Architecture Synthesis and Reduced-Cost Architectures for Human Exploration Missions

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Development of architectures for human exploration missions has been pursued in the international aerospace community for a long time. This paper attempts a different approach and way of looking at architectures. Most of the emphasis is on lunar architectures with a brief look at Mars. The first step is to set forth overarching goals in order to understand origins of requirements. Then, principles and guidelines are developed for architecture formulation. It is argued that safety and cost are the primary factors. Alternative mission profiles are examined for adherence to the principles, and specific architectures formulated according to the guidelines. The guidelines themselves indicate preferred evolution paths from lunar to Mars architectures. Results of example calculations are given to illustrate the process, and an evolution path is recommended. Safety and cost criteria tend to conflict, but it is shown that cost-efficient architectures can be enhanced for good safety ratings at modest cost.

I. Note

During fiscal year 2004, NASA's Space Architect's Office explored the reaches, boundaries and constraints of exploration missions and their implementing architectures, and effects arising from policy options. Diverse opinions and suppositions were entertained, some of which included participation by the author as a consultant to the Space Transportation Team, both with respect to in-space propulsion and earth-to-orbit transportation. In keeping with the spirit of the deliberations to "stretch the enveloped" for insight, this paper was conceived, pursued, and is presented to offer personal views of the author in support of the new exploration vision.

II. Introduction and Purpose

Mission architectures for human exploration have been formulated since about 1952, when Wernher von Braun published concepts for a space station, and lunar and Mars missions. Von Braun applied the technology of storable propellant rocket propulsion to these missions to create his architectures. His was the usual method of architecture synthesis that begins with selection of a suite of technologies, followed by systems engineering processes of setting mission requirements and creating exploration systems concepts that satisfy the technology requirements and incorporate the technologies. The method is also usually performance-oriented since the requirements are usually stated in performance terms; "First I believe that this Nation should commit itself to achieving the goal, before this decade is out, of landing a man on the Moon..." - John F. Kennedy, May 24, 1961

In this paper I propose a different approach, beginning with the origin of requirements. Requirements come from national goals. It is essential to understand the goals behind the requirements; different or added requirements are likely to be found. This approach further attempts to build in desirable architecture attributes by using these attributes as a guide to architecture formulation rather than merely as evaluation criteria. This approach is the basis for several prior papers [1-3]

The real goal of Apollo was to accomplish a space spectacular challenging enough that the U.S. could do it before the Russians. At the time, the Russians had achieved several space spectaculars ahead of the U.S. and this was causing problems for the U.S. in maintaining international leadership. NASA is reported to have initially proposed a human flyby of the Moon, but Kennedy (probably on consultation with his advisors) said that was not challenging enough, and that NASA should land a man on the Moon.

Now we have a new Presidential initiative for human exploration. The requirements are taken to be to return to the Moon and send humans to Mars. When the President announced this initiative in January, he said two things that appear to be underlying goals: "Today I announce a new plan to explore space and extend a human presence across our solar system. ... we will undertake extended human missions to the moon as early as 2015, with the goal of living and working there for increasingly extended periods of time."

III. Goals and Requirements

These goals imply opening the frontier, not just sending occasional visitors. Meeting these goals imposes two often-overlooked requirements on the overall program: (1) develop self-sufficiency technology so that people can survive, work and prosper on the Moon and Mars. Unlike Earth, in these places air to breathe, water to drink, food to eat, and shelter all must be provided by technological means. An economically practical activity must rely on in-situ resources for these means to a very high degree. (2) Underpin the program with robust low-cost access to space so that we can afford more than sending occasional visitors. Today the price of sending payload to Earth orbit is about the price of gold, $10,000/kg. We need to reduce it at least to about the price of silver, a few hundred dollars per kg. Even at that price, there remains some doubt that economically viable lunar/interplanetary commerce and travel can occur.

Most prior studies of exploration architectures have been strictly performance-based. It has been assumed that the best architectures are those requiring the least mass launched to Earth orbit to accomplish a mission. I argue that only two evaluation criteria matter: Crew safety and cost. To avoid the trivial solution "don't go; completely safe and zero cost", we must include mission success as a necessary criterion. Giving these criteria top priority changes architecture formulation and selection.

IV. Methods

The usual method of formulating architectures is contrasted with the recommended method in Figure 1. The usual formulation is relatively unstructured, and may depend on analyst preferences for technology. The recommended approach does not introduce technology choices until architecture features are identified that enhance the selection criteria. These features become a guide to selection of technologies; technologies can be prioritized based on the features. Then we can, for example, select for safety and cost among those technologies capable of needed mission performance.

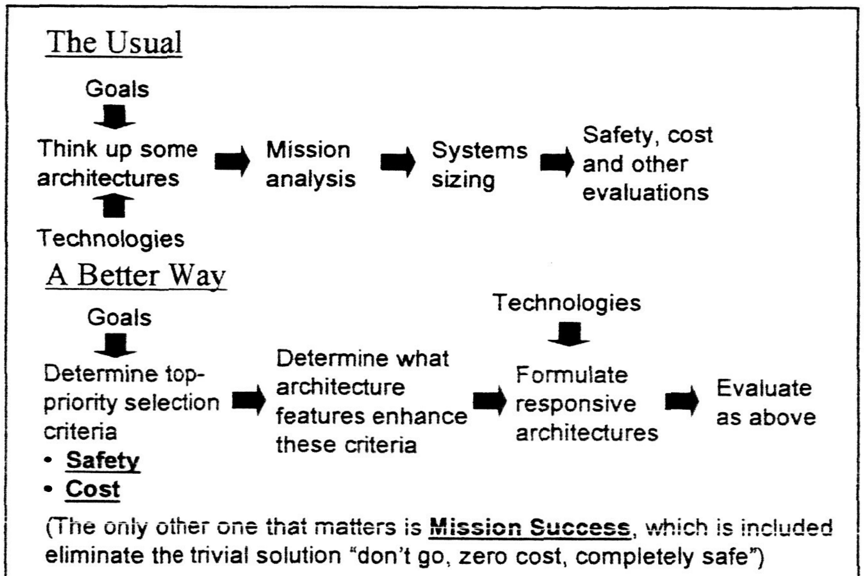

Figure 1: Architecture Formulation Methods

Safety – Features that enhance safety are:

- Redundancy of main propulsion and other subsystems
- Fewer safety-critical events for which failure leads to crew loss, e.g. rendezvous in lunar orbit or on the lunar surface
- Avoidance of hazardous subsystems, e.g. nuclear propulsion
    - Hazardous operation, e.g. proximity operations with nuclear propulsion must keep all crewed systems behind the reactor shadow shield;
    - Nuclear propulsion systems are not in-flight maintainable or repairable;
- Avoid highly stressed components or elements
- Provide aborts and safety workarounds; rescue capability
- Maximize flight and operational experience. Fly only mature subsystems and equipment on long-duration missions.

