g)	80 g	1 W	None*/ Aptina
Navigation	IMU (Gyro & Accelerometer)	10 g	0.1 W	Variable/ Blue Canyon Tech.
Power	Body-Mounted Solar Panels (20 x UJT Cells)	40 g	-	Variable/ Spectrolab
	Batteries (6x18650 Li Ions, ~16 Whr)	270 g	-	INSPIRE/ Panasonic
	Electric Power System & Battery Board	80 g	-	RAX & INSPIRE/ JPL
Computing & Data Handling	Gumstix Flight Computer & Storage	10 g	0.5 W	IPEX/ Gumstix
Telecom	UHF Proxy-1 Radio	50 g	2 W	Variable/ JPL
	UHF Low Gain Antenna (Whip)	5 g	-	Variable/ JPL
Mechanical & Others	Shelf (68 g), Brackets (26 g), Wing Actuator (19 g), Springs (48 g), Hinges (7 g), Fasteners (20 g), Harnessing (50 g), and others (20 g)	256 g	-	Variable/ JPL
Thermal	Heaters (3 x 50 g), Aerogel (10 g)	160 g	2 W	Variable/ JPL
TOTAL	Total No Margin/ With 20% Margin	**2.8 kg/ 3.4 kg**	~3 W (avg)	-

The design leverages CubeSat components, many of which have been flown in previous Low Earth Orbit (LEO) missions, or will fly in upcoming deep space CubeSats, such as INSPIRE, MarCO, LunarFlashlight, and NEAScout; see Table 1. The total mass with 20% margin approximates the masses demonstrated in the Aerospace Corp. REBR missions (3.5 kg). Nearly all components are existing hardware elements, many of which the team physically has and works with regularly. The Backpack, mechanical interfaces, and spring for jettison are estimated as an additional 0.7 kg, or 0.9 kg with 30% margin. We also expect that after being sterilized, the probe will be contained within a sterilization bag (<100 g) to maintain its sterility during integration and transportation to Mars. The bag will burn up during entry. The payload, described in “Instrumentation”, is <0.35 kg, but future MARS$_{DROP}$ designs could accommodate heavier/larger payloads.

To control the parawing, we anticipate using a small stepper motor to manipulate the outer tip lines. Line loads are low enough so that a small motor can easily fit in the available space. We envision a simple motor controller that counts steps along with a simple home position indicator to maintain knowledge of motor position.

The MARS$_{DROP}$ configuration is an initial point design using a 300 mm outside diameter aeroshell with a 25 mm honeycomb thick shell. All components and instruments fit within this envelope with some volume to spare, however the resulting CM location is farther aft than desired. With a slightly larger aeroshell it may be possible to move the CM forward significantly.

A detailed analysis of forces this configuration can withstand has yet to be done, however volume exists for additional structure or rearrangement of components. Preliminary analysis suggests that even with its significant shocks, a rolling, bouncing deceleration can be handled with the honeycomb thicknesses shown in this configuration in such a way that the ruggedized equipment will survive and operate with reasonable probability. Because of the small size (and mass) of the components, it is likely that they will be able to withstand landing loads of several hundred g’s or more.

Power and Thermal

The current design requires an average of approximately 3W to maintain normal daytime operations with the appropriate component duty cycles. The 20 photovoltaic Ultra-high Junction (UTJ) solar cells mounted on 3 “platters” are expected to generate an average of approximately 10.8 W, assuming the probe is 1.54 AU from the Sun, and 70% collection efficiency due to sub-optimal sun angles and shadowing. There is a 45% margin on the amount of average collected power relative to what is required to sustain normal operations during the day.

Energy storage is provided by six 18650 Li-Ion batteries, which are selected for their high heritage and energy capacity. The batteries are placed as close to the heat shield cone's nose as possible in the structure to move the CM forward. During nighttime, the batteries provide the required power (2 W) to a heater to keep the electronics and batteries warmer than -40^{o}C (minimum acceptable battery operating temperature, which is the driving temperature requirement). A preliminary thermal analysis considering radiation, convection, conduction (to the Mars surface at -120^{o}C at night), and heating shows it will be feasible to keep the system near 0^{o}C using aerogel insulation and vapor deposited gold tape to minimize radiation losses. Assuming the worst-case night with maximum eclipse duration for lower latitudes (12.5 hrs of 1.02 day sol), there is a 188% margin on the battery energy storage. This initial result suggests that the system will be capable of surviving months or longer even at somewhat higher latitudes.

