SSC98-X-5

$3M Planetary Missions: Why Not?

Consideration of deep-space spacecraft mission requirements

Tomas Svitek
Stellar Innovations, Inc.
1436 Clover Creek Dr.
Longmont CO 80503
(303) 651-7580 (voice), 651-6757 (fax)
tomas@stin.com

The dramatic cost reduction of the Earth orbiting spacecraft has become the established fact, over the period roughly coinciding with existence of this conference. 10-15 years ago, the median cost of the spacecraft mission was around $100M (in today's dollars). One-million dollar missions were unheard of, except in the amateur radio community.

Today, missions with the total cost of under $10M are common. Besides the well-established amateur radio programs, many low-cost university-led spacecraft programs took place. Plenty of other science, technology, experimental and know-how technology transfer programs have been or are being implemented.

The last remaining frontier of the low-cost mantra are the deep space missions. The great progress has already been achieved in reducing the cost of planetary exploration but no credible mission was ever seriously considered under $40M (the lowest-cost examples are Clementine 1 and Lunar Prospector, both well over that limit). The minimum cost of planetary missions is about a factor of ten higher than for Earth-orbiting missions with roughly similar capabilities and lifetimes.

Why is that? We will address this question in this paper.

The answer to this question appears quite obvious: Of course, the deep-space missions are more difficult than the LEO missions. But we will try to show that this is not inherently true. Step by step, we will analyze and compare requirements between the deep-space and Earth-orbiting missions, note the differences and provide estimates of cost impact.

There are some legitimate complications involving the deep space mission requirements that would command the cost premium for a deep-space project when compared with a similar Earth orbiter. But, we will argue that this premium is nowhere as large as commonly perceived.

Why is this misconceptions occurring? We do not really know precisely and can only speculate. Knowledge and design aspects of the deep space environment have not been as widely disseminated as those for the LEO environment. Or perhaps, it is for a historical reason: it *used* to be significantly more difficult and that assumption has never been questioned again. Or, maybe, it is the exclusive-club issue: there are many more teams that have put together the LEO spacecraft, much more than a deep space mission. Or, it is just a fear of distant unknown places.

Introduction

The purpose of this paper is to provoke the debate about how to best design the low-cost missions for exploration of our solar system. The paper was written as honest and frank analysis to foster this debate. This effort was focused on generic mission requirement analysis and is trying to avoid specific design descriptions.

Also note that this paper is not discussing the extremely difficult missions that are challenging the technology state-of-the-art (Sun probe, Europa or Kuiper Express, etc.). Rather, the topic of this discussions are the types of the missions that are often proposed for the Discovery program (Venus, Mars, comet or asteroid missions).

Finally, towards the conclusions, the paper will try to show that Mars Observer, Galileo and Clementine failures would occur exactly the same way if the identical spacecraft was operated in the LEO environment in the same way. This is done to rationally address the issue of the LEO-orbiting spacecraft hardware heritage for the deep-space missions.

The overriding philosophy of the design approach that is implicitly advocated in this paper could be called a *design-to-complexity* (in contrast to the design-to-cost or design-to-performance). This approach is based on unwavering belief in simplicity and the fact that complexity and reliability are inherently and inversely related, in a strong and steep functional dependence.

Propulsion

The significant propulsion requirements are the first obvious distinguishing characteristic of a deep-space mission, as compared to the low-cost LEO missions that often are often not even equipped with a propulsion function. Clearly, the propulsion subsystem cost must be included in the design of any low-cost mission.

Two low-cost launch modes have been analyzed in detail in the past (ref 1):

1) Pegasus (SELV) launch to the LEO parking orbit with the interplanetary injection accomplished by a small spin-stabilized solid rocket motor, typically Thiokol Star-27 or Star-30 (the launch cost of this option would greatly exceed the desired mission cost target, obviously).
2) Ariane piggyback (ASAP) launch into the GTO and subsequent interplanetary injection with an integrated liquid propulsion (this option could be compatible with the desired mission cost target).

For the Pegasus/SRM launch, the minimum spacecraft propulsion requirement must support the required trajectory correction maneuvers after the SRM burn (typ. 80-140 m/sec) and successive mission propulsion requirements (deep-space burns, target orbit insertion and maneuvering). For the Ariane ASAP launch, this propulsion requirements includes the orbit transfer from GTO into a interplanetary trajectory (typ. 800-1400 m/sec).