A significant misconception about flight safety is that common sense and safety engineering can produce systems that are safe beginning with the first flight. Experience does not support that view. We think of flying machines as safe because commercial jets have an excellent safety record. But the truth is:

- Flying machines are by their nature unsafe.
  - Hazardous speeds, altitudes, propellants, subsystems, events
- They can be made safe only by combining safety engineering and flight experience.
  - Safety engineering identifies and eliminates probable unsafe failures.
  - Flight experience discovers improbable accident modes not identified by safety engineering, and these are then corrected.
  - Actual loss probability is not really calculable except via experience.
- Aviation has had a century and billions of flights to figure out how to make it safe; aviation also has make-or-break economic incentives to be safe.

The aviation industry was not always so safe. Consider the early days of the jet age; the British Comet and the Electra: Both had structural failure modes that led to midair breakup. The Comet (fracture mechanics) was retired as a failure. The Electra (destructive dynamic coupling on propellor failure) was fixed, and flew for a long time as a military patrol aircraft. Later jetliners, beginning with 707 and DC-8, were always relatively safe. But even they had higher loss rates early on, until people learned what not to do. This big family of similar aircraft has probably logged over a billion flights. We still find failure modes, as in TWA 800 (center fuel tank explosion).

Space flight is relatively unsafe because it has relatively little flight experience. There is no practical evidence that we can design-in safety much better than Shuttle/Soyuz (1/50 - 1/100). There are an unknown number of (perhaps many) unidentified accident modes for any and all vehicles. Only shuttle, Soyuz, and Delta II have even a hundred flights. These three have reliability history about 0.97 - 0.98.

Low cost and safety are inextricably coupled. You can't have one without the other. Without low cost, one cannot afford to log enough flights to reach safety goals. Without safety, one cannot afford to fly often; (1) can't tolerate loss of life; (2) high cost of vehicle and payload losses; (3) high cost of accident investigations and fixes.

A practical approach to launch safety is to (1) apply safety engineering to (a) reduce potentially hazardous events, such as in-flight staging events and engine starts; (b) reduce potentially hazardous subsystems such as solid propellant rockets and explosive devices; (c) work towards fewer engines ... horizontal takeoff helps here; (d) provide an escape system for human-carrying flights and fly without crew unless crew is essential to the mission (e) provide redundancy of critical systems, including engine-out abort anywhere in trajectory ... horizontal takeoff helps here; redundant flight controls and actuators, electrical power and avionics; and survivable structure (in event of heat shield failure); after all these measures, then (2) aim for low per-flight marginal cost, ideally through a fully reusable system, and build flight experience to root out the low-probability failure modes that remain. Significant improvement in the initial loss probability may require hundreds of flights: for example, if the initial expected loss rate is 1/100 and this comes from 10 failure modes each with a probability of 1/1000 (obviously oversimplification), expectation is that in the first 100 or so launches, one of these will occur. When it is eliminated, the remaining failure modes result in an expected loss rate 1/111.

For in-space systems, especially those to be used for humans to Mars, the situation is difficult and requires special considerations:

- High flight rate is not in the cards. Therefore,
- Use subsystems with flight experience where possible, e.g.
  - RL-10 engines, about 500 in-flight uses.
  - SSMEs, about 400 in-flight uses
  - Solar electric propulsion, hundreds of uses for solar generation; electric propulsion has flight experience on comsats, and is beginning service for robotic space missions
  - Avionics systems with pedigrees
  - Human systems with shuttle and ISS experience
- Develop as much flight experience as practical through orderly evolutionary program, e.g. Moon before Mars; common subsystems
- Don't overstress hardware/systems; use adequate margins
- Use redundancy, backup, abort, safe havens, rescue to minimize severe consequences of failures
- Provide adequate response to environmental risks, especially zero g, radiation

Cost – Main considerations are:

- The main cost issue is $/kg since human missions involve lots of kg.

- The primary factor in determining launch cost is launch rate. Second is size (bigger is more expensive). Design configuration is secondary unless it is really bad. There are curves (actually families) for expendables and for reusables. The crossover is between 20 and 50 per year,depending on amortization treatment.
- Expendables:
  - Amortization of non-recurring cost
  - Efficient utilization of personnel and facilities ... mainly production cost of hardware, launch operations, etc.
  - Reliability, enhanced by higher launch rate; as noted in the safety discussion above.
- Reusables:
  - Amortization of non-recurring cost (much more important than for expendables)
  - Amortization of fleet acquisition cost; demands fast turnaround < 2 weeks.
  - Efficient utilization of personnel and facilities, mainly launch and flight operations.
  - Reliability, enhanced by higher launch rate; see safety discussion.
  - Since the value of the vehicle may be ~ 100X the target cost per flight, vehicle loss probability per flight must be < 0.001 to achieve desired economies.
- Payload size and mass capability ... if these are too small, division of large payloads into assembly packages introduces extra interfaces and complexity.
  - For lunar missions, the limit is about 5 meter fairing diameter and 20 t. mass. A 6-meter fairing and 25 – 30 t. would be better. Payoff for fairings and lift capability larger than this are minimal, and overshadowed by the added cost of larger launchers.
  - For Mars missions, the limit depends strongly on in-space transportation architecture. Some can live with the lunar limits cited above and some need much larger size and mass.
- Launch reliability ... the conventional logic is backwards.
  - The conventional argument is that mission success probability = r^N where r is launch reliability and N is number of launches; thus you want to minimize N. This is wrong-headed; see below
  - But smaller launchers are more reliable (launch rate factor). Reusables are expected more reliable than expendables.
  - What is actually important is (1) Minimize expected loss (higher flight rate, smaller launchers) and (2) Maximize architecture resilience/robustness re launch failures. This is also aided by more, smaller payloads
- Amortization: It is not a real cost to an ongoing program, it's a sunk cost.
  - Amortization is a real cost to the government and should be included in architecture decisions
  - The investment is made using borrowed money.
  - The cost shows up as interest on the national debt.
  - The government typically pays about 4% for capital, expressed in constant dollars.
- Most of what we believe about launch system investment trades is based on trying to stimulate industry to develop a reusable launch vehicle.
  - Commercial aerospace industry uses a "hurdle rate" cost of capital about 20% in evaluating new investments. Their actual cost of capital is about 10% to 15% depending on debt/equity ratio.
  - Commercial aerospace doesn't want to reduce launch cost; their analysis of commercial demand shows that it isn't very elastic, and reduced price means less revenue.
- If the government has enough demand to possibly justify reusable launch, the problem needs to be re-thought.
- For commercial cost of capital, the reusable/expendable break-even launch demand is 50 per year or more; for government cost of capital, less than 20.

