ROUGH LANDER CONCEPT FOR MARS EXPLORATION

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Martian surface exploration to date has been limited in large part due to the cost, risk, and complexity of the delivery systems employed. A new concept is proposed herein for small spacecraft that use a "rough" landing approach to enable a substantial reduction in delivery system complexity relative to previous "soft" landing systems. First use of this type of vehicle is targeted toward the 2007 launch opportunity in NASA's Mars Scout Program. Once on the surface the Rough Lander relies on high energy density primary batteries, coupled with a low power avionics configuration, to perform up to a 30-day science mission. The ruggedness and simplicity of this type of vehicle may also enable access to areas of the Martian surface that are otherwise deemed too risky until more sophisticated soft landers are developed.

INTRODUCTION

Exploration of the Martian surface has proceeded at a slow pace during the past three decades, due in large part to the sheer technical difficulty and cost of reaching the surface safely. Several different engineering approaches to Entry, Descent, and Landing (EDL) have been employed in previous missions, using combinations of aerodynamic and propulsive deceleration to reach some desired terminal velocity near the surface, coupled with mechanical elements for shock attenuation and arresting of the vehicle's motion upon contact. With one exception, the EDL systems used in NASA's Mars missions such as *Viking*, *Mars Pathfinder*, and *Mars Polar Lander*, have been designed to limit the impact load factors experienced by landed elements to the 20-50 g range (relative to the terrestrial value of "g," or 9.8 m/s^2). The sole variant from this has been the experimental development of two small (3 kg) penetrators, designed to tolerate up to 60,000 g's at impact. Although low in cost relative to larger, "soft" landers, the shock environment associated with penetrators places severe limits on payload size and capability.

An alternative Mars lander concept is proposed herein, a "rough" lander, designed to provide an intermediate landed payload capability targeted toward first use in NASA's Mars Scout Program of low-cost, focused science missions beginning in 2007. The Rough Lander concept provides an impact shock environment in the 500-1,600 g range, low enough to accommodate a variety of modern scientific instruments, ruggedized electro-mechanical components, electronics, and power sources, but large enough to enable a substantial reduction in EDL system complexity and cost relative to soft landers. In addition, the ruggedness and simplicity of this concept may allow access to areas of the Martian surface that are otherwise too risky for all but the most sophisticated future soft landers. Similarities and differences between the proposed concept and earlier, similar concepts for Lunar and Mars exploration dating back to the 1960's[1-5] will be described as well.

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SYSTEM DESIGN TRADE SPACE

In preparation for the concept definition effort, a generic set of goals and objectives was established for the design of a new lander spacecraft that could potentially serve as a flight system platform in several different landed missions. The class of missions envisioned involved the delivery of multiple small, low-cost instrumented landers to the Martian surface, possibly at widely separated locations, capable of operating semi-autonomously for relatively short periods, and relying on Mars-orbiting assets as relays for communication with Earth. This set of objectives, crafted to be consistent with the anticipated cost and schedule constraints of Scout-class missions, is listed in Table 1.

Table 1: Small Scout-Class Lander Mission Performance Targets

Parameter	*Value*	*Remarks*
No. of Landers	2-4	To support multiple site investigations
Lander Mass (each)	< 120-180 kg	Based on expendable launch vehicle capability
Landing Site Elevation	< 2,000 m	Relative to current reference ellipsoid
Landing Site Latitude Range	±60 deg	Seasonally dependent
Science Payload Mass	10-15 kg	Targeted at smaller science payloads
Science Payload Volume	*12-15,000 cm^3*	*Consistent with anticipated small payloads*
Payload Energy Consumption	100-150 W-hr/Sol	Dependent upon landing site location
Science Data Transport	50-100 Mb/Sol	Dependent upon landing site location
Surface Mission Lifetime	20-30 Sols	Min. needed for anticipated investigations

The trade space taxonomy developed while defining the potential options for meeting these goals is presented in Figure 1. The options considered were organized in terms of the approach for carrying the landers to Mars, the entry, descent, and landing phase, and the avionics/power subsystem architecture for the surface mission. Although the cruise/approach phase does not involve the lander directly, the impact of the options considered (separate carrier spacecraft vs. cruise stage supporting each lander as a "free-flyer" vehicle) on interfaces with the lander merits consideration, as such this element of a mission was included as part of the Lander system design trade space.