This could be accomplished with a simple blowdown monopropellant system that also supplies the attitude control torque requirements (by tapping off from the pressurant gas supply). As an illustration, Table 1 shows the cost breakdown for the lowest-cost monopropellant propulsion subsystem that could be credibly procured from an established propulsion supplier (the cost of the bipropellant option would be approximately 2.5-4x higher).

Communications

The communications subsystem design for a deep-space mission clearly differs from the LEO mission. Extra 100+ dB of link losses is hard to argue with. But that is only half of the story. It turns out that the NASA Deep Space Network (DSN) communications performance is so tremendous, particularly in contrast to the often marginal and inferior ground station 'kludges' that are so often used for low-cost LEO missions (because of cost constraints, lack of attention, or the last minute effort). Additionally, there is a real problem of using the single ground station to communicate with the LEO spacecraft in 2 brief passes per day.

Table 1. Estimated Total Propulsion Subsystem Cost for a Simple Deep-Space Mission

Hardware item or task	Cost ($K)	Note
Small propellant tank (2500 cu in)	120-240	off-the-shelf (no mods), w/bladder
Four monopropellant thrusters	4 x 15-25	could be only one for a spinner
Attitude control thrusters (12)	160-220	incl. plumbing (could be only 4 for a spinner)
Pyro valve (2)	2 x 15	for hydrazine and ACS thrusters isolation from tank (range safety)
Filter (2)	2 x 5	downstream from pyro valve
Latch valve (2)	2 x 30-60	leak isolation after pyro valve firing
Misc. materials	40-60	tubing, fittings, etc.
Subtotal – hardware	510-810	
Subsystem engineer	100	½ man-year (senior engineer)
Mechanical layout	80	½ man-year (junior engineer)
Support structure fabrication	60-120	depends on complexity, mostly conventional materials (Al, GrEp)
Subsystem welding	80	specialized vendor
Finish subsystem integration	40	
Structural and thermal analysis	80	½ man-year
Qual testing support	80	2 people x 3 months, done at system level (vib/shock, thermal-vac)
Range safety documentation	50-150	range dependent
Fueling (planning)	60	
Fueling (execution)	100-200	subcontracting the task
Subtotal – labor + subcontracts	730-970	
Total	1500 ± 20%	higher with customization, lower with relaxed performance requirements

The comparison between a) simple deep space mission communications and b) LEO spacecraft communications is illustrated in the two tables below. The first table compares downlink communication margins, with the following assumptions:

- the daily downlink data volume is 50 Mbit which has to be transmitted in one 4-hour DSN pass or two brief LEO communications passes (instantaneous data rate is adjusted for the downlink session duration),
- the deep-space spacecraft uses the X-Band
- the LEO spacecraft uses more typical S-Band frequency,
- the transmitter DC power consumption is identical in both cases (10 W),
- the deep-space spacecraft uses a 30-cm reflector antenna for the downlink (the spacecraft is earth-pointed during the downlink session)
- the LEO spacecraft has a small patch antenna which is more typical for this class of missions and is required due to the varying geometry between the spacecraft and ground station during the single pass,
- the deep-space spacecraft uses the DSN standard concatenated coding whereas the LEO spacecraft signal is not coded (as is typical), and
- the deep-space spacecraft downlink signal is received through the 34-m BWG antenna (the most common antenna in the DSN complex)
- the LEO spacecraft ground station uses a typical steerable antenna with the gain limited by minimum beamwidth that is compatible with the orbit prediction accuracy.

Table 2. Comparison of Typical Deep-Space Mission and LEO Mission Communications Downlinks

Downlink	Deep-space	LEO
Frequency [MHz]	8425	2200
Transmitter DC power [W]	10.0	10.0
Transmitter efficiency [%]	25.0	35.0
HGA diameter [cm]	30.0	N/A
HGA efficiency [-]	0.55	N/A
Transmitter antenna gain [dB]	28.6	3.0
Transmitter EIRP [dBW]	32.6	8.4
Path range [km]	1.5E+08	2200.0
Free path loss [dB]	-274.5	-166.1
HGA pointing loss [dB]	-0.5	0.0
Polarization loss [dB]	-0.2	-3.0
Ionospheric loss [dB]	-0.2	-0.3
Atmospheric loss [dB]	-0.3	-0.2
Signal strength [dBm]	-213.1	-131.2
Receiver antenna gain [dB]	68.2	15.0
Receiver noise temp [K]	28	150
Data rate (for 50 Mbit/day) [kbps]	4.0	59.5
Actual Eb/No [dB]	3.2	12.9
Demodulation loss [dB]	0.0	-2.0
Required Eb/No [dB]	0.8	10.0
Data link margin [dB]	2.4	2.9