Telecommunication & Data Plan

Telecommunication is accomplished with an ultra high-frequency (UHF) Proxy-1 compatible micro-transceiver[34] and low-gain whip antenna mounted on one of the panels. The $\text{MARS}_{\text{DROP}}$ lander will perform two-way communication with a Mars orbiter (e.g., *Mars Reconnaissance Orbiter* and others). With a 1 W RF output power and a 0 dBi antenna, $\text{MARS}_{\text{DROP}}$ can achieve a 16 kbps uplink rate for a worst-case range (971 km at 20^{o} elevation) with a 4.1 dB margin.

Commanding the $\text{MARS}_{\text{DROP}}$ direct-from-Earth is not feasible nor required and no real-time link during entry, descent, and landing is planned (however this may be considered in future revisions). The lander will continuously collect, store, process, and transmit data to a Mars orbiter *autonomously*. The orbiter will also command $\text{MARS}_{\text{DROP}}$ from Earth to request desired data or change the operational plan.

There will be approximately three to four overflights per sol for a Mars relay orbiter (assuming a ~370-400 km Sun Synchronous orbit like MRO) that last longer than ten minutes (horizon to horizon). Assuming we have one 8.5 minute pass per sol (to account for a minimum elevation angle for transmission), the $\text{MARS}_{\text{DROP}}$ probe is expected to return approximately 1 megabyte (MB) every sol.

After landing, there will be considerable data onboard from the descent vide, which dominates all data. The full resolution descent video will be just under 2 GB and will be stored in the 8 GB data storage capability (providing a 300% data storage margin). After landing, $\text{MARS}_{\text{DROP}}$ will upload 6 MB temporal and spatial low-resolution video and geological images in the first ~6 sols. This will include 4.4 MB of VGA time lapse video and 9.4 MB of VGA thumbnails from the 8 geological cameras providing a 360^{o} panorama of the scene, and engineering data. (Note that a different, single-lens camera is shown in Figs. 8 & 9. The 8-lens camera has about the same mass.) Thereafter higher-resolution video and desired regions of geological images will be requested and returned over time, driven by telecomm link availability.

$\text{MARS}_{\text{DROP}}$ will continue to collect weather and TLS (methane) data, with data rates and volumes shown in Table 2. Weather data will be collected at a relatively low cadence but the rate is highly flexible depending on observations and scientific interest. The TLS will collect methane measurements continuously at a rate of approximately once an hour, however onboard processing will determine when the data taken represents significant variation at a level making uploading the data worthwhile. Normally, approximately one 4 kbit spectrum will be collected and returned from the TLS every week for calibration purposes. The data management and upload strategy are highly flexible to respond to data content and ground commands. For example, if methane is detected by a change in the spectrum, the TLS sampling rate will increase, and methane data will displace video playback data within the transmit allocation.

Table 2: Data volume to be uploaded in sols 7-80 at a rate of ~1 MB/pass, and 1 pass/sol.

Data Source	Type	Data Volume (MB)
Descent Video	Full resolution VGA Video (1/4th of video)	65.92
Geology Image	Full resolution (1 camera)	3.00
Weather Data	Temperature, Humidity, Pressure (300 bits/min, 80 sols)	2.16
TLS	Methane Spectrum Data (4 kbits/7 sols)	0.006
Total	Including 1% Housekeeping Data	71.1

The collected science data will be prioritized for upload, and the full resolution video and geology images will also be slowly uploaded. A representative data volume budget is provided in Table 2, which is expected to be uploaded within the first 80 sols after landing. The budget is performed for 80 sols as our target lifetime is approximately 3 months. Although there is nothing technically limiting the probes' lifetime (in particular battery and electronics cycling are not expected to be limited to this timeframe), this is a typical lifetime goal for this mission class.