V. Architecture Design Guidelines

These points lead to the following architecture design guidelines; brief discussion is offered for each:

- ***Strive for architectures that are practical with modest launch capability***
  This reduces launch cost, especially cost of new developments, and makes it possible to consider reusable launchers in cases where expected launch rate merits reuse. It also reduces expected launch losses, and makes coping with losses easier (less to replace). It appears practical to base lunar architectures on existing

EELVs. Some Mars architectures are compatible with modest launch capability (more on this later), but would require quite high launch rates on the order of one a week.

The conventional view is that larger vehicles significantly reduce cost per kg of payload. Historical trends for production cost of expendable launch vehicles support this view, but the slope is small and overshadowed by increased cost of R&D amortization, facilities and operations. The cost of production Cp follows a trend typically $m^{0.65} N^{0.85}$ where m is mass of hardware produced per unit and N is units per year production [4]. For vehicles of the size considered here, $R = m/P$ is essentially constant (for a given configuration type) where P is payload. $N = M_T/P$ where M_T is annual, or per-mission, delivery mass, also essentially constant. Then $Cp = m^{0.65} (M_T/P)^{0.85} = (RP)^{0.65} (M_T/P)^{0.85} = R^{0.65} P^{-0.2} M_T^{0.85}$. Doubling the payload decreases cost of launch vehicle production per mission about 13%, strongly overshadowed by the other factors noted.

A cost trade for lunar missions evaluated annual cost over a wide range of annual launch rate (expressed as equivalent Saturn V launches), for several launch options, as shown in Figure 2. The 40 t. vehicle was penalized with an 80% packaging factor and the 20 t. launch vehicle with a 60% packaging factor, to represent problems of efficiently dividing mission payloads into smaller elements. One can conclude from this chart that heavy lift never wins. The shuttle-derived launch vehicle (SDV) is competitive with existing ELVs above a launch rate of about 2 Saturn V per year, but in a "go as you pay" program, adopting an SDV would delay initial missions by several years in order to pay for the SDV development. Not reflected in this trade is the issue of whether continued use of the shuttle infrastructure could be done efficiently. There seems a risk that using this infrastructure would continue the current annual cost of about $3.5 billion, essentially constant whether flights occur or not. A prudent strategy is to initiate lunar missions with existing ELVs, develop a sound technology base for effective reusable launch vehicles (RLVs), and develop an RLV to serve the exploration program at a later date when launch rate makes the RLV an economic option.

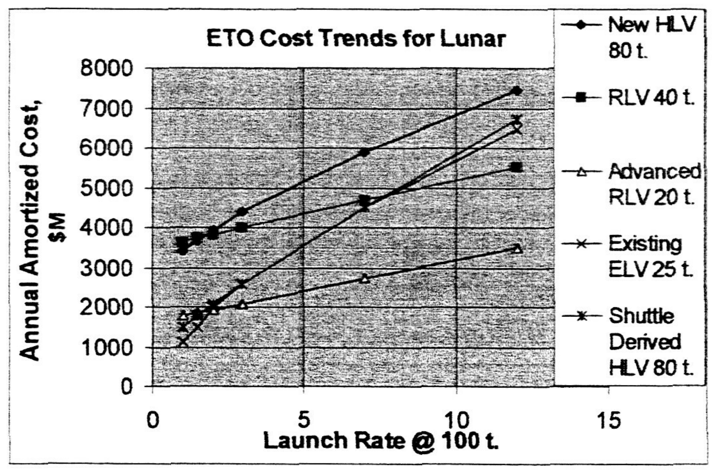

Figure 2: Launch Cost Trends

- ***Minimize expected launch loss (higher flight rate, smaller launchers); maximize architecture resilience/robustness with respect to launch failures.***
Smaller launchers will have higher reliability. The expected loss is N(1-r)C where N is the number of launches, r is reliability and C is cost of the launch (launcher plus payload). Since C is cM, where c is cost per unit mass and M is mass per launch, the expected loss is NcM(1-r). But $NM = M_T$ is the total mass to be placed in orbit, roughly a constant for each mission (for a particular architecture), so the expected loss is really just $M_Tc(1-r)$. Therefore, what we want is maximum r, and <u>number of launches does not matter.</u> Maximizing resilience requires a comprehensive overall relaunch, sparing and logistics approach. Spares not used for replacement can be used on future launches. The strategy will be different for lunar and Mars architectures because one lunar mission requires relatively few launches, and windows for launch to the Moon (even from an assembly orbit) open more or less weekly; Mars missions require more launches and the window only opens for about a month every 26 months.
- ***Use in-space subsystems with flight experience where possible. Develop as much flight experience as practical through an evolutionary program, e.g. Moon before Mars.***
Under a well-structured program, in-space hardware will tend to be less mature than launch vehicle hardware because the number of uses is fewer. It is important to gain flight and surface operations

experience on lunar missions and on the Moon where practical, to increase maturity. This can have a significant influence on lunar and Mars technologies selection.

- ***Minimize new developments and new technology insofar as consistent with other guidelines.***
  While there are many calls for a lot of new technology for the exploration initiative, history shows that development programs have difficulty digesting more than a few new technologies. This is true even when readiness level is brought to 6 before program start.
- ***Don't overstress hardware; systems; use adequate margins.***
  Technology performance projections for new programs and new technologies often reach for the outer limits of credibility, to make the program appear more attractive and less costly. The real effect, if such projections are baselined, is delays, more cost, and more operational risk, including risk to the lives of crews.
- ***Provide redundancy, backup, abort, safe havens, and rescue to minimize severe consequences of failures. Then, design efficient in-space architectures to minimize launch mass and mission cost, to the extent possible without compromising safety.***
  This is not in conflict with the previous recommendation. Efficient architectures should be defined with conservative technology performance.
- ***Provide adequate response to environmental risks, especially zero g, radiation.***
  Long periods of crew exposure to zero g are not likely on lunar missions. Crew operations on the Moon will be in 1/6 g. We presently have no knowledge of whether 1/6 g will be less degrading to human physiology than zero g, and will have to wait for operating experience on the Moon for periods of a month or more. Transits to and from Mars are expected to require several months. While crews have been aboard the Mir and ISS for periods this long, a crew landing on Mars will have no help from ground per-sonnel. Whether this will dictate an artificial-g spacecraft is still debated. Mars surface operations will be 3/8 g. Artificial g appears not practical on a planetary surface.