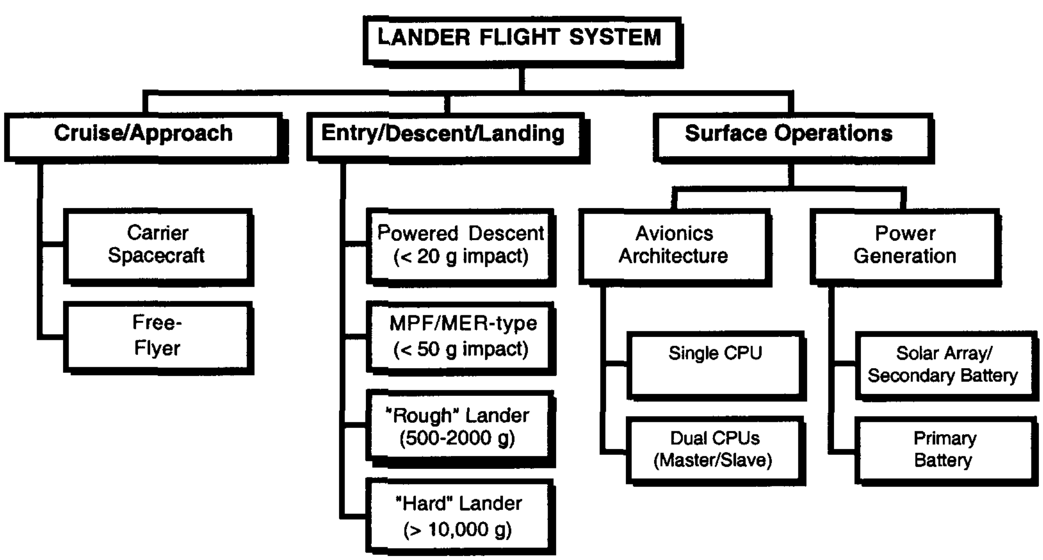


Figure 1: Small Scout-Class Lander Trade Space

Early efforts focused on EDL design concepts that were architecturally based on the successful Mars Pathfinder (MPF) lander, which employed a combination of a parachute system, retrorockets, and an airbag system for landing shock attenuation to accommodate impact velocities in the 10-30 m/s range. The upcoming Mars Exploration Rover (MER) EDL system design is similar, but with additional sensors and vernier rockets added to reduce the impact velocity envelope to 10-20 m/s. Scaled versions of the MPF and MER airbag designs were considered, in conjunction with the addition of a large, second parachute intended to bring the lander to a terminal velocity low enough to eliminate the need for retrorockets. This MPF-heritage concept was ultimately abandoned, though, due to the requirement for a large and costly test program for the design and qualification of a new airbag system, and difficulty in packaging both a large parachute and airbags into a small lander volume.

The potential cost and complexity of even relatively small soft landers motivated a search for alternatives that may be significantly simpler and less costly, yet still capable of approaching the performance targets of Table 1. This search led to a study of EDL systems for rough and "hard" landings (see Fig. 1), which were found to be substantially simpler than the soft landing systems considered previously, but required the landed payload to withstand impact loads of 100's to possibly 1,000's of g's. Subsequently, the design effort began to focus on concepts for rough landing in the 500-2,000 g range due to the potential for a relatively simple EDL system, while still providing adequate landed payload mass capability.

For perspective, Figure 2 shows a comparison of the landed mass and load factor experienced at touchdown (or initial impact) for all of the robotic spacecraft developed by NASA for Lunar and Mars exploration, along with several spacecraft for which prototypes were built and tested, but were never used in an actual mission. The Lunar spacecraft shown in Fig. 2 include the *Surveyor* series of soft landers, in which five successful landings were achieved between 1966 and 1968, and the impact capsule carried by three of the nine *Ranger* spacecraft in 1962-63. Although engineering models of the *Ranger* landing capsule were tested successfully, none of the *Ranger* flights equipped with these capsules were ultimately successful.

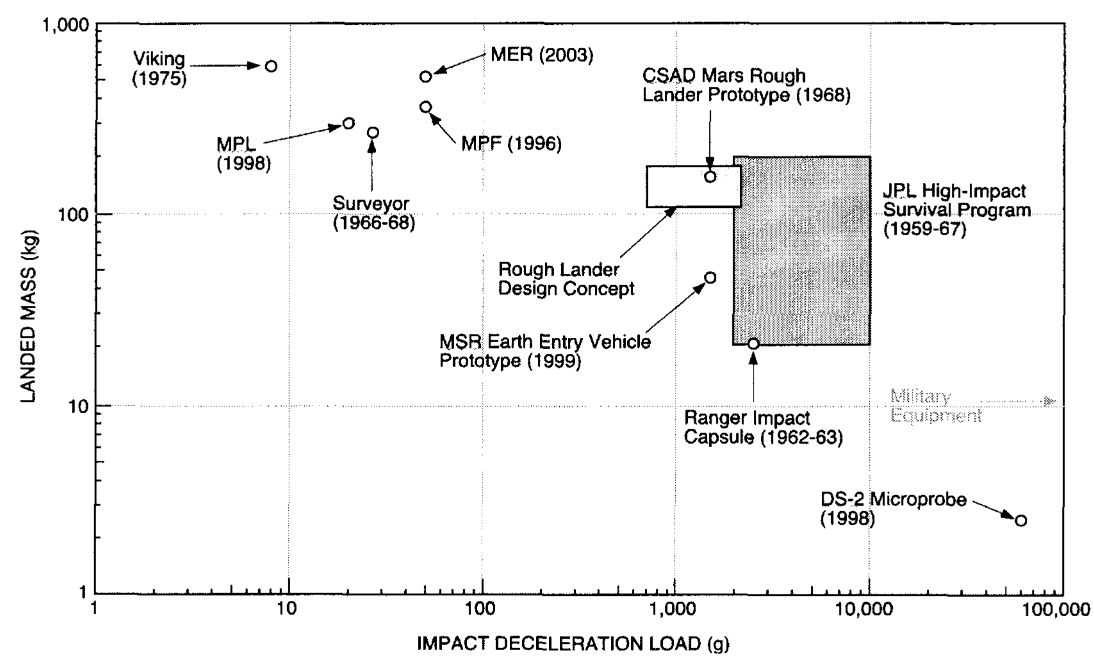
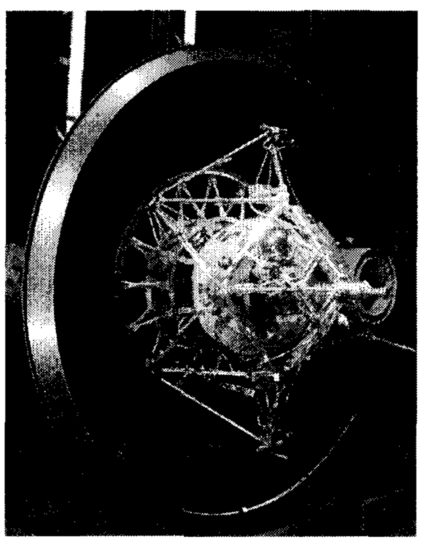


