Human Missions Throughout the Outer Solar System: Requirements and Implementations

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Distance scales and mission times set the top-level engineering requirements for in situ *space exploration. To date, the implementation of various planetary gravity assists and long-term mission operations has made for a better cost-trade than technology development to decrease flight times. Similarly, crewed missions to date have not had mission time limits per se as drivers to implementation. However, unconstrained cruise times to the outer solar system are not acceptable for either robotic sample returns or human crews. Galactic cosmic ray fluxes likely provide a human limit for total mission times of ~5 years, and more restrictive limits may be driven by lack of gravity. We consider the implications for taking humans to the Neptune system and back, and, using this example, we deduce the minimum-cost path to realizing human exploration of the entire solar system by 2100.*

INTRODUCTION

The Vision for Space Exploration calls for a variety of objectives in support of the goal "to advance U.S. scientific, security, and economic interests through a robust space exploration program."[1] Although the current emphasis is on the time period leading up to the year 2020, fully carrying out all of the objectives in support of the goal actually will be a multi-decadal scientific and technical undertaking requiring the international participation that is part of the Vision for Space Exploration.

Scientific themes, questions, and priority science investigations for solar system science have recently been addressed in the report of the National Research Council known as the "Solar System Decadal Survey."[2] Although this report maps out a strategy for the 2003–2013 time frame, the process of obtaining final definitive answers to many of the questions will again be decades long, employing and requiring more and more advanced approaches, techniques, and technologies. The ongoing robotic program continues its evolution from reconnaissance (flyby missions), to exploration (orbiters, atmospheric probes, and landers), to intensive study (in situ laboratory/rover, sample return). Actually completing all of these modes of exploration for all solar system bodies remains challenging. The next step past intensive study involves human *in situ* exploration.

Fulfilling all of these stages of exploration and intensive scientific study will require the remainder of the current century.[3,4] From the point of view of both science and destinations, one can parse the solar system into five categories, namely (*i*) Primitive Bodies, (*ii*) Inner Solar System, (*iii*) Mars, (*iv*) Giant Planets, and (*v*) Large Satellites.[2] Primitive bodies (category *i*) occur throughout the solar system and are thousands in number. The inner solar system (category *ii*), namely the planets Venus and Mercury, provides significant thermal challenges, as well as other environmental challenges for Venus. Mars (category *iii*) represents a special case in many respects. For categories ii and iii, flight times and travel times, even to Mercury for sufficiently capable propulsion, are not significant issues because the maximum distance from Earth is, at most, 2.7 AU, the sum of the aphelia of Mars and Earth.

The final two categories of giant planets and their associated large satellites all span the distance from ~5 AU (Jupiter) to ~30 AU (Neptune). Many of the primitive bodies, including both the trans-Neptunian objects and Kuiper-Belt objects, which are some of the most primitive, also are found at large distances. Comparable to the large satellites in terms of size (Fig. 1) but isolated from giant planet systems, only Pluto is currently targeted for in situ scientific study, a reconnaissance flyby by New Horizons.[5,6]

We focus here on the intensive scientific study, and, in particular, human exploration of categories *iv* and *v* (i.e., the systems of the giant planets Jupiter and Saturn and of the ice giants Uranus and Neptune). In addition, we include the system of Pluto and its moons in this analysis, but not the larger objects that are far more distant [e.g., Sedna or Quaoar (88.3 and 43.3 AU from the Sun on 30 August 2007, respectively; minimum heliocentric distances of 76.1 AU on 5 April 2076 and 42.0 AU on 8 November 2068, respectively)].[7]

In what follows, we explicitly consider reaching the systems of the four large outer planets and of Pluto with 2-year, one-way trip times.

TIMETABLE AND MISSION DESIGN

Timetable

Previously, we introduced the ARchitecture for Going to the Outer solar SYstem (ARGOSY) approach and explored some of the implications. Here, we consider a more detailed treatment of the mission design for each of the planetary system targets. A rough timetable for human flights interspersed with sample-return missions was proposed in this initial treatment.[8] Table 1 shows these initial dates along with optimized launch dates for fast round-trip, crewed missions to each of the targets.

Table 1. Launch plan timetable.