The second table compares downlink communication margins, with the following assumptions:

- two possible DSN uplink approaches are shown: X-Band uplink from a 34-m antennas (recommended approach by DSN) and S-Band uplink from a 70-m antenna (for real emergencies),

- the deep-space spacecraft X-Band receiver (if used) would be fed a low-gain antenna
- the S-Band receiver (if used) would be fed by a true omnidirectional antennas,
- the LEO ground station hardware represents a setup that is typical for many of these low-cost missions, and
- the uplink data rate for the deep-space spacecraft is higher than is typical for these missions; this is due to limitations of a low-cost receiver (lower data rates would require narrow-band receiver tracking which is difficult and expensive).

Table 3. Comparison of Typical Deep-Space Mission and LEO Mission Communications Uplinks

Uplink	DSN	DSN	LEO
Frequency band [-]	X-Band	S-Band	
DSN antenna	34-m	70-m	
Frequency [MHz]	7170	2090	2200
Transmitter power [W]	2000	20000	50
Transmitter antenna gain [dB]	68.1	62.7	8.0
Transmitter EIRP [dBW]	101.1	105.7	25.0
Path range [km]	1.5E+08	1.5E+08	2200.0
Free path loss [dB]	-273.1	-262.4	-166.1
Polarization loss [dB]	-1.5	-1.5	-3.0
Ionospheric loss [dB]	-0.2	-0.3	-0.3
Atmospheric loss [dB]	-0.3	-0.2	-0.2
Signal strength [dBm]	-144.0	-128.7	-114.6
Receiver antenna gain [dB]	4.5	-3.0	-3.0
Receiver noise temp [K]	120.0	240.0	240.0
Data rate [bps]	200	200	2400
Actual Eb/No	15.3	20.1	23.3
Required Eb/No	6.0	6.0	12.0
Data link margin [dB]	9.3	14.1	11.3

The following conclusion can be drawn from this analysis: The communications subsystem for a small deep-space mission with the moderate requirements is not significantly more expensive than a comparable subsystem for the LEO mission under these two constraints:

- established DSN performance is fully utilized, and
- navigation requirements (discussed below) are satisfied.

The deep-space communication subsystem design imposes some additional requirements (compared to the LEO missions) on other subsystems. For example, attitude control subsystem must be capable of pointing the high-gain antenna towards the Earth and the spacecraft software must make the decision about switching between low-gain and high-gain antennas. But these requirements are quite reasonable.

This analysis would change, of course, if the mission planners would choose not to rely on DSN and use alternative ground communications network that has less performance (thus resulting in placing more burden for the communication performance on the space segment). This could be a reasonable tradeoff for some missions. The conclusions would also change by a requirement for delivering substantially larger data volumes or for a mission to the more distant target in the solar system.

Navigation

The navigation is another unique deep-space function. The state-of-the-art navigation performance for deep-space missions can deliver the spacecraft to a point anywhere in the solar system with roughly 1-km error (assuming that the location of the target is known to the same accuracy). Typically, a low-cost deep-space mission will not require such an exacting navigation accuracy and the reliance on lower-cost navigation techniques is possible.

The navigation function relies on these measurements: range, range-rate, Doppler (one or two stations) or optical imaging. The range-rate measurement is the most commonly used measurement but requires a coherent spacecraft transponder that is expensive (around $2M for the X-Band unit). The lower-cost solutions would rely on:

1) the coherent S-Band transponder that costs about one third of the X-Band unit (used by Clementine, Lunar Prospector, Stardust and Genesis), or
2) the one-way Doppler measurements that are occasionally supplemented by the two-station differential Doppler measurement (this was tested successfully by Magellan and seriously considered for Mars Pathfinder).

The second solution does not require the coherent transponder and thus, the receiver and transmitter can be procured independently to minimize the cost (for example, off-the-shelf S-Band receiver and X-Band transmitter). This navigation method requires a stable oscillator source on the spacecraft but there are several solutions to this requirements:

a) low-cost low-power ovenized crystal oscillator (not radio-science stability, rather only navigation capability with Allan variance of 1E-10 vs. 1E-13),
b) dual-frequency crystal oscillator, or
c) technique proposed recently by APL for the Contours mission (ref 2).