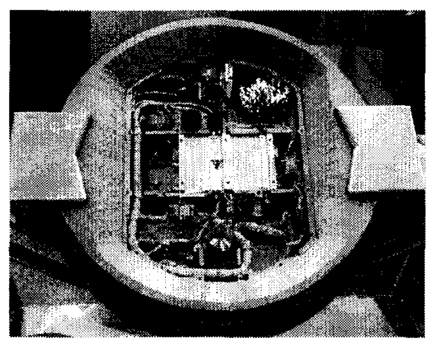
Figure 2: Lander Mass/Impact Load Factor Historical Comparison

In addition to the envelope considered for a Rough Lander design, the Mars Landers shown in Fig. 2 include soft landers employing powered descent, such as *Viking* and *Mars Polar Lander* (MPL), and the retrorocket/airbag landers (MPF and MER) discussed above. The twin Deep Space-2 (DS-2) Microprobes, flown unsuccessfully in 1998/99, represent the sole example of a hard landing approach, as these vehicles were designed to withstand up to 60,000 g's at impact. Also shown is the point represented by a prototype Earth Entry Vehicle, built and tested in 1999 as part of a possible Mars Sample Return (MSR) mission. When considered in the overall context of previous landing vehicle designs, the Rough Lander design envelope is found to be consistent with the mass vs. impact load trend shown in Fig. 2.

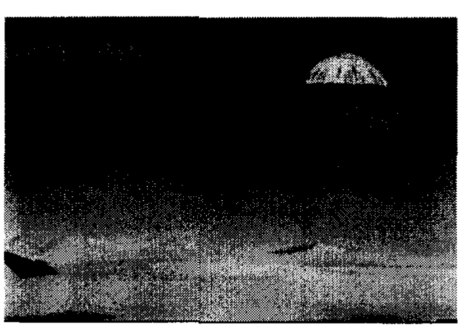
The most surprising example encountered in the development of Fig. 2 is a prototype Mars rough landing vehicle created under the auspices of the Capsule System Advanced Development (CSAD) Project[4] at JPL during 1967-68. CSAD designed, built, and successfully drop-tested a fully functional prototype lander at shock levels up to 2,500 g in just 15 months. Several views of the CSAD lander are provided in Fig. 3 below. Figures 3a and 3b show close-ups of the impact capsule and its balsa wood shock limiter during assembly and after the vehicle's second test, which was conducted on an asphalt road surface near Goldstone Dry Lake Bed in California.



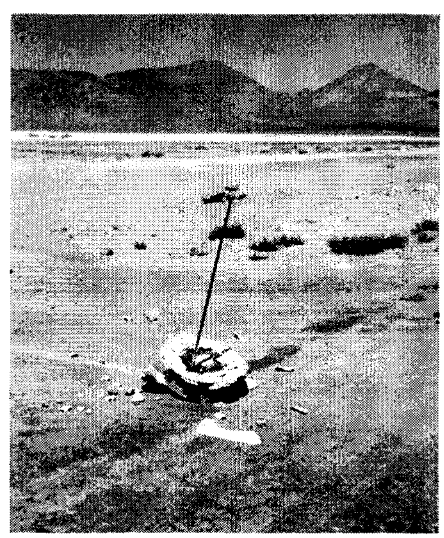
a) Integrated lander system



b) Impact capsule assembly



c) Mars landing (artist's concept)



d) 2,500 g drop test (28 May 1968)

Figure 3: CSAD Mars Rough Lander Prototype

ROUGH LANDER FLIGHT SYSTEM

In this section a Rough Lander design developed for a proposed 2007 Mars Scout mission, entitled The *Naiades*,[6] is described. An expanded view of the flight system is shown in Fig. 4. The Lander and its two-stage parachute system are encapsulated inside an entry body whose geometry is inherited from the Viking and Mars Pathfinder vehicles. The first parachute, deployed supersonically, uses a scaled variant of the Mars Pathfinder design. The second parachute, deployed at subsonic speeds, employs similar heritage from the Viking canopy geometry, taking advantage of its greater drag efficiency. The touchdown system is an impact attenuation substructure made of balsa wood, or any one of several alternative shock absorption materials. It is integrated into a Kevlar shell that provides significant rock-strike protection, as well as providing the desired aerodynamic configuration. In this particular example, power is provided by a primary battery system, sized to provide sufficient lifetime for a short duration science mission with ample margin. Key system design parameters are summarized in Table 2.