Target system	Plan launch year	Optimized launch date
Jupiter	2050	28 July 2050
Saturn	2075	11 Dec 2075
Uranus	2085	20 Mar 2086
Neptune	2090	26 Sept 2090
Pluto	Arrival before 2110	13 June 2100

Mission Design

As a scoping exercise, we consider optimized trajectories to the five systems with nominal targets of Callisto (Jupiter system), Enceladus (Saturn system), Miranda (Uranus system), Triton (Neptune system), and Pluto, although an appropriate tour design for reaching these targets within each system (except that of Pluto) is not included (i.e., we focus on the outbound trip from Earth to each target system). We began with requirements of a flight time not to exceed 2 years and an overall power level of 100 MWe power for a nuclear electric propulsion system.[9] (This is to be distinguished from megawatts of thermal power provided by the power source.)

In each case, the starting condition from Earth was escape speed (i.e., C_3 = 0 km^2·s^{-2}),[10] the initial mass ratio[11] was taken as 4.9, the optimum for velocity changes in gravity-free space, and the trip time was set not to exceed 2 years. The general methodology was to minimize the trip time until it was <2 years (or close). This was done under the constraints of maximum power, initial mass, and final mass, as well as a "launch no earlier than" constraint. All other parameters were allowed to vary freely.

The results of this analysis are shown in Fig. 2. Several factors are worthy of note. Not surprisingly, the total ΔV (the total change in speed) required is greater than the simple average based on a zero acceleration time. With the constraints given, the acceleration time varies from 1.0 to 1.3 years, and the optimized specific impulse increases by a factor of ~10. Farther targets require more energetic propulsion as well as lower specific masses for the propulsion system. In each case, the results can be scaled as long as the thrust-to-(initial)-weight ratio is kept the same.

In Fig. 3, we show the corresponding trajectories. These polar plots are all scaled appropriately for the target