In any case, the spacecraft navigation will, most likely, be supplemented by the optical navigation measurement using either the spacecraft star tracker or payload camera images. The methods of the optical navigation (for a moderate accuracy) are relatively mature.

Attitude Control

The attitude control subsystem design for a modest deep-space mission is relatively straightforward and with no surprise (unless some unique requirements or functionality is demanded of the mission). The deep-space missions cannot use the common (for the LEO missions) complement of the attitude control sensors and actuators: magnetometer, horizon sensors, magnetic torquerods, and gravity gradient booms. Rather, the attitude control hardware suite of a low-cost deep-space mission consists of:

- coarse sun sensor (used only for fault recovery),
- moderate-accuracy star tracker (1-3 kg mass, typ. \$500K/unit),
- moderate-accuracy inertial reference unit (most likely fiber-optic design, 1 kg, \$100K or less), and
- set of 12 attitude control thrusters (really a part of the propulsion subsystem).

The major trade-offs for the low-cost deep-space attitude control subsystem configuration are:

1) Complementing the attitude control thrusters with a set of 3-4 small reaction wheels (1 kg and \$50K each). However, a real (not knee-jerk) justification for the reaction wheels is difficult for the majority of low-cost missions (less than 5 year lifetime and no extensive attitude maneuvering).
2) Selection of the attitude control thruster propellant: hydrazine or cold-gas. This choice is very mission dependent and represents a typical mass vs. cost tradeoff vs. minimal impulse bit.
3) Opting for a gyroless operation: relying exclusively on continuous star tracker measurement and eliminating the requirement for inertial reference sensor. Currently, this is not a desirable trade-off (gyroless operation has not been proven reliably whereas the mass and cost and reliability of inertial reference units has improved dramatically).
4) Adding the horizon sensors if the mission ultimately becomes an orbiter. This choice does not simplify the attitude control subsystem design and would be, most probably, only used as a backup and for the functional redundancy.

It should be emphasized that an attitude control software for an inertially-based system (like the one described in the previous paragraph) is inherently simpler than for an Earth-orbiting spacecraft (this is in contrast to the hardware that is often simpler for the Earth-orbiting spacecraft). The Earth-orbiting attitude control software must deal with complex relationships: The attitude sensor suite is different for a Sun-lit and eclipsed portions of the orbit. The horizon sensors measure only in two axis and must be complemented with the magnetometer measurements (but that works only in a fraction of the orbit). Often, the raw sensor accuracy is inadequate and must be sanity-checked and heavily filtered. Any magnetic actuator commands are complicated by the fact that magnetic torquers have two active axis that are different throughout the orbit. Sometimes, the spacecraft cannot be controlled in a desired axis, and, almost always, the magnetic torquer command has an undesirable component.

The situation is much simpler for an inertially-based (deep-space) spacecraft: Both the star tracker and the inertial reference unit always produce the three-axis measurement. Mostly, the small angle approximation applies and each axis can be controlled by its independent set of thrusters (or a reaction wheel). The gyro bias compensation approach is well-understood. A typical attitude control subsystem software for a low-cost (simple) deep-space mission can be accomplished in less than 1000 lines of high-level code (excluding the star tracker algorithm).

Further simplification can be achieved if the proposed spacecraft can be a spinner throughout all the mission phases (for example, a cruise stage for delivery of an entry probe into planetary atmosphere). In that case, the attitude control hardware complements is simplified to:

- V-slit star/sun crossing indicator that replaces the imaging star tracker as well as multiple coarse sun sensors (replacing a framing CCD camera with a single photodiode is very desirable trade-off),
- inertial reference unit is not essential,
- only four-thruster configuration is required, and
- simple nutation damper must be added (maybe as simple as adding the damping vanes to the propellant tank).

Clearly, the spinner attitude control subsystem has a major advantage in simplicity, fault tolerance and robustness. This solution enables the highly-reliable attitude control design that is low-cost, and has proven its long-life on a number of the previous missions. (It does not mean that a three-axis attitude control cannot function reliably for the long time, just that it is difficult to achieve that with a low-cost design.)

Thermal Control

Spacecraft thermal control requirements are often highlighted as a presenting the special difficulty in designing the low-cost deep-space mission. This is a partial misconception. Clearly, there are a number of potential deep space missions with the severe thermal control requirements (Mercury orbiter, Kuiper Express, etc.).

But the thermal design of a small LEO spacecraft is no trivial task, neither (and it has been repeatedly and successfully accomplished). A LEO spacecraft must live through several tens of thousand of severe thermal cycles. This difficulty is further accentuated by a low-value of the thermal time constant for a typical small spacecraft that is often comparable to the orbital period.