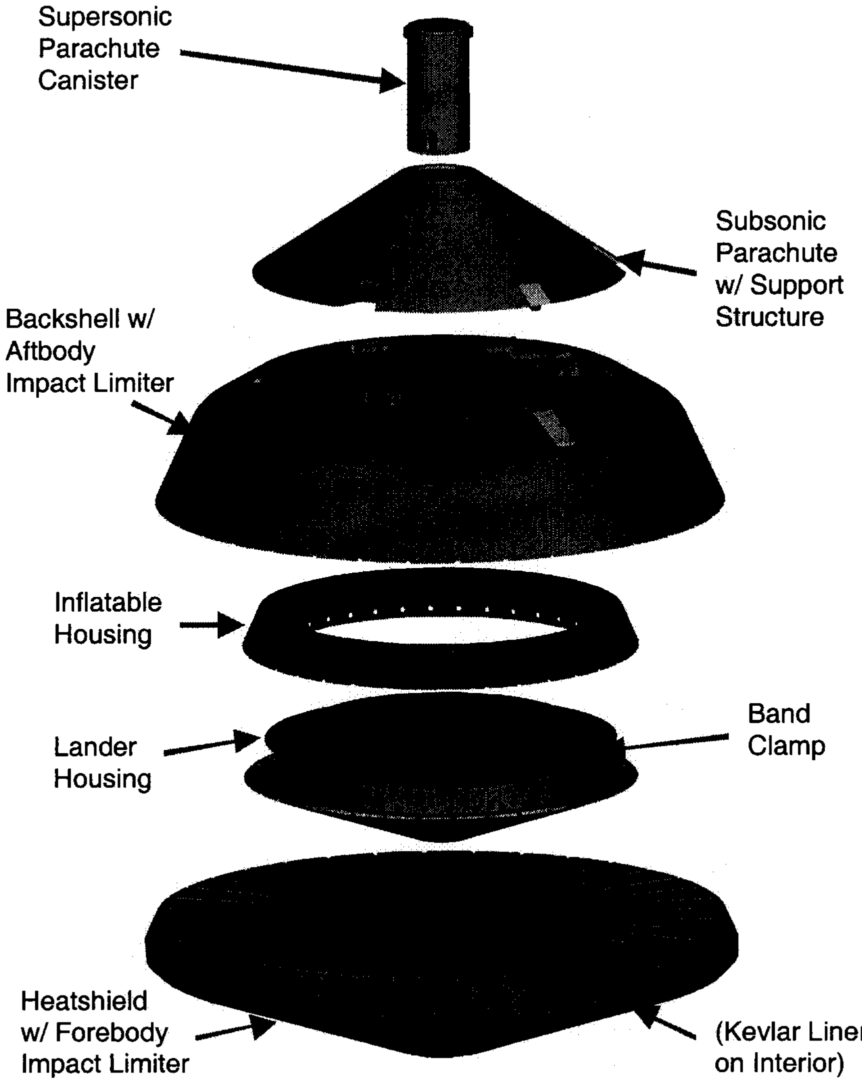


Figure 4: Flight System Expanded View

Table 2: Key System Design Parameters

Parameter	*Value*	*Remarks*
Lander Mass	150 kg	Includes subsystem contingencies
Diameter	1.25 m	Scaled Viking/MPF forebody
Impact Shock Tolerance	2,000 g	4-π steradian protection
Primary Battery Capacity	6,000 W-hr	Discharge capability at 0° C
Onboard Data Storage	2.0 GB	Non-volatile memory capacity
Data Transport Capability	72 Mb/Sol	For two 5 min. sessions to MRO*
Landing Site Elevation Capability	2,000-2,500 m	Dependent on arrival date
Landing Site Latitude Range	40°S-70° N	Dependent on arrival date
Surface Mission Lifetime	15-30 Sols	Varies w/ latitude and arrival date

*Mars Reconnaissance Orbiter, planned for operation at Mars during the 2006-2011 time period

Entry/Descent/Landing Sequence

The key events occurring during EDL are illustrated in Fig. 5. Prior to atmospheric entry, the spacecraft must be delivered to within the desired entry corridor (acceptable flight path angle range). During atmospheric entry the lander decelerates aerodynamically using its Viking/MPF heritage 140 deg sphere-cone forebody configuration. Thermal protection is provided by SLA-561V ablative material, or some other alternative with similar capability.