Computing

Computing loads are highest during real time processing of the glide images taken by the forward-looking descent camera to support navigation during descent.

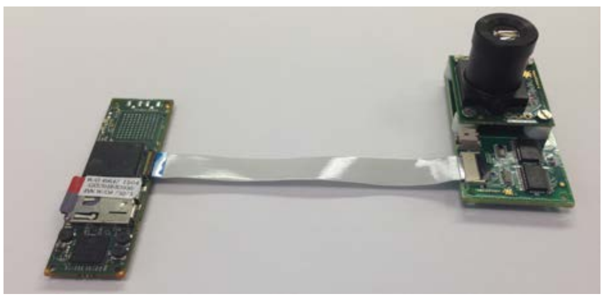

Figure 10: Gumstixtm module (left) mounted on a programming board and connected via flex cable to a 5 MPixel OV5640-based camera with M12 lens.

The selected Gumstix Overo Earth Storm COM with TI AM3703 DSP and camera (as shown in Figure 10) can perform in excess of the required processing capability while consuming ~500 mW. A unit has been tested to 3.2 krad in JPL's Co-60 beam, a relevant dose for this type of mission. A MEMS IMU (not shown) will allow synchronized image and attitude knowledge for terrain relative navigation and location determination. Real-time feature detection will be computed on individual frames and compared to preprocessed maps while built-in hardware encoders compress and store video of the descent video to a microSD card connected directly to the processor. The microSD card allows for multiple processed maps corresponding to different altitudes to be stored while allowing ample room for 10 to 20 minutes of high quality video. The preprocessed maps are weighted to highest priority landing sites, and MarsDrop continuously reevaluates its location and navigation ability to ensure it lands in a location of highest scientific priority regardless of initial trajectory or local winds.

PROOF-OF-CONCEPT TESTING

Aerospace Corporation people undertook a series of proof-of-concept testing from high-altitude balloons, see hardware in Figure 11 and Figure 12. A prior conference paper provides a more complete report, but a summary of this drop testing follows.[2] At ~100,000 feet (~30 km) we find an ideal laboratory replicating the Martian atmosphere, a cold and thin atmosphere with a density a scant 1% of that at sea level on Earth. The minimal size and weight of the probe, combined with subsonic test speeds, permitted the use of standard weather balloons, minimizing test costs. A mock capsule, fitted with the proposed parawing deployment architecture, was attached to a weather balloon, which towed the vehicle 20 miles (~32 km) up. Cutting free from the balloon, the capsule free fell briefly until matching the speed (~400 mph; ~640 km/hr) and dynamic pressure (~200 Pa) it would see during parawing deployment on Mars. A switch of a power relay fired off the backshell and with it the packed parawing. During deployment, inflation loads were measured with a 100-g accelerometer, while GPS readings tracked the probe's forward and vertical velocity during its descent.

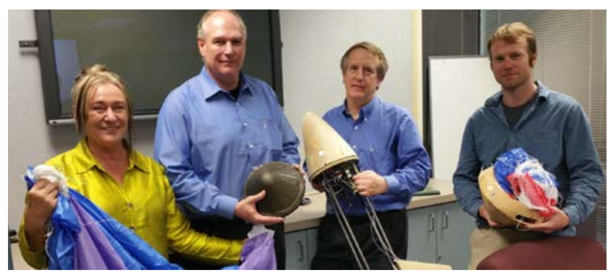

Figure 11: (left to right) Full-size MarsDROP parawing, entry vehicle, balloon drop apparatus, and balloon drop configuration with quick-folded parawing, held by Aerospace Corp. personnel.

In summary, we have demonstrated to date:

- Verified the parawing construction (the ability to withstand inflation, at the expected deployment dynamic pressure, without damage).
- Verified a mortar-less deployment scheme of the parawing using the backshell as a pilot feature.
- Confirmed that sufficient volume is available for a useful payload by integrating the landing system within a mock capsule.
- Verified parawing deployment from a mock capsule.