This is the crucial argument: The difference between the environmental input with the full solar illumination (incl. Earth albedo) and the eclipse (only the Earth thermal radiation input) is equivalent to conditions in an interplanetary cruise that varies from 0.78 AU to 3.4 AU. In another words: the current LEO smallsat experience in varying the environmental input covers deep-space conditions from a cruise to Venus to the main-belt asteroid missions. Thus, the spacecraft thermal control for such missions is a challenging problem but it should not be a mission cost driver.

Radiation Environment

The effect of the ionizing radiation on the spacecraft functionality is frequently invoked as an explanation for absence of any truly low-cost deep-space mission. With a brief analysis based on facts, the situation is not as bleak:

1) *Total ionizing radiation dose* (TID) is remarkably low for the majority of moderate deep-space missions (missions to the Jupiter system or close to the Sun are the striking exceptions). The LEO missions are often exposed to higher TID because of the van Allen belts. In a typical near-Earth environment, background TID is less than 1 krad/year after 100 (Si) mil. A great (once per cycle) solar flare adds another 1 krad. If the mission relies on the piggyback Ariane ASAP launch to GTO, the exposure from repeated passes through van Allen belts will add another 1 krad. In summary, it is difficult to imagine a low-cost deep-space mission (besides the few obvious exceptions) that would anticipate much more than 5 krad of the actual dose. This dose results in the design requirement of 15-20 krad, at most, that is quite compatible with the bulk of current electronics technology that is being adopted for space applications.
2) *Single-event upsets* (SEUs) occur in the space environments, although often less frequently than predicted before the launch. Any spacecraft designer must deal with mitigating SEUs, no matter how radiation-hard electronics is used (the fact is that there is no truly SEU-resistant modern 32-bit processor). The only issue is the frequency of the event. The SEU mitigation measures are well understood and widely adopted: a) memory error detection and correction codes, b) frequent memory and register content scrubbing, c) hardware watchdog timers and hardware timeout counters, d) command message double-verification and other software defensive techniques).
Occurrence of the SEU events and recovery from them represent no threat to the mission success during the overwhelming majority of the mission, with the exceptions occurring around the propulsion maneuvers. In those few mission-critical situations, the processor recovery is essential, or even better, a simple hardware-only is the preferred solution (if at all practical). Notice that the similar defensive techniques must be adopted for the software failures, as well. In the today's spacecraft, the processor is actually more likely to crash because of bad software than due to a radiation-induced SEU. But that does not prevent most program managers to worry 90% of time about the radiation and only 10% of time about the flight software quality.
3) *Single-event latchup* (SEL) seriousness and recovery techniques have developed the least amount of industry consensus, so far. The simple fact is that there is yet to appear a credible published report about the destructive SEL occurring on orbit. There are some speculative reports of destructive SELs, but with other possible and more likely explanations. It can plausibly occur but the issue at what actual rate. And with the realistic SEL estimates, it is not obvious that the proposed SEL-mitigation techniques (active current switching) are the higher-reliability solution (i.e., the mitigation measures introduce more failure modes than they solve). Non-destructive SELs are known to occur and must be recovered from. Fortunately, the same measures as for the SEUs and software crashes (hardware time-outs and hard restart) will also recover from a non-destructive SEL. The more advanced (commercial) electronics that uses

lower voltage supply and less power will also prevent more SELs from damaging the electronics.

In the summary about the radiation effects: The effect of ionizing radiation on the spacecraft system must be obviously considered and protected against, in some cases. But the facts are:

- the most planetary missions operate in the TID environment that is no any different from the LEO mission environment (in fact, sometimes even more benign, particularly for missions away from Sun),
- the mitigation measures for dealing with SEUs (and non-destructive SELs) are well-understood and widely implemented and present no serious hazards to the spacecraft operations (the real challenge in this area is devising methods for the high-fidelity system testing of the SEUs mitigation measures), and
- the destructive SELs are not worth worrying about for the majority of low-cost deep-space missions (in fact, most currently implemented measures create higher hazard to the mission success than ignorance).

Fault Detection and Management

The fault detection and management was a unique achievement of the deep-space spacecraft design community (specifically, NASA Jet Propulsion Laboratory). Thanks to their pioneering work, these fault-handling approaches have been widely disseminated and adopted within the industry (more on new smallsat missions, less on more mature established programs).