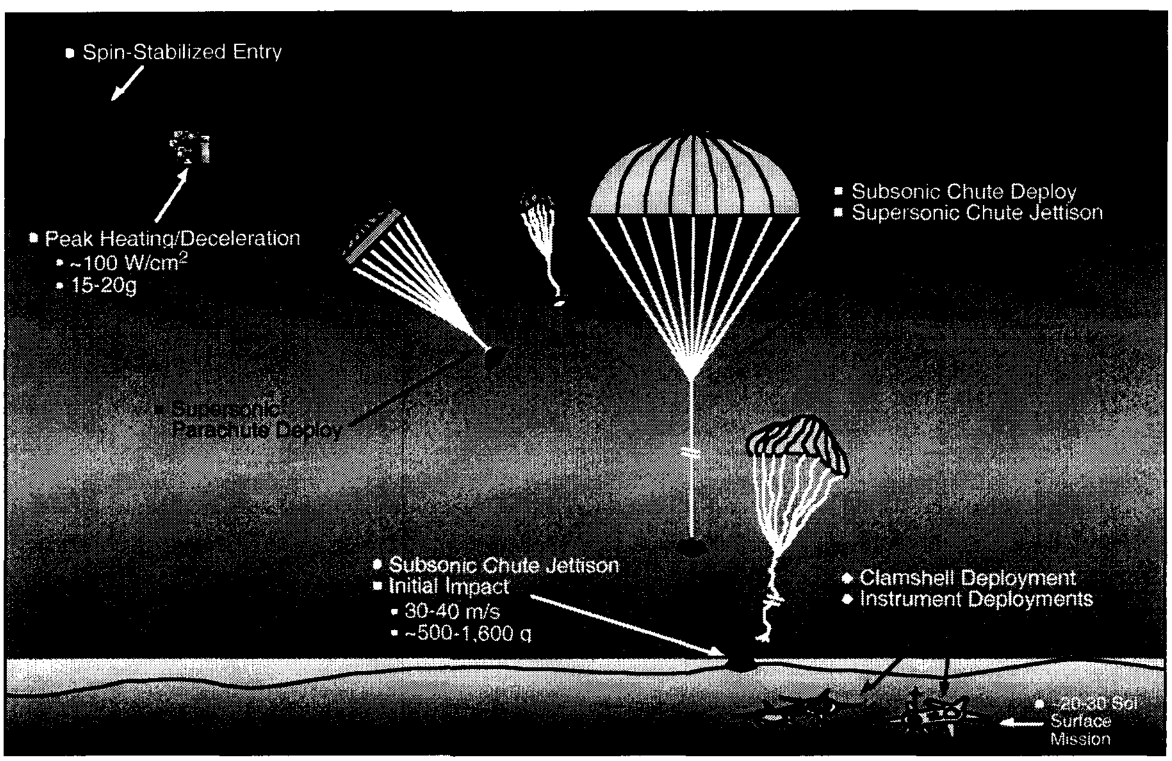


Figure 5: Entry, Descent, and Landing Sequence

Supersonic parachute deployment is triggered in the Mach 1.5-2.2 range, based on the sensing of two predetermined deceleration values by g-switch devices, using the same type of algorithm employed successfully by the Galileo Jupiter Probe and MPF. As the vehicle approaches Mach 0.8, the parachute support structure is pyrotechnically separated, so that the supersonic parachute's drag extracts and deploys the subsonic parachute and descent bridle. The subsonic parachute is sized to bring the Lander to a terminal velocity of 30 m/s in a mean atmosphere at 2,500 m elevation. Depending upon the landing site location, season, and local solar time of landing, wind-induced horizontal velocities of up to 30 m/s may also be encountered during parachute descent. An important factor behind the choice of a 30-40 m/s impact velocity range was the ability to perform drop testing in an ambient terrestrial environment. This velocity range is achievable in the Earth's atmosphere for a test article with the same mass, size, and configuration as the actual flight system when dropped from heights of 60-70 m.

Upon contact with the surface, release of the subsonic parachute is triggered via another g-switch. The Lander is suspended below the subsonic parachute canopy by a 100 m bridle, to provide adequate time for separation between the two after touchdown. The aeroshell and impact attenuation substructure are intentionally designed to accommodate impact conditions beyond the maximum expected terminal velocity envelope, to provide margin. Payload impact load factors will be limited to <1,600 g while tolerating impact velocities with components up to 40 m/s vertical and 30 m/s horizontal, respectively, including cases involving impact on rocks, depending upon their exact shape and hardness. Under nominal conditions (landing site elevation and wind velocities), the anticipated landing loads decrease by a factor of two to four. To provide additional margin for landing survival, the Lander's avionics and instrumentation are designed to withstand shock levels up to 2,000 g, regardless of touchdown orientation. This conservative approach to impact attenuation guards against any subsequent rebound or rolling motion the Lander may experience before coming to rest.

The self-righting and instrument deployment sequence of events is shown in Fig. 6. After coming to rest, the Lander may be in any orientation. At a preset time after touchdown, a mechanical band clamp holding the backshell, forebody, and Lander together is released. A Self-Righting Inflatable (SRI) system is then deployed via a pressurized nitrogen bottle. In the initial phase of inflation two interconnected annular bags, each with four radial outriggers, are filled. Outrigger inflation ensures bi-directional stability of the Lander, while the annuli force the two halves of the Lander apart. The large surface areas and a high inflation pressure (4-6 psig) provide large force margins ensuring a clean separation of the forebody and backshell. Continued pressurization of the inflatable assembly causes both halves of the Lander to hinge apart and open, completing the self-righting process. The hinging and articulation is purely pneumatic and does not involve any metallic or motor-driven hardware. Velcro strips maintain the positioning of the fabric within the Lander to control the opening geometry.

Immediately after the Lander has completed self-righting, a second valve opens to initiate the inflation of a set of "forearms" for deployment of an instrument loop antenna onto the surface. This entire process occurs in less than one minute. After this deployment is complete, the inflatable system gradually deflates via a built-in leak rate; small springs are incorporated into the "forearms" to ensure that no obstruction remains in the way of subsequent sensor deployments.