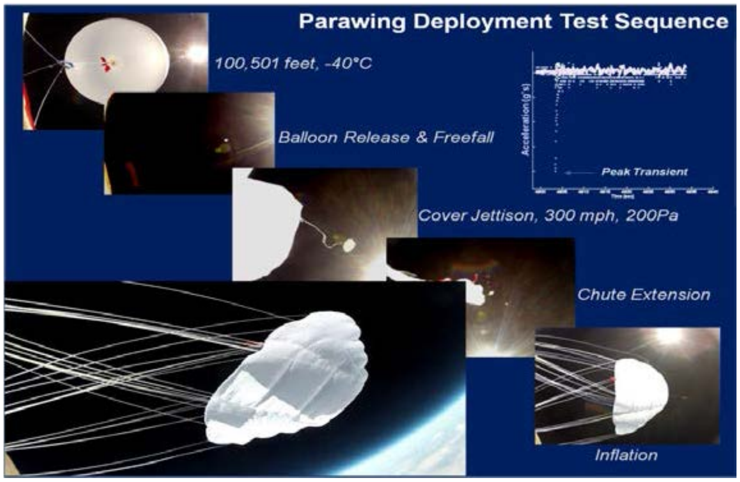

Figure 12: Sequence of a high altitude test run.

Future tests will likely be performed in order to:

- Explore techniques, such as sail sliders, to slow down the parawing inflation and reduce high-altitude twisting/fluttering.
- Measure the integrated capsule glideslope at altitude.
- Explore parawing steering and guidance toward a precision landing capability.
- Drop test a capsule integrated with a proposed scientific payload.

INSTRUMENTATION

What can we land on Mars that is at once small and light, requiring little power and limited in the data sent back, but at the same time bold, scientifically valuable, and engaging to the general public? Technology surveys identified enabling hardware elements, many of which are maturing through recent CubeSat research and development at JPL, Aerospace, and elsewhere. Small, lightweight radios, scientific instruments from seismometers[35] to a lab-on-a-chip, cameras, and higher-energy-density batteries all advance the utility of the landed mass far beyond DS2 capability of the 1990s.[36,37]

For our reference 30 cm diameter $\text{MARS}_{\text{DROP}}$ lander, scientific instrumentation can mass up to a 1 kg payload. For our proof of concept design, we chose the following, though a great many other selections and combinations are possible: 1) Descent video camera for navigation and science; 2) Multi-head geology panoramic camera; 3) A TLS-based Open-Path Laser Spectrometer to measure methane and differentiate its isotopic forms; plus three sensors from Mars Science Laboratory for: 4) Temperature; 5) Pressure; and 6) Relative Humidity (RH).

Desired payload lifetime is determined by the nature of the scientific investigation, while feasible duration is determined by configuration and sizing of the electrical power system and thermal control components. For missions carried out through descent and landing up to a few hours or perhaps a few days, primary batteries are the best choice. For our proof-of-concept design, we chose a three-month design life target, with no design parameter necessitating end-of-life at that point. With upgraded components, and a method to periodically dislodge dust gathered on the solar panels, multi-year low-power, low data rate network science investigations could be possible. If components cannot be made to survive the deepest nighttime temperatures within battery and heater energy constraints, then radioisotope heater units (RHUs) offer an alternative[38] with somewhat complex regulatory requirements. With a mass of ~40 g and continuous 1 W thermal output, RHUs were employed aboard the low-cost Mars Pathfinder *Sojourner* rover, and aboard Mars Exploration Rovers *Spirit* and *Opportunity*.

We held a science mini-workshop to decide on an example payload for an existence-proof-level design, so that we would have instrument accommodation requirements to work with and attempt to reach design closure.[39] The first decision from this Workshop was that the idea of a purely "technology demonstration" first $\text{MARS}_{\text{DROP}}$ mission with a token science payload should be, and could be, upgraded to a "first science demonstration" affording a significant scientific result. The investigators associated with a UV fluorescence instrument to be a next-generation of the Scanning Habitable Environments with Raman and Luminescence for Organics and Chemicals (SHERLOC) instrument being built for Mars2020 reported that 1 kg represented too-aggressive a mass target compared with their technical status. They noted that their desired $\text{MARS}_{\text{DROP}}$ -compatible mass could likely be achieved by the time a second-generation $\text{MARS}_{\text{DROP}}$ would be ready to fly.