This presents us with a good news: The previously-mysterious domain of implementing the fault detection and management techniques creates no special barrier to a low-cost spacecraft development. The detailed work of the algorithm design and (more importantly) testing still must to be done to a specific circumstances of each mission, however.

But this effort consists of mixture of a) good system engineering practices, b) enthusiastic focus on simplicity, and c) willingness to learn from the past (needless to say, many low-cost and high-cost missions lack any or all these three attributes).

Typically, the fault detection and management approach would be designed around these specific problems:

1) Loss of attitude knowledge (i.e., on-board attitude estimator does not provide self-consistent output): restart attitude determination from scratch (incl. Kalman filter and star tracker algorithm initiation). If the problem persists, reorient the spacecraft towards Sun with its solar panels (*hunt for sun* mode)
2) Power bus undervoltage: perform priority-based electric load-shedding.
3) Excessive battery discharge: also perform priority-based electric load-shedding. If the problem persists and worsens, enter the *Phoenix* mode (the whole spacecraft electronics is off except for the battery trickle charge circuits).
4) Propulsion anomalies: loss of pressure (cycle and close-off all valves), redline temperature (stop propulsion activity and turn-on heaters if required), excessive thruster firing duration and excessive valve cycling (close off thruster valves with the hardware-based watchdog circuit).
5) Ground command / contact time-out: switch to the redundant receiver, eventually reorient the spacecraft towards the Sun.
6) Loss of flight software integrity (caused by a loose software pointer or by SEU and detected by the data buffers checksum error or successive memory read errors or unexpected interrupt or unexpected subroutine entry or mission mode violation): soft restart of the processor. If the problem persists, attempt the hard restart and then execute software from ROM.
7) Loss of on-board processor heartbeat (either in the primary processor or from distributed nodes): soft and then hard processor restart. If the problem persists, carefully attempt redundancy (if any) switching.

Many of the fault detection algorithms would disabled during certain mission-critical events (e.g., launch and initial injection, planetary orbit insertion, aerobraking phase, etc.)

Redundancy, Complexity and Mission Reliability

The block redundancy and functional redundancy have been the hallmarks of the most deep-space missions in the past. From the extensive analysis of the many recent low-cost mission successes and failures (by us and others), the continuing reliance on redundancy to assure the mission success must be questioned. Redundancy (as reflected in Mil-Hdbk-217F) aims at mitigating random parts failures. But the most root causes of modern spacecraft failures are in the areas of poor design, misapplied parts, poor integration workmanship, and faulty software. None of these causes are reflected in Mil-Hdbk-217F. Additional problems with the Mil-Hdbk-217F-type analysis are because of:

- focus on a simple parts count; but what fails are *interfaces* between parts, not parts themselves (that is why adding block redundancy appears so beneficial in Mil-Hdbk-217F but rarely pays off in the practice),
- conceptual reliance on failure rate dependence on the static temperature (i.e., Arrhenius equation); it has now been firmly established that the Arrhenius equation approach does not apply to a complex systems (like IC die, bonds and package assembly); also, the dynamic temperature (and rate of the temperature range, from the power cycling, for example) is much more significant contributor to part failure rates than a simple static temperature (ref 3),
- not considering the combination of stresses (e.g., temperature and humidity) that create much higher parts failure rates than individual stress mechanisms alone ,
- total lack of consideration for the hardware design quality and design review depth as well as no consideration for the fidelity and extent of testing (this approach assumes the perfect design and test process which is far from true for all the missions),
- ignorance of how appropriately are the parts used and misused (many electronics parts experience dramatic variations in the failure rates depending on their actual application in a circuit), and
- neglecting any impact that has the software has on the system reliability (currently, software is the leading cause of spacecraft catastrophic failures).

Hopefully, this discussion convinced the reader that not only is the Mil-Hdbk-217F-type analysis a useless exercise, it is in fact a dangerous practice because it will drive the system design towards a less reliable design point.

The only meaningful correlation with the mission success rate is the clarity of development requirements and simplicity of the design. This statement does not apply that a complex sophisticated design cannot be reliable. Rather, the statement claims, based on actual flight experience, that the high cost the of complex missions does not statistically guarantee its reliability. And the low-cost of simple missions is quite consistent with high success rate.

Actual Redundancy Experience

It has been suggested (ref 4) that the uplink/downlink hardware redundancy was essential for the mission success of the five recent JPL planetary missions (out of total six considered). However, the author of this paper cannot draw the same conclusions from raw data presented in the study if the conclusion implies recommendation for the future missions.