Surface Operations

In considering this element of the trade space shown in Fig. 1, the choice of an avionics and power architecture is heavily dependent upon the requirements of a particular science mission. For the example given above, a long duration surface mission is not needed to accomplish the desired scientific measurements. In this case, a large capacity primary battery was chosen for the power system, coupled with a master/slave processor configuration, in which near-continuous instrument operation is managed by a low power micro-controller, while the master processor is active only for brief periods, to perform data management and uplink/downlink operations with a Mars-orbiting relay asset. For missions in which long life is paramount, though, a solar array coupled with a secondary battery may be most appropriate. Options for both types of power systems were developed in the Rough Lander concept study, to provide flexibility in accommodating the needs of different future missions.

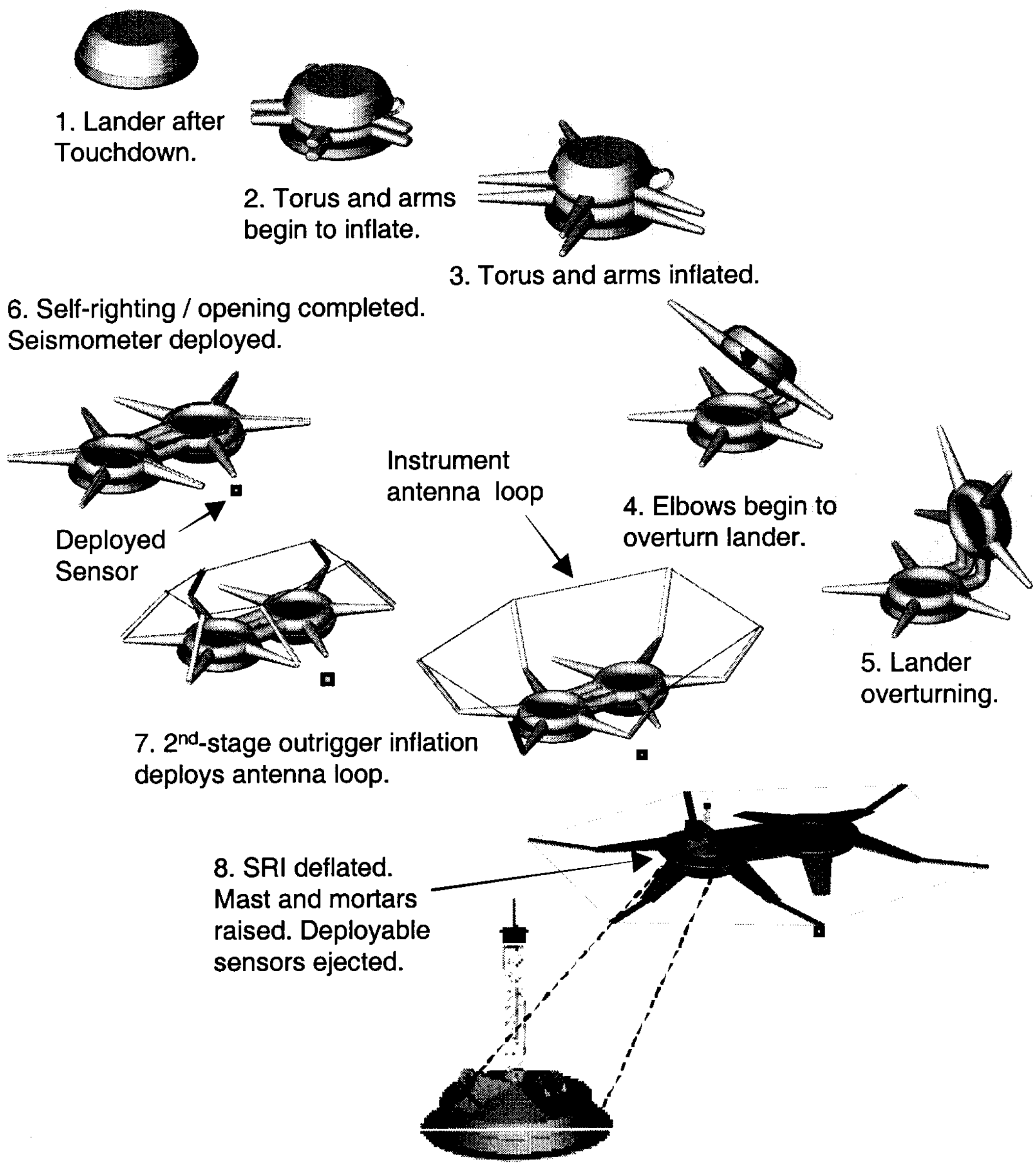


Figure 6: Self-Righting and Instrument Deployment Sequence

Lander System Description

The Lander system design incorporates proven components into a single, compact configuration capable of accommodating a variety of science instruments. The vehicle is capable of operating in a wide range of landing sites with relatively low sensitivity to variations in environmental conditions. A block diagram illustrating the principal subsystems is shown in Fig. 7. An expanded view of the physical packaging of the Lander housing is presented in Fig. 8.