Investigators associated with the Tunable Laser Spectrometer (TLS) aboard the Curiosity Mars rover reported that they had a miniaturized version of their instrument in the laboratory, and that a version had flown on a quadcopter in a development funded by a natural gas provider. This Open-Path Laser Spectrometer (OPLS) measures methane to 1 part-perbillion (ppbv) and water to 1 part-per-million (ppmv) in 1 second, weighs 150 g including electronics and optical head, and has time response of 10 Hz for high resolution vertical profiling. OPLS uses miniaturized tunable lasers and mid-infrared laser technology designed, built, and deployed at JPL for NASA Earth and Planetary science missions for over three decades. Custom electronics include real-time signal-processing which reduces the telemetry load to 100 bytes-per-second when data is being taken. The instrument is currently being field-tested to find and quantify terrestrial natural gas leaks in collaboration with the energy industry.

Obtaining methane measurements from more than one simultaneous location could help constrain candidates for origin of the methane. For example, it is not yet possible to tell from Mars2020 data if the methane peaks measured there are a regional or global phenomenon, or simply a local feature, for example the result of a small carbonaceous meteoroid. Methane peaks corresponding with seasons might have a different set of explanations than emissions not synchronized with Mars' revolution around the Sun.

Prior to the science mini-workshop, we performed an informal survey amongst close colleagues of the types of instrumentation and enabled science investigations that could be performed within the very tight constraints of the $\mathrm{MARS_{DROP}}$ instrument payload capability. While our survey was by no means comprehensive, the breadth of viable instrumentation considered feasible in the 2020+ timeframe (with some potentially ready earlier) was promising, ranging from high resolution science-grade and mineralogy cameras, to simple weather instrumentation, to seismometers.

Surface-based imaging provides a critical perspective and higher resolution than is achievable from orbital instruments. The viewing geometry is complementary to the satellite view, helps verify and understand the surface context for orbiter imagery, and can be instrumental in resolving stratigraphic relationships of rock units. Higher resolution images are also critical for determining sediment grain size, an important parameter for accurately constraining flow magnitudes, as was done for the fluvial conglomerates observed in the first images at Gale crater prior to rover driving.[40] Spectral filters on, or illumination from a camera and/or spectrometers can help determine rock compositions for targets that are not resolvable in orbital data.

To meet this imaging objective, we initially chose a copy of the camera used for terrain relative navigation (and the video) during descent. We expect the descent camera to be torn off the lander during the roll-and-bounce landing, so will shut off its electrical connections. But after the lander comes to a stop and the petals are deployed, a science camera is deployed in a ~15 cm stalk. Further analysis suggests that while the descent camera would provide quality images, a better geological awareness could be constructed using a circular array of cameras creating a full panorama of the landing site, all for about the same mass and volume as the larger descent camera.

For seismic studies, MicroElectroMechanical Systems (MEMS) accelerometers for seismology are small (grams) instruments that can measure displacement resulting from seismic waves. The data can be compressed and transmitted efficiently through the use of a triggering algorithm to only transmit data from detected events. If paired with a simple anemometer, measurements could be blocked at times when they might be confused with wind gusts. This instrument would be ideal for recording the crustal response of new impact craters. This could build upon, and augment, the Seismic Experiment for Interior Structure (SEIS) onboard InSIGHT, which plans to have surface operations ~2017-2018. This capability could further refine our understanding of the planet's interior structure. With two seismic detectors simultaneously operational on Mars, new impact crater locations can be identified and imaged using orbital assets.[41] However, because of the difficulty separating seismic from wind and thermally-induced motion, we chose not to include a seismometer on the science demonstration mission concept.