  Beyond low Earth orbit, solar proton events (SPEs; flares) and galactic cosmic rays (GCR) present significant risks to flight crews. Solar flare radiation is contained in a large (millions of km), fast-moving magnetic "bottle" ejected from the sun at the time of the flare. The magnetically confined radiation is isotropic (comes from all directions); shadow shielding will not protect from solar flares. Since the duration of a flare is only days, the usual design strategy provides a small shielded area for the crew, often called a "storm shelter", occupied during the high-radiation period. The crew need only stay in this volume for one or two days, and could exit for brief periods of a few minutes such as bathroom breaks. 25 g/cm^2 shielding for such an area would be about 0.5 to 1 t. mass for a crew of 4 to 6. Strategic placement of stores such as food and water can reduce the net mass penalty attributed to shielding.

  Galactic cosmic ray shielding for long-duration missions such as Mars transfers is more difficult, because the dose rate is continuous. A 25 g/cm^2 shield for a space station module (4.4 m diameter by about 10 m length) would have mass about 30 t, more than doubling the mass of the module. A possible strategy is to shield sleeping areas and a central work area, using to the extent possible stores of consumables (and of accumulat-ing human waste, suitably hermetically canned) to reduce the GCR dose by ½ to ⅔. There is controversy about how much GCR shielding is needed. My guess is that a GCR shield will be a sandwich, with a median atomic weight outer layer to fragment super-energetic heavy nuclei, a low atomic weight mid-layer to moderate and absorb neutrons, and a high atomic weight inner layer to attenuate Bremmstrahlung gamma rays.

  On lunar or Mars surface, lunar regolith can be used for shielding. Either body provides 2π (50%) shielding by its mass, and Mars' atmosphere provides about 20 g/cm^2. Mars does not provide magnetosphere shielding as Earth does. It is sensible to shield a habitat on lunar or Mars surface with regolith to save mass, but the time to emplace the shielding is likely to be at least several days.

- ***Design architectures to evolve to greater reuse if/when launch cost comes down.***
  This means mainly that architectures should be designed to accommodate modest launch payload capability. While it is not impossible to design reusable heavy lift systems (there are design concepts in the literature with payload up to hundreds of metric tons) it is likely to be impossibly expensive to develop one.

VI. Lunar Architectures

Lunar architecture concepts have been known since the time of Apollo; little in an overall sense has changed. There are about 20 practical architectures as summarized in Figure 3.

The direct profile diagrammed in Figure 4 is very simple. The mission uses one large heavy lift launch, flies to the Moon, lands, and after completing the surface mission, returns directly to Earth. The crew occupies the crew return vehicle (CRV) the entire mission, and has abort capability anywhere on the mission, except the return to Earth, already effectively an abort profile. There are no rendezvous events. Simplicity leads to a relatively high safety score. This profile is relatively inefficient, requiring a 150 t. (all cryogenic) to 175 t. (pressure-fed storable Earth return stage) launch capability. (Note: All chemical propulsion performance calculations in this paper are based on existing engine technology.)

Initial Profiles
1. Direct
1a. Direct with storable return
2. Direct refueled at L1 or LO
2a. Direct refueled at L1 or LO with storable return
3. LOR
3a. LOR refueled in LO by tanker (SEP or chem)
3b. LOR refueled in LO by LO depot
4. L1 rendezvous
4a. L1 rendezvous refueled at L1 by tanker (SEP or chem)
4b. L1 rendezvous refueled at L1 by L1 depot
Rescue capability applied to any of these as appropriate; designate as 4ar, etc.

Downstream performance enhancement
Lunar surface refueling (ISRU) for any of above profiles
Lunar surface refueling and supply of lunar propellant to L1 or LO depot
(For these, evaluate LOX only and LOX + LH_2)

Figure 3: Candidate Lunar Mission Architectures

One can improve the efficiency of the direct mode by refueling in the lunar vicinity, either in low lunar orbit or at a libration point, typically Earth-Moon L1. The most efficient is to refuel twice, once for lunar landing and again for return to Earth. It is safer to refuel only once, for lunar landing, and to always have sufficient propellant for return to Earth. This avoids a rendezvous and fuel transfer from being a safety-critical event. To improve efficiency, the refueling propellant can be delivered in advance by a solar-electric propulsion (SEP) stage with high specific impulse.[5]