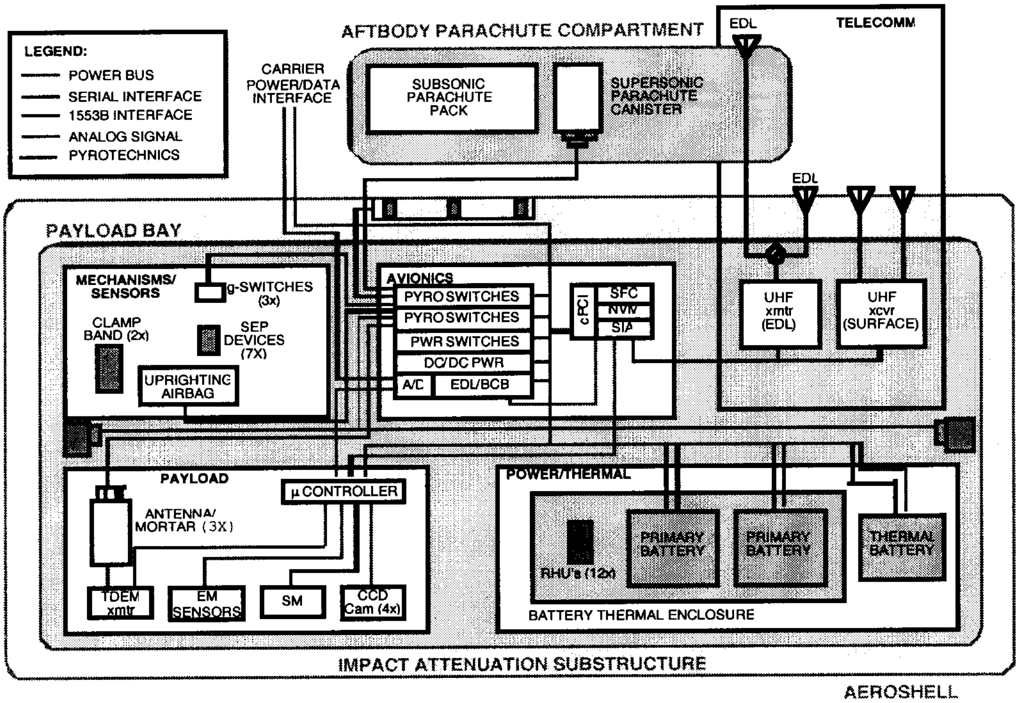


Figure 7: Lander System Block Diagram

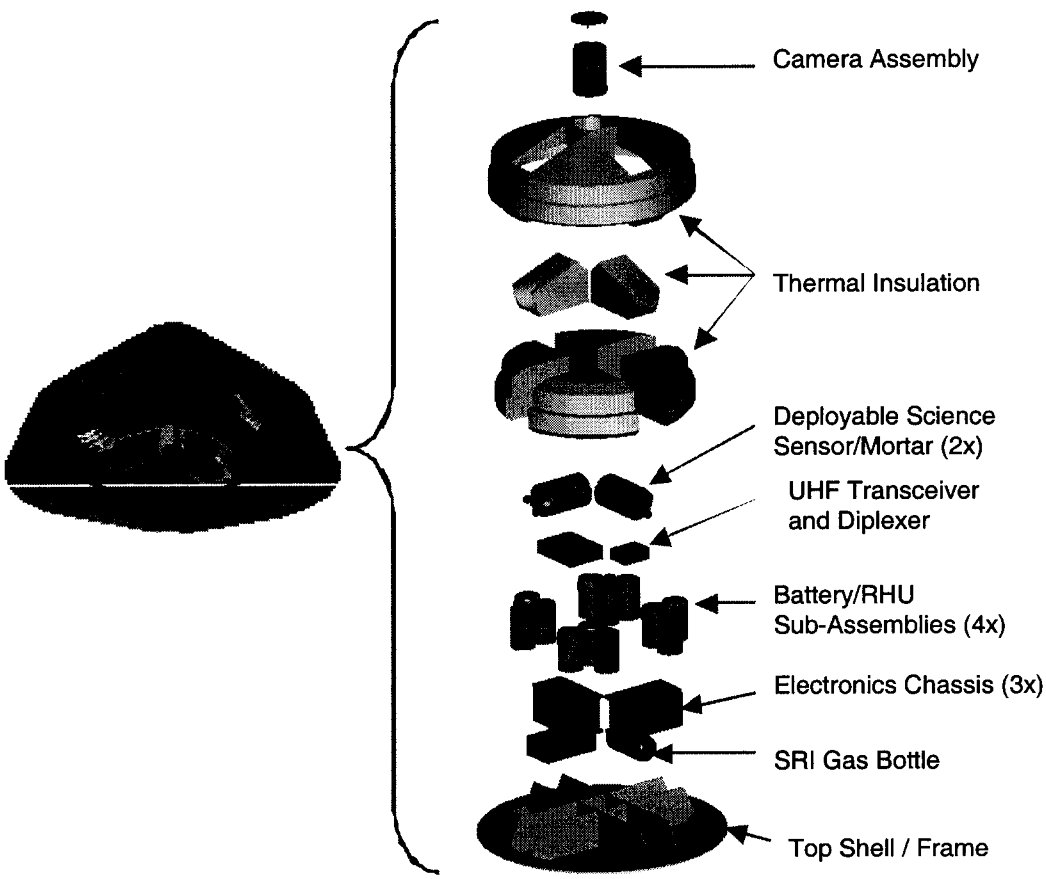


Figure 8: Lander Housing (Payload Bay) Expanded View

Avionics Subsystem

The avionics subsystem employs select elements of the X-2000 system (originally developed for the Europa Orbiter mission), coupled with several custom-designed components to perform specialized tasks. A single-string design was adopted due to the short duration of the surface mission. Two processors operate in a master/slave relationship to provide for near-continuous instrument operation at very low (< 1 W) power levels. The master processor is a RAD-750 CPU, integrated into the X-2000 System Flight Computer (SFC) card, while the slave processor, a Mongoose V CPU, serves as a dedicated low-power instrument controller. Other X-2000 components employed include a 2.0 GB Non-Volatile Memory (NVM) card, and a System Interface Assembly (SIA) card, which supports both synchronous and serial interfaces with the other components. Features of the X-2000 components originally incorporated for radiation shielding make these cards particularly well suited for the Rough Lander. The radiation shield panels in the cards will be replaced with mechanical stiffeners, giving them the capability needed to withstand a 2,000 g landing without further modification.

Power Subsystem

Electrical power is provided by two Lithium Thionyl Chloride primary battery assemblies, along with the appropriate DC/DC conversion elements (these can be tailored to the specific components and instruments that need to be accommodated in a given mission). Each battery provides 3,000 W-hr of energy storage at 0° C, and 1,500 W-hr at -20° C. This battery was developed for the Titan/Centaur upper stage to be both rugged and reliable, and has already been qualified to withstand a 3,000 g shock load.

Thermal Control Subsystem

The Lander's internal configuration provides two types of thermal enclosures. The primary enclosure creates the externally visible enclosure and keeps the avionics and science instruments within acceptable operating temperature limits (-40° to +50° C). Four secondary enclosures provide additional insulation to keep the batteries above -20° C. Under typical conditions for a landing in early Martian summer at mid-Latitudes (+30-45° N Latitude for northern summer or -30-45° S Latitude for corresponding southern summer conditions), 12 Radioisotope Heater Units (RHU's) with 1.0 W output each are sufficient to keep the batteries above -5° C overnight.

Telecommunications Subsystem

During surface operations a UHF-band transceiver, coupled with a mast-mounted monopole antenna, enables high-rate communication with Mars orbiting relay assets, such as the Mars Reconnaissance Orbiter (MRO), planned for operation in Mars orbit in the period 2006-2011. Using MRO as an example relay asset, the Lander is capable of transmitting at data rates up to 128 kB/s, and up to 72 MB/Sol (Martian Solar Day) total data volume assuming two 5 min. daily transmissions to MRO. The Lander is also equipped with a separate UHF transmitter coupled with a single low-gain antenna embedded into the backshell, which is optimized to support near-continuous real-time communication with the parent carrier vehicle or orbiting asset during entry, descent, and landing.