Failures of mechanical tape recorders and discrete logic (CD4000 series) electronics must be removed from the consideration because these types of failures are not relevant to the modern spacecraft. Further, the Galileo HGA failure cannot be counted in this contact (it will be discussed in more detail below). Thus, four RF hardware failures occurred between six missions (one of them 15 years after the launch and thus, again, irrelevant to any low-cost mission design). The Voyager 2 receiver capacitor failure occurred on the primary unit, the switchover to the backup unit did not work (it failed catastrophically right away, for an unknown reason) and thus the spectacularly successful mission was accomplished on a partially-failed primary receiver (i.e., receiver redundancy had absolutely no effect on chances of the mission success).

This leaves the two Magellan transmitter failures that occurred at 2 and 3 years into the mission. In summary, based on this reading of raw data, the block redundancy extended the lifetime of one mission (our of total six missions) by one year. The redundancy impact on mitigating other failures is questionable (admittedly, this is a different interpretation that the original study presented).

Compared to this benefit, the redundancy has the following negative impacts:

1) The obvious growth in the hardware mass and cost: some mission simply cannot afford this impact.
2) Increased system complexity because of the additional hardware and software that is required for the redundancy switching and that requires additional testing.
3) Reduced testing time in the A-A block configuration (because A-B, B-A and B-B block configurations must be also tested).
4) Substantial software reliability degradation due to the complexity growth. Software complexity and its reliability (resp. lack of it) are tied exponentially.

This negative impact must be fully considered and quantitatively assessed before committing to the redundant design due to its postulated benefits. It should be noted that the root causes of both Voyager-2 and Magellan RF failures have been noted and known during the ground test before the launch (inadequate design and misapplied part). Thus, it could be argued that, given the fixed budget, a project is better off fixing the problems discovered during the ground test, rather than developing a fully redundant design to recover from the faults discovered during the ground test.

Finally, it must be stressed that many of the moderate low-cost deep-space missions that are considered in this paper, do not actually have too stressing lifetime requirements. Such a typical Mars or Venus or asteroid missions have a lifetime requirements of 2-4 years which is well within an experience base for a simple single-string design (perhaps, with some carefully justified redundancy).

Adopting Hardware from Earth-Orbit Missions

It has often been noted that adopting the heritage hardware from an Earth-orbiting missions for the deep-space missions has serious reliability implications. Two the most often invoked examples are Mars Observer and Galileo. Because adopting already-developed hardware (and software) is very essential in achieving the low-cost mission design of any kind, some reflection on MO and Galileo experience seems to be warranted.

The *conclusion* first: Adopting this hardware had serious reliability implications but not for an obvious reason. Typically, a deep-space mission project adopted any heritage hardware only after the significant modifications. It could argued that if the design would left alone and hardware used as it was, the project would be better off.

For example, it has not been widely recognized that the Galileo high-gain antenna was substantially modified from its TDRSS predecessor. Thus, ten successful TDRSS deployments have no relevance on the Galileo deployment success or failure. The most significant change of the Galileo design (from the deployment reliability viewpoint) was a requested change from a U-shape groove for antenna ribs on the TDRSS antennas to a V-shape groove for Galileo. Intuitive analysis indicates that a round-cross-section rib in the V-groove would experience more point pressure and more lubrication removal, than a rib in the U-groove. It can be further speculated by extrapolation that a JPL-modified antenna might have failed as well on the TDRSS mission and that the original TDRSS design would definitely have better chance of functioning properly on the Galileo mission given its special circumstances. This is, of course, a speculation but it indicates that a *unique* deep-space requirements had less impact on the antenna deployment reliability than unilaterally requested design changes to the proven hardware.

The Mars Observer case could be argued is a similar way. The MO failure report (ref 5) identified the three most likely causes of the spacecraft failure:

- *Liquid oxidizer mixing* with the fuel. It turns out that a minute mixing of fuel and oxidizer *vapors* occurs on many spacecraft. It is not desirable but it does not cause any real damage. The original RCA design, based on their typical configuration, did not have pyro valves in the pressurization system and thus would allow a slow vapor migration and mixing. The JPL project requested the addition of pyro valves in the pressurization system to improve the system reliability. Ironically, it had an opposite effect: the pyro valves blocked the oxidizer vapor migration, thus forcing oxidizer condensation and its collection for extended period of time, and later injection of liquid oxidizer into the fuel tank. It would appear that the original (unmodified, Earth-based) would have a better chance of the mission success.
- *Transistor failure in oscillator unit.* This unit has been used on many RCA-designed spacecraft for over 15 years. Before this unit was used on MO, it was requested to upgrade one of its transistor to a higher, more-reliable, grade. Ironically, this upgraded transistor turned out to be from a bad batch and could have caused the mission loss. This particular failure scenario would be also less likely if the original design would be left alone.
- *Pyro-valve failure.* The particular batch of pyro-valves, used on MO, was noted by a different program to experience a particular failure mechanism (blowby). This failure mechanism is unique only to the redundant design (a single-string pyro valve could not fail this way).