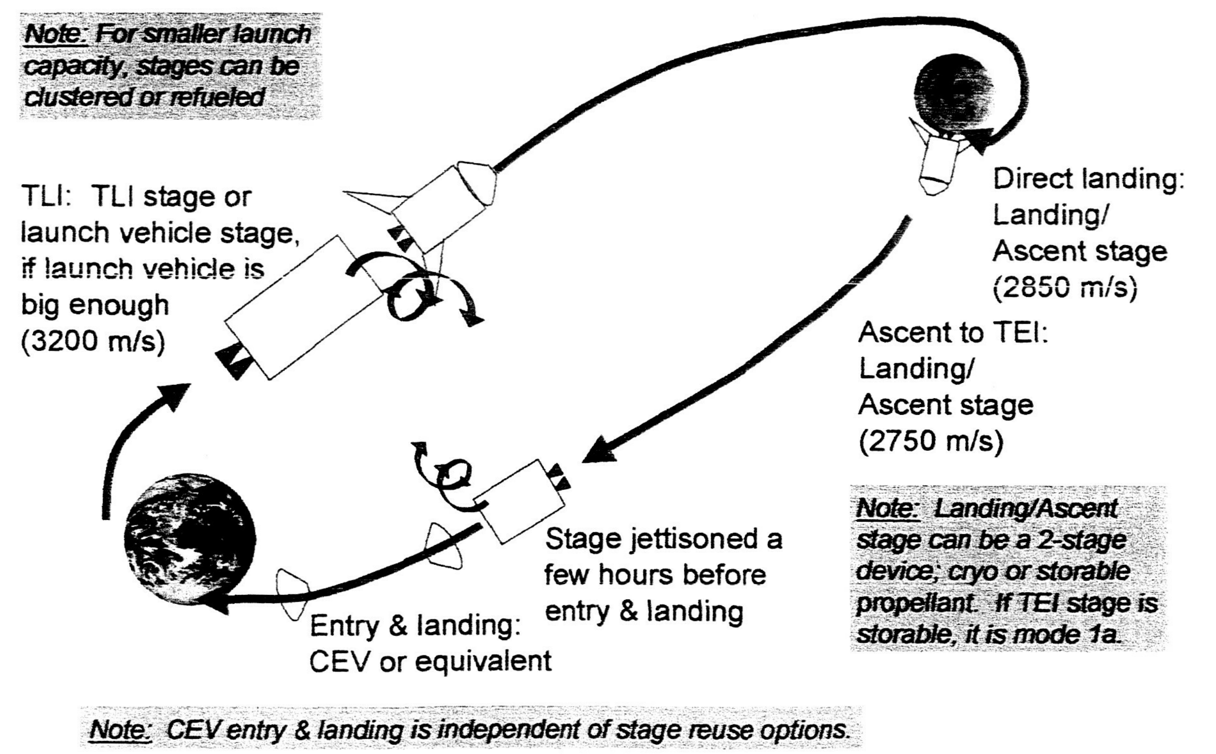

Figure 4: Single-Launch Direct Profile

These profiles show initial mass range from about 100 t. to about 110 t., the latter for the safer single-refueling case. At least two launches are used, and the launch capability can reasonably be as little as 35 – 40 t.

Lunar orbit rendezvous (LOR) was used by Apollo. It is the most efficient known lunar mission profile. Reasons for efficiency are that the lunar landing and ascent crew module can be very light compared to an Earth entry-capable crew module, and the propellant and propulsion for trans-Earth injection is not landed on the Moon, avoiding the propulsive energy cost of doing so. The LOR profile, to land 4 people and using cryogenic propellant, required about 110 t. launch to low Earth orbit, and can use launchers in the 25 – 30 t. class. Like the direct profile, the LOR profile can employ refueling by SEP. The most practical method is to deliver the lunar landing/ascent vehicle to lunar orbit by SEP; the crew vehicle makes rendezvous with this vehicle in lunar orbit. After the surface mission, the lunar landing/ascent vehicle returns to the main crew vehicle, the crew transfers, and returns to Earth. The lunar landing/ascent vehicle can remain in lunar orbit for subsequent use, but would require propellant transfer from a SEP-delivered tanker for its next surface excursion. Using SEP reduces launch mass to 60 – 70 t. The refueled LOR profile is diagrammed in Figure 5. Rendezvous after the surface mission is a critical maneuver and makes this profile less safe than the direct ones. This can be mostly corrected by changes described below.

L1 profiles are not discussed in this paper for reasons of brevity. They are very similar to LOR profiles insofar as safety and cost. There are operational reasons for selecting L1 rendezvous over LOR. While L1 rendezvous requires somewhat more delta V, use of a SEP for propellant delivery, or delivery of a LLAV to L1, nearly eliminates the greater delta V penalty in terms of launch mass requirement.

VI. Safety and Cost Comparisons

In this paper, in the interest of brevity, we present results only for selected profiles, to show a few significant trends.