SUMMARY/CONCLUSIONS

A new concept for a class of small, low-cost landers using a rough landing approach for exploration of the Martian surface has been described. Design options and considerations for

Scout-class landed missions were discussed, driven by a target set of capabilities established for this type of mission. The proposed EDL system employs an innovative combination of proven components and subsystems to limit the touchdown load factor to 500-1,600 g, capable of safely accommodating a variety of electronics, and science instrumentation, while enabling a substantial reduction in EDL system complexity and cost compared to previous soft landing systems. Either solar array/secondary battery or primary battery power systems can be used in the Lander system for surface operations, depending upon specific science mission requirements. This type of vehicle may enable initial exploration of relatively rugged regions of the Martian surface that are otherwise unreachable or deemed too risky for all but the most sophisticated future soft landers

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BIOGRAPHY

Sam Thurman received his BS and MS degrees in Aeronautics and Astronautics from Purdue University (1983) and Massachusetts Institute of Technology (1985), respectively, and a PhD degree from the University of Southern California in Aerospace Engineering (1995). Over the past 15 years he has worked in a variety of positions at Jet Propulsion Laboratory, ranging from applied research and development to spacecraft design, systems engineering, and mission operations. He was the entry, descent, and landing systems engineer for NASA's Mars Pathfinder mission, for which he later received the NASA Exceptional Achievement Medal. He went on to work as Project Engineer and later Mission Manager for both the Mars Climate Orbiter and Mars Polar Lander missions. Since then he has served as the Entry/Descent/Landing systems manager and later Deputy Manager of the Mars Smart Lander Project. Most recently he has participated in several proposal development efforts for new missions in NASA's New Millennium and Mars Scout Programs.

Tom Rivellini received his BS degree in Aerospace Engineering from Syracuse University in 1989 and his MS degree, also in Aerospace Engineering, from the University of Texas at Austin in 1991. He has been working at Jet Propulsion Laboratory since then, primarily in the development of Martian entry, descent, and landing systems. Tom has worked on several different projects within NASA's Mars Exploration Program including Mars Pathfinder, Deep Space-2, Mars Sample Return and Mars Exploration Rover. He was the lead engineer for the airbag subsystem in the Mars Pathfinder mission, for which he was later awarded the NASA Exceptional Achievement Medal. In addition, Tom was the recipient of the AIAA Engineer of the Year award in 1998 for his role in Pathfinder. He went on to serve as the lead mechanical engineer for the New Millennium Deep Space-2 Micro-probe mission. Most recently, he has been leading the development of an advanced landing system for the Mars Smart Lander Project.