At any suitable location, lightweight suites of environmental sensors can be flown to better characterize the near-surface atmosphere and even surface interactions. For example, at the south polar cap, the payload could measure CO2 sublimation rates to confirm polar cap changes detected from orbit.[42]

A further expansion of $\mathrm{MARS_{DROP}}$ capability is represented by the possibility of carrying a micro-rover, perhaps along the lines of those being developed at a number of universities and in JAXA's planetary exploration program.[43]

BEYOND MARS

Mars, of course, is not unique in our Solar System for having an atmosphere. Venus, Titan, Earth, Triton, and Pluto (in descending order of surface atmospheric density) all have known atmospheres sufficient to create significant aerodynamic braking down to a definitive solid surface (Figure 13). As with Mars, any knowledge we gain of their atmospheres will help not only understand EDL constants, but also current environmental constraints on habitability.

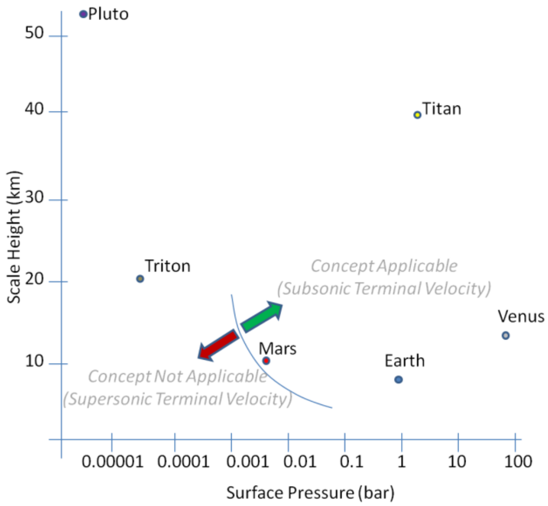

Figure 13: The MARSDROP concept requires enough atmospheric weight (surface pressure/density) and thickness (scale height) to provide for a subsonic parachute deployment.

Titan is an especially interesting case because missions there are likely to be very infrequent, so having a microlander(s) can double or triple the number of landed payloads aboard a particular mission for a small cost increment. Unique challenges are presented, including Titan entry velocity, low temperature, low light level, and longer telecommunications distances to a mother spacecraft if it is in orbit around Saturn, as was *Cassini*. If a mother orbiter is around Titan, then telecommunications distance is dramatically reduced from the *Cassini* case. But unless aerocapture is employed, Titan orbiters need to be much smaller than Saturn orbiters for a given size launch vehicle and desired instrument payload, and trip time. The large variety of terrain types, lakes, and potential rivers viewed by the *Cassini* radar increases the science value of landing in multiple locations. Specific target locations could be uploaded to mcirolanders before separation from a parent spacecraft. However, if the target ellipse size proves too large, as seems likely given Mars experience, a mathematical description of the desirable terrain type and configuration could be loaded into a site-seeking algorithm, to which the parawing-borne lander would steer to during glide. A hierarchy of target priorities could even be programmed such that if the highest priority were not visible by a particular time during descent, then targeting could default to the next-highest priority. Target types could be mathematically described, such as mid-lake, lake shore or cove, flow channel intersection with size thresholds, dendritic terrain, trough in the middle of a dune field, etc., using IR imagery through an atmospheric window of low opacity to guide the glide path.

Probe entry velocities can be similar for Titan and Mars. The *Huygens* entry probe carried with the *Cassini* orbiter had an entry velocity of 6 km/sec,[44] slightly above the design value for the Mars Science Laboratory entry vehicle.[45] Direct entry from an Earth-to-Titan trajectory would be much higher without an intervening maneuver to enter Saturn orbit. For comparison, 11 km/sec was the entry velocity for the *Genesis* sample return capsule at Earth.[46]

Jupiter, Saturn, Uranus and Neptune afford a different kind of opportunity for small piggyback entry vehicles, where use of a parawing, as in the MARSDROP concept, would afford both extended descent periods compared to that of a ballistic, parachute-borne entry probe as used with the *Galileo* probe, and the ability to survey extended horizontal distances during descent. The combination of these two advantages suggests the possibility of employing an autonomous interpreter of video imagery coming from an onboard camera to steer toward targets of specific interest, or toward "unusual" cloud features, where "unusual" is defined mathematically in onboard algorithms that search for portions of images showing marked differences or contrasts from the rest of an image or from prior images in a video sequence. Science-driven search algorithms were developed and employed aboard the Mars Exploration Rover,[47] and it is clear that extending this technique to the world of small probes would be valuable.