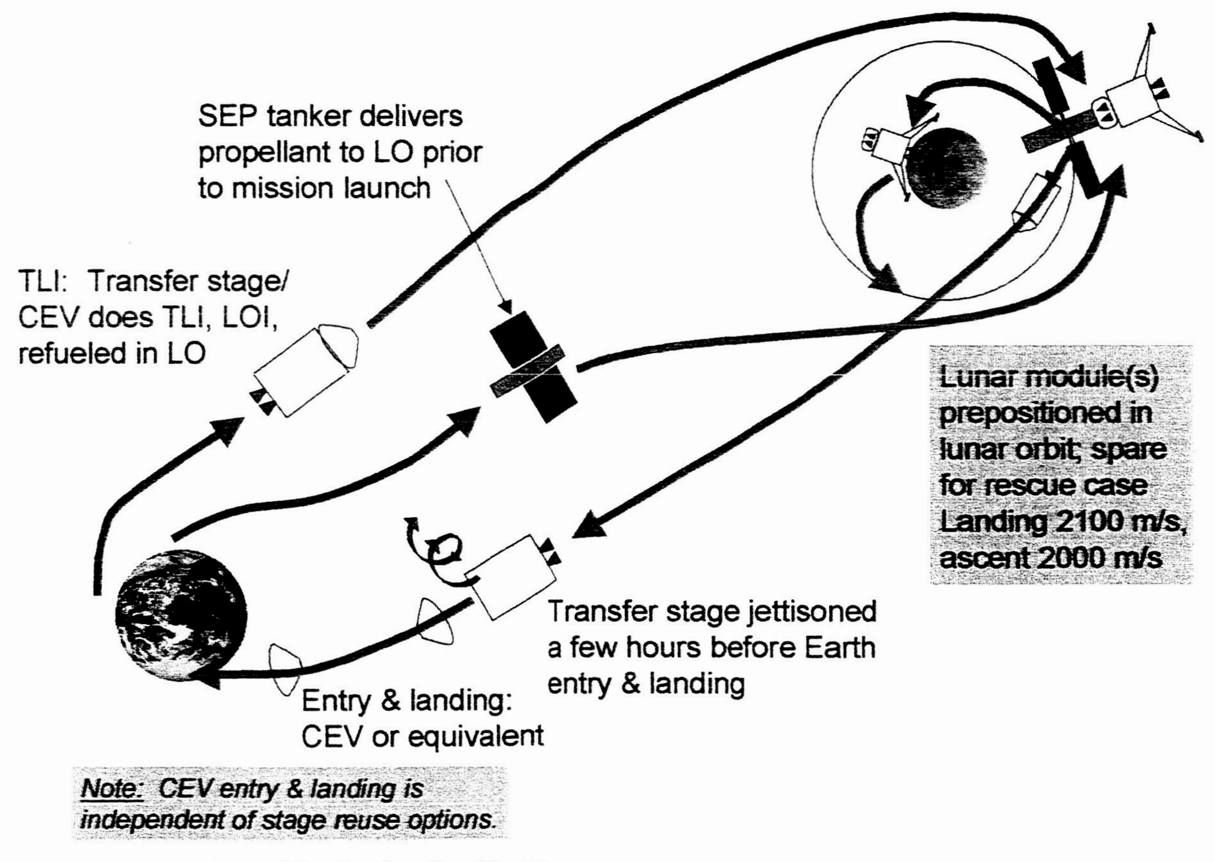

Figure 5: Refueled LOR Mission Profile Diagram

Safety comparisons were developed by a spread sheet with probability estimates for each mission event, and chains of probabilities from beginning to end of mission. Each outcome was designated as "mission success", "mission loss" (but crew safely recovered), or "crew loss". Crew loss is of course also a mission loss, but these categories separate a successful abort from crew loss. The sum of probabilities in the three categories is always 1 (i.e. there are no other outcomes), which provides a means of checking calculations. Failure probabilities for each event were judgmental estimates based on history. For example, Earth launch success probability was set at 0.97 based on the history of the most successful launch vehicles. Other judgments were more difficult, such as for success of entry, descent and landing of the Earth Crew Recovery Vehicle (ECRV); the U.S. has had no failures; the Russians have had one. This probability was set at 0.995. Mission success probabilities were based on one re-launch capability for failed launches (this means one spare payload unit for each launch). The values resulting from the chain-of-probabilities calculations are for comparisons of architectures and are not literal predictions of mission safety.

The chain-of-probabilities format enabled graphical presentation such as Figures 6 and 7.

The direct, storable return profile is a single-launch mission. A storable propellant stage is used for return to Earth from the lunar surface, and is available at any point in the mission for abort. The direct refueled profile is similar except that the propellant for lunar landing is delivered to lunar orbit by SEP. The storable return stage is also used for this profile and its propellant load is present throughout the mission, available for abort. Refueling reduces the total launch mass from 170 t. to 109 t. and the largest single launch to 86 t. Because the storable stage is available for abort, this increase in efficiency is obtained at no cost in crew loss capability. Mission success probability is reduced due to the added events.

The most efficient profile investigated was LOR with refueling in lunar orbit. The LLAV is presumed parked in lunar orbit, having been used on a prior mission, and is also refueled there. Propellant is delivered to lunar orbit by a SEP tanker. This mission has the least IMLEO of any profile examined at 60 t. but has a poor safety score because the crew, once inserted into lunar orbit, must successfully rendezvous and dock, and refuel, to get home.

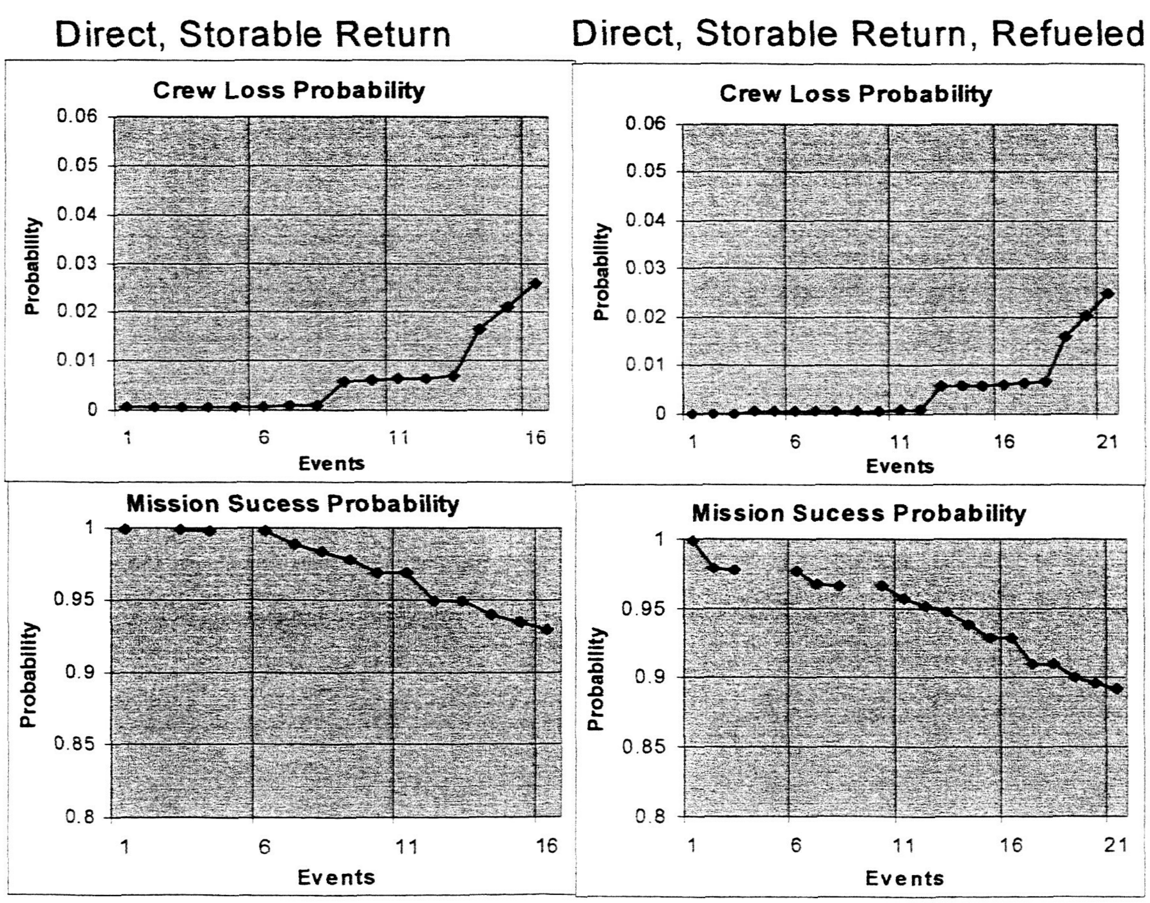

Figure 6: Success and Crew Loss Probabilities for Direct Profiles

Much better safety score was obtained by the following changes: (1) eliminate refueling of the lunar transfer mission; (2) provide a storable Earth return stage (for TEI from lunar orbit); (3) deliver a complete LLAV, or refuel an LLAV parked in lunar orbit, by SEP. The IMLEO increases to 77 t., and about 65 t. with LLAV refueling rather than delivery. A slight further improvement can be made by parking a spare LLAV in lunar orbit for rescue from the surface.