Finally, the Clementine 1 failure was neither caused by unique deep-space requirements. Rather, it was precipitated by the confused set of project requirements which directed that the same set of software functionality was supposed to be executed on an previous generation (heritage) processor as well as on more advanced processor which the mission was trying to demonstrate. This dual-redundancy requirement overloaded the software team and required that an unproven software was running on the primary processor which was not even suitable for a mission of this complexity. This was a management failure (desire for more reliability than was warranted by the design) more than anything else.

The purpose of this section is not to engage in the polemics but rather to respectfully challenge a hypothesis that adopting hardware designs from existing programs is automatically somewhat harmful to deep-space missions. As discussed above, it turns out that there are only a few unique requirements of deep-space missions. Much greater danger to the mission success is to arbitrary and extensively change an existing design to make it somehow more suitable to the particular mission requirements (*better is enemy of good enough*). This trend is particularly strongly embedded in the deep-space spacecraft development because of their inherently customized designs.

Table 4. Adaptations of LEO-Spacecraft System for Low-Cost Deep-Space Mission

Function / subsystem	Typical LEO spacecraft	Typical deep-space spacecraft
Propulsion	Non-existent or minimal dv	Moderately stressing dv and mass fraction requirements
Attitude control	Magnetic + horizon sensors	Easy design for spinner Moderately expensive hardware for 3-axis design (primarily star tracker)
Communications	UHF / S-Band Typ. must provide both ends of link	S-Band / X-Band with complex interface to high-performance system (DSN) Long communications sessions
Navigation	GPS receiver	One-way Doppler (rely on DSN)
Electric power	Complex shadowing/eclipse considerations PPT often essential Complex battery charging	Simpler configuration Typ. simple battery trickle charging only PPT not required
Command and data	Typ. requires high thruput design	Less stressing design (lower data rates but more on-board data handling)
Software	Complex integration (often heritage code)	Comparable (must constraint growth in distinct mission modes)
Fault detection and management	Implemented	Similar (perhaps more streamlined version)
Thermal subsystem	Orbit cycles Beta plane changes	Typ. more static design
Structure / mass budget	Ordinary	Similar (but must manage tight mass margin)
Schedule	Sometimes flexible but often not (piggyback launch, politics, budget overrun)	Often very inflexible

Conclusions

The purpose of this paper was to review and analyze generic requirements for deep-space missions with modest objectives. The purpose was to demonstrate that the inherent cost premium of a deep-space mission is not high as commonly perceived. Indeed, the deep-space missions are more difficult than an ordinary LEO mission, but not significantly more so (with the exception of a few extremely difficult missions such as the Sun probe or Europa/Kuiper-Express class). The table 4 summarizes impact of deep-space mission requirements on a typical small spacecraft design.

This paper tried to show that there is no black magic to a deep-space spacecraft design. However, there is one key and absolutely essential difference in the development approach for a LEO-based spacecraft and its deep-space equivalent. Due to the increased role of the software and its inherent flexibility because of ease of on-orbit fixes, an unfortunate trend has developed to launch not-fully-completed and -tested spacecraft with the expectation that the full mission capability will be only achieved after some time after the launch, by continuing the system test, debug and fix cycle on orbit. It is a dangerous but, sadly, more and more accepted mode of operations.

This would be unacceptable for even a modest deep-space mission. Those spacecraft must be launched fully ready for the stressful mission operations. The first and second trajectory correction maneuvers happen within days after the launch and those events fully exercise the propulsion, attitude control and power subsystems. Also, the spacecraft is rapidly receding into deep space and there is only a short window to address any inadequacy in the communications subsystem maturity.

It remains to be seen how ultimately affordable these missions can become. However, it is becoming clear that simple missions (like comet or asteroid flyby, Mars entry, etc.) can be accomplished with modest resources so that the non-traditional funding sources (like public interest organizations, entertainment concerns, etc.) could be considered.

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