CONCLUSIONS

We have shown a non-optimized existence-proof-level design for Mars microlanders, which utilize high-heritage small spacecraft and Mars mission hardware and software.

This suggests the feasibility of an approach to augment Mars exploration by enabling precisely-targeted landers with focused science objectives that could be delivered to Mars' surface at dramatically lower cost than any prior Mars landing mission. One or perhaps a few such landers could be carried to Mars on any larger Mars-bound mission as piggyback payloads, with targeting enabled by additional cruise-stage maneuvers to place the landers on their own impact trajectories. Alternatively, a cruise stage dedicated to a small fleet of such landers could be launched to Mars, or a smaller cruise stage could carry one or two MARSDROP-style landers, e.g., from a suitable ESPA-class propulsion module departing out of geostationary transfer orbit (GTO). Such landers could glide for ~10 minutes after parawing deployment with a 3:1 glide ratio and impact the surface at ~ 20 m/s.

Further, a $MARS_{DROP}$ lander appears capable of being equipped with terrain-relative terminal guidance software that could pick out one of several pre-programmed landing locations from a forward-looking camera image, steer to that target, and land using a flare maneuver within ~10 m at a speed <10 m/s. At < ~1 kg, the available payload mass for science instrumentation and telecommunications to an orbiting asset is small, but capable of making focused, potentially breakthrough-class measurements, at specific sites of interest, such as craters that show evidence of recent near-surface liquid water flow.

Measurements and data, ranging from descent videos, to geophysics and mineralogy, to biosignature detection, are plausible from multiple locations at relatively modest added cost using the $MARS_{DROP}$ architecture to augment the number of landed payloads from any Mars-bound primary mission.

Though small in number, the few atmospheric stations operated on the Martian surface have added tremendously to our understanding of the near-surface environment.[48,49,50,51] Even MER, the only exception to many Mars landers and rovers in not having a dedicated meteorology station, used its instruments to characterize the local environment.[52] The different landing sites of *Viking*, *Mars Pathfinder*, *Phoenix*, and MSL showed that location strongly influences the environment. A network of landed meteorological stations could add to the existing knowledge and is directly responsive to the MEPAG recommendations and to NASA's Mars Exploration Program.[53]

As a stepping stone to implement these capabilities, we propose a first science demonstration to be carried as a daughter spacecraft on a Mars-bound mission in 2019 or later. A piggyback lander using the Aerospace Corporation's REBR backshell/heat shield forms the heart of a 10 kg package carried on a Cruise Stage (~5 kg of interface equipment remains with the Cruise Stage). The capsule is spring-ejected on command from the Cruise Stage at 1 m/s above and behind the Cruise Stage immediately after separation of the primary lander, in order to prevent re-contact. While both the primary and daughter lander enter Mars' atmosphere at approximately the same time (possibly separating entries by a few minutes to avoid data relay conflict during entry and landing), the lower ballistic coefficient of the daughter lander ensures a retarded trajectory behind the primary lander. Slowing to subsonic speed by virtue of its ballistic coefficient, the science demonstration daughter lander deploys its parawing, orients its glidepath and uses small lanyard actuators to steer the parawing left or right. A camera aims forward, centered on the glide slope. After landing, a UHF radio and antenna broadcasts to an orbiting asset when in view, the landed science data, and piece by piece a compressed version of the full descent video file and interesting scientific discoveries on the surface.

We have yet to demonstrate the ability to sterilize a $MARS_{DROP}$ probe, but note that its small size can enable simplified handling after sterilization. Once sterile, the microlander can be placed is a shaped plastic bag for integration onto the host mission. Upon entry into Mars' atmosphere, such a bag will simply burn off, and the sterile probe may be targeted to biosensitive areas with highly specific instrumentation.

Once demonstrated, each future primary Mars mission can carry several piggyback $MARS_{DROP}$ landers, thereby doubling or tripling the number of landings for a small additional cost, while enabling true network science. This will allow heavy university and small business involvement, at a level just now starting with beyond-Earth CubeSats.[54,55,56]

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