Representative SEP performance is given in Table 1. This performance level represents relatively advanced solar array technology at 350 W/sq m and 350 W/kg. Typical projections of array performance in the literature, in the time frame of these missions, is higher than these figures [6].

The LOR profile missions could be launched by existing expendable launch vehicles. The latter one would require separate launch of the TLI stage and the crew module with return stage. Analogous L1 rendezvous profiles can also be launched by existing expendable launch vehicles. The direct profiles require either multi-staging or fueling launches to low Earth orbit for compatibility with existing ELVs; as described here they presume heavy-lift launch capability.

Table 1: Representative SEP Parameters

Parameter	Value
Payload, kg	28,000
Power, KWe	500
Propulsive Efficiency	65%
Array Area sq m	1429
"UP" Propellant, kg	11,691
Retyrn Propellant, kg	3,098
Total Propellant*	15,528
SEP Inert Mass, kg	11,209
"Up" Trip, days	174
Return Trip, days	46
Thrust, N	23.7
Isp, sec	2,800

* Includes allowance for unusable propellant.

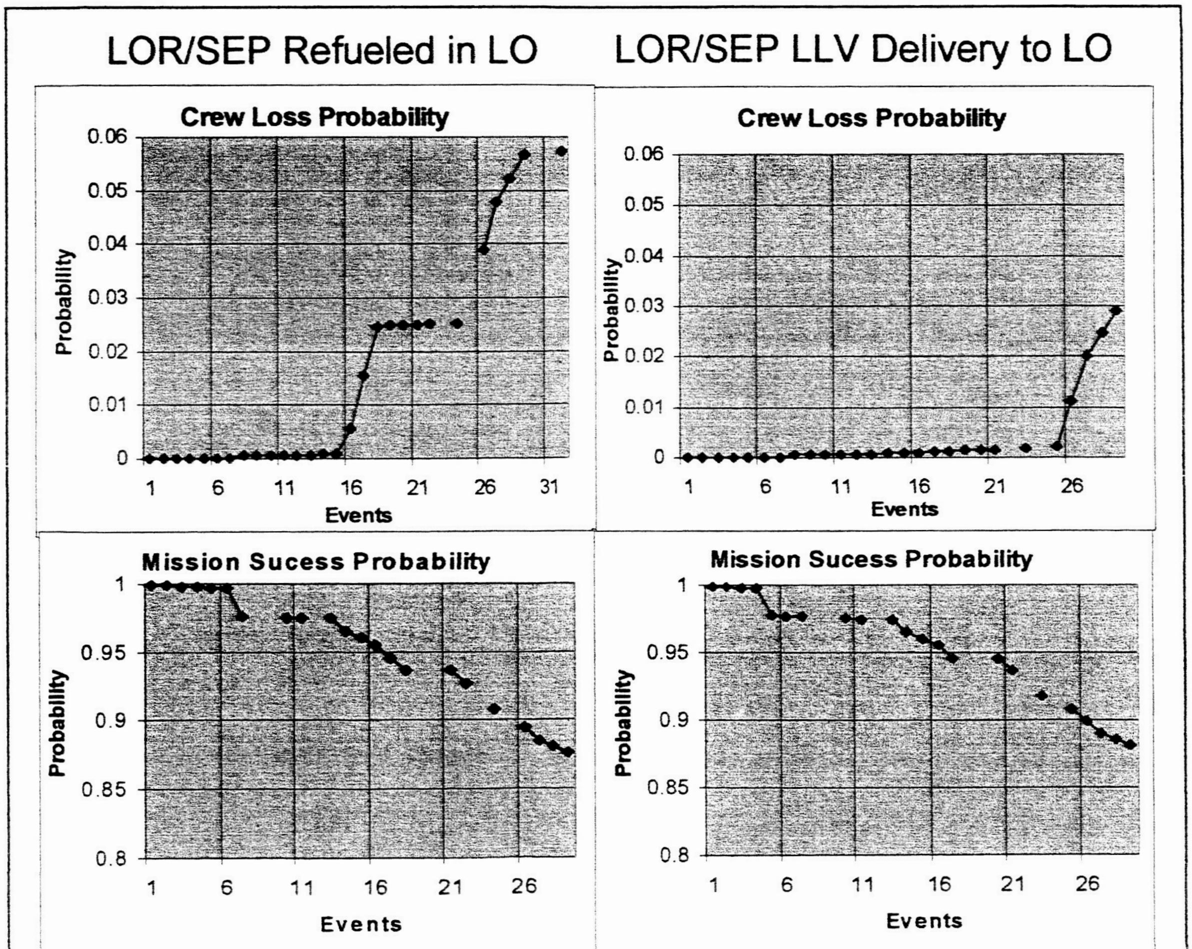

Figure 7: Mission Success and Crew Loss Probabilities for LOR Profiles

Cost ... The difference in recurring cost per mission between the most efficient and least efficient architectures is about a factor of 2, somewhat more than 2 for LOR refuel and about 2 for the safer LOR profile. The more efficient architectures also are more adaptable to high levels of reusability, as well as use of lunar-produced propellants.

VII. Lunar Architecture Trends

1. More efficient architectures are more complex.
2. These more complex architectures require careful tailoring to obtain high safety scores.
3. More efficient architectures are compatible with launch by existing ELVs.
4. SEP delivery of mission equipment and/or propellant to the lunar vicinity makes a significant contribution to reducing launch mass. This is amplified if the SEP is reusable. (Trends in solar array technology are making arrays more radiation-resistant, which is very important to practicality of reusing a SEP in a LEO-lunar vicinity transfer mode.)

VIII. Looking Ahead to Mars

An important part of lunar-Mars evolution is maturing hardware and operations for Mars. This means, especially for safety-critical systems, operating experience should be obtained on lunar missions where it is practical to do so. It is clear that for habitats, life support, rovers, crew modules, surface power systems, and chemical propulsion for ascent/descent, commonality and/or direct evolution can be practical. ISRU considerations make the preferred ascent/descent propellants oxygen-hydrogen for the Moon and very likely oxygen-methane for Mars, but one engine development can cover both options, since both propellant combinations are amenable to expander cycles. Fuel pump and injector modifications are probably needed to convert from one fuel to the other. I note that the RL-10 engine has been so modified to run on methane in ground tests.

Main propulsion for interplanetary transportation is the big (huge) issue. Chemical propulsion is strongly preferred for Earth-Moon transportation and is a poor contender for Earth-Mars transportation. The remaining options are nuclear electric, nuclear thermal, and solar electric propulsion.

The nuclear options, in view of high cost and political objections to extensive testing on the ground, will be under-tested, immature technologies if selected for Mars transportation. They could be used for lunar transportation to gain operating experience, but the same cost and political objections make this unlikely.

Solar electric propulsion technology is more advanced and mature than nuclear propulsion technology, and it will continue to be so since flight experience is being accumulated continuously. Solar array technology today is capable of power generation specific mass about 4 kg/kWe (250 W/kg), a performance level that nuclear power generation in space is unlikely to reach for decades. Solar array technology is predicted to reach 500 to 1000 W/kg in the next 10 to 15 years. REF

A simple comparative evaluation, presented in Figure 8, scores solar electric propulsion as a clear preference for interplanetary transportation. Two main issues exist: (1) Is it practical to build in-space solar electric power systems large enough (several megawatts) to provide human-capable electric propulsion to and from Mars, and (2) What propulsion performance can be obtained; in particular, is it possible to fly a short-stay mission with solar electric propulsion?

Solar electric generators are low-voltage DC systems. Most small solar generators in space operate at 28 volts. The large arrays on the ISS operate at 160 volts. At some point, higher voltage will lead to problems with arcing. Presuming a representative system power of 10 megawatts and a symmetric design with half the power on either side, at 160 volts the current from each side is 31,000 amps. Conductor masses would nearly double the array specific mass. One avenue that appears promising is on-array processing. The array is divided into segments of 20 to 50 kilowatts. Each segment consists of a number of solar cell strings at a moderate dc voltage. The strings are switched at a frequency on the order of 20 kHz and connected to 3 transformers, with switching phased so that the output of the transformers is 10 kHz three phase power at a high voltage such as 10 kV rms. At 10 kV the total rms current is 500 amps, 167 per phase. The rms current for one segment is a few amps. The required distribution conductor mass drops from over 50 kg/m to less than 0.01 kg/m. Insulation needed for the high voltage will probably result in a cable mass about 0.25 kg/m, still an improvement factor of at least 100. The magnetics required for the transformers is less than 1 kg/kWe. DC-AC conversion is required to condition power to run the thrusters, so this is merely a question of distributing conversion that is required in any case. The problem remaining is to synchronize all the segments as in a power grid. This, of course, has been done for about a century in terrestrial power grids but not at such a high frequency.

Criterion	**All-Cryo**	**Cryo-AC**	**NTP**	**NEP**	**SEP**	
Risky Events	2	5	4	3	1	
Operational Safety	2	3	4	5	1	
Redundancy	2	5	3	4	1	
Flight Experience	1	3	5	4	2	
Life	2	3	5	4	1	
Testability	1	4	5	3	2	
Devel. Cost	1	2	4	5	3	
Recur. Cost	5	2	3	4	1	
Reusability	5	5	4	2	1	
Performance	5	3	1	2	4, maybe 2	MS*
Adaptability	5	4	3	2	1	
	31	39	42	38	18	

*Mission Success

Figure 8: Simple Ranking Evaluation of Interplanetary Propulsion Systems (1 is best)

Future array performance has been projected as high as 1000 W/kg, and reasonable extrapolations of existing technology point to about 500 W/kg, corresponding to an alpha of 2 kg/kWe. Power processing and distribution will add another 2 kg/kWe, and thrusters (installed) about another 2. Nuclear generators at multi-megawatt power levels have been projected at about 5 to 7 kg/kWe, including an equivalent to the 3-phase step-up at 1 kg/kWe for the solar generator (this assumes a rotating machine nuclear generator; the rotating machine will produce 3-phase power at a suitable voltage and frequency). The comparison numbers are 3 kg/kWe for the solar generator and about 5 to 7 for the nuclear generator. The solar generator performance varies with distance from the Sun and is quoted at 1 AU. For a Mars mission, the solar generator must be discounted to account for operation near Mars at about 40% of the insolation at Earth, such that either type of generation is expected to fall in the general range of about 5 kg/kWe with the solar generator appearing to have a slight edge. Both projections are, of course, quite uncertain.

Rudimentary trajectory estimates indicate a 10-megawatt SEP with Isp 5000 and alpha 7 (including propulsion system) can perform a short-stay mission, but that the Earth return trajectory will be about a year rather than 8 to 10 months typical for a nuclear thermal rocket. Much more analaysis needs to be invested in this.

IX. Summary of Results

A method of formulating architectures that begins with goals and desirable attributes as well as requirements was presented. Examples of desirable attributes, and example lunar architectures were presented. The LOR/SEP and L1/SEP architectures were indicated as preferred. These architectures can be served by existing ELVs, avoiding up-front costs for developing a heavy lift launch vehicle and thereby advancing the date by which a return to the Moon can be achieved. Evolution to Mars missions was discussed, and SEP was described as a very rpomising avenue for use in lunar operations, and scale-up for use as interplanetary transportation for Mars.

The LOR/SEP lunar architecture was evaluated against the architecture design guidelines expressed earlier in this paper, as summarized in Table 2. Some of the guidelines are difficult to assess at the conceptual level, but this architecture appears reasonably responsive.

Table 2: LOR/SEP Architecture Evaluation Against Guidelines

Design Guideline	Evaluation	Remark
Strive for architectures that are practical with modest launch capability	Satisfied	
Minimize expected launch loss (higher flight rate, smaller launchers); maximize architecture resilience/robustness with respect to launch failures.	Considered, needs more evaluation of ETO payload modularity, sparing	
Use in-space subsystems with flight experience where possible. Develop as much flight experience as practical through an evolutionary program, e.g. Moon before Mars.	Satisfied	Lunar use of SEP is a plus
Minimize new developments and new technology insofar as consistent with other guidelines.	Appears satisfied; difficult to assess at this level	
Don't overstress hardware; systems; use adequate margins.	Satisfied in re assumed propulsion performance	
Provide redundancy, backup, abort, safe havens, and rescue to minimize severe consequences of failures. Then, design efficient in-space architectures to minimize launch mass and mission cost, to the extent possible without compromising safety.	Satisfied	SEP is inherently highly redundant and long-life
Provide adequate response to environmental risks, especially zero g, radiation.	Difficult to assess at this level.	Artificial g SEP concepts exist.
Design architectures to evolve to greater reuse if/when launch cost comes down.	Satisfied	

X. References

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