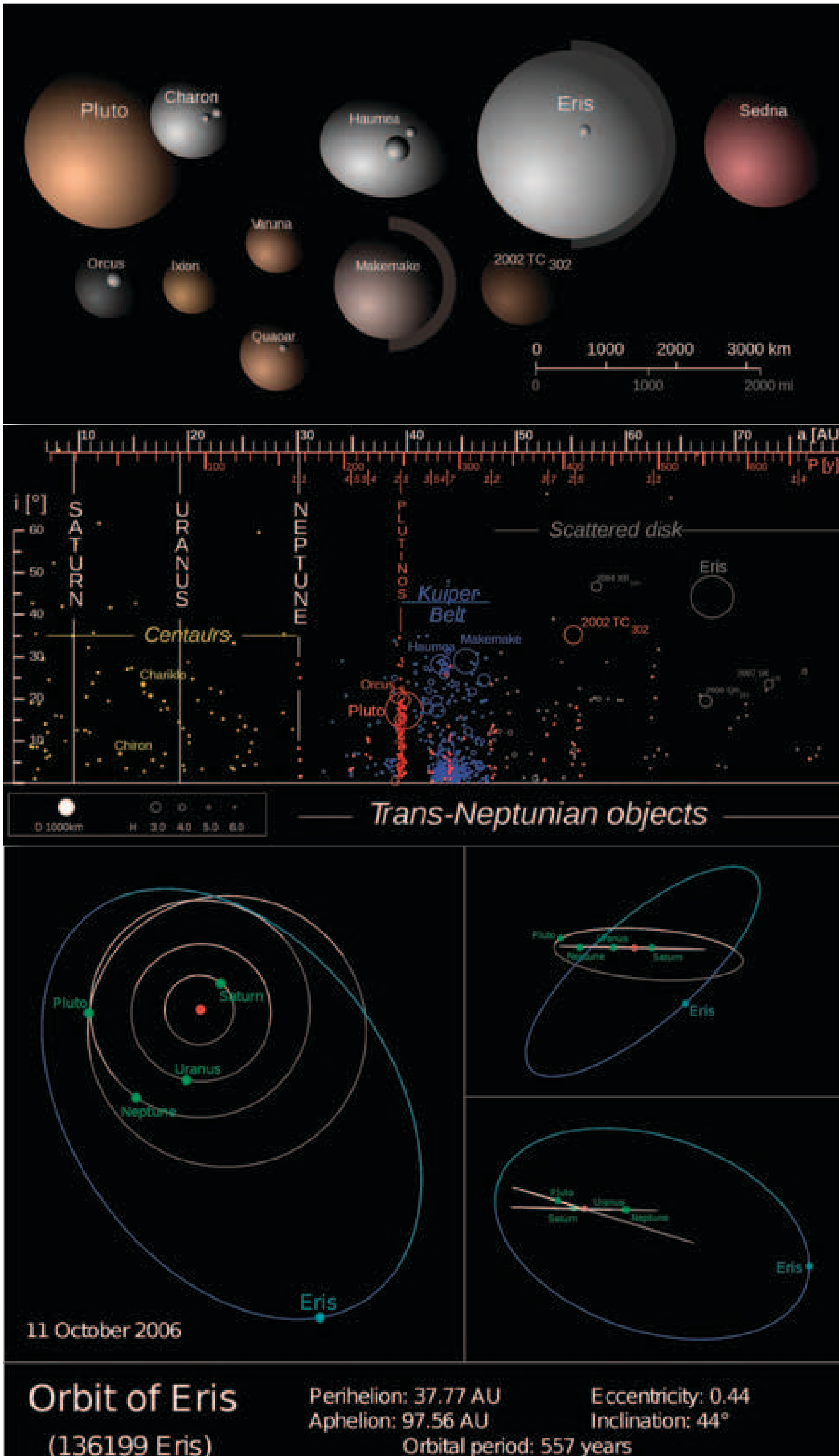

Figure 1. Trans-Neptunian and Kuiper-Belt objects. The relative sizes, colors, and albedos[79] (Top) as well as the locations and orbit parameters[80] (Middle) of the large trans-Neptunian objects. (Bottom) Also shown is the typical orbit of Eris (blue) compared to those of Pluto and the three outermost planets (white/grey).[81] Eris is now a "dwarf planet" along with Pluto and the closer Ceres.

	Jupiter	Saturn	Uranus	Neptune	Pluto
Launch Date	28 Jul 2050	11 Dec 2075	20 Mar 2086	26 Sep 2090	13 Jun 2100
Arrival Date	23 May 2052	1 Dec 2077	10 Jan 2088	29 Aug 2092	15 Jul 2102
Trip Time (yr)	1.8	2.0	1.8	1.9	2.1
Launch Mass (kg)	20,000,000	6,000,000	2,000,000	1,000,000	1,000,000
Propellant Mass (kg)	15,918,368	4,775,511	1,591,837	795,919	795,887
Final Mass (kg)	4,081,632	1,224,489	408,163	204,081	204,113
Power (MWe)	100	100	200	200	450
I_{SP} (s)	1792	3567	8642	12,675	19,559
EP System Efficiency (%)	80	80	80	80	80
Thrust (N)	9105	4574	3776	2575	3754
Thrust/Weight$_0$	4.64×10^{-5}	7.77×10^{-5}	1.93×10^{-4}	2.63×10^{-4}	3.83×10^{-4}
C_3	0.0	0.0	0.0	0.0	0.0
EP ΔV (km/s)	27.9	55.6	134.7	197.5	304.8
Thrust Time (yr)	1.0	1.2	1.1	1.2	1.3
Nominal Launch Year	2050	2075	2085	2090	2100
Nominal Target	**Callisto**	**Enceladus**	**Miranda**	**Triton**	**Pluto**
Heliocentric Distance (AU)	5.20	9.50	19.20	30.10	
Orbital Period (yr)	11.90	29.40	84.00	165.00	
2 × Distance/2 yr = Total ΔV Estimate (km/s)	24.60	45.00	91.00	142.60	
Error in Estimate (%)	11.92	19.05	32.43	27.81	
Figures of Merit:					
$[(gI_{SP})^2/\text{Thrust Time}]^{-1}$ (kg/kWe)	99.42	29.83	4.97	2.49	1.10
Final Mass/Power (kg/kWe)	40.82	12.24	2.04	1.02	0.45

Figure 2. Trajectory details for optimized flyout trajectories to the planetary systems in the outer solar system. The scalings of the two "figures of merit" are as expected for optimized trajectories in gravity-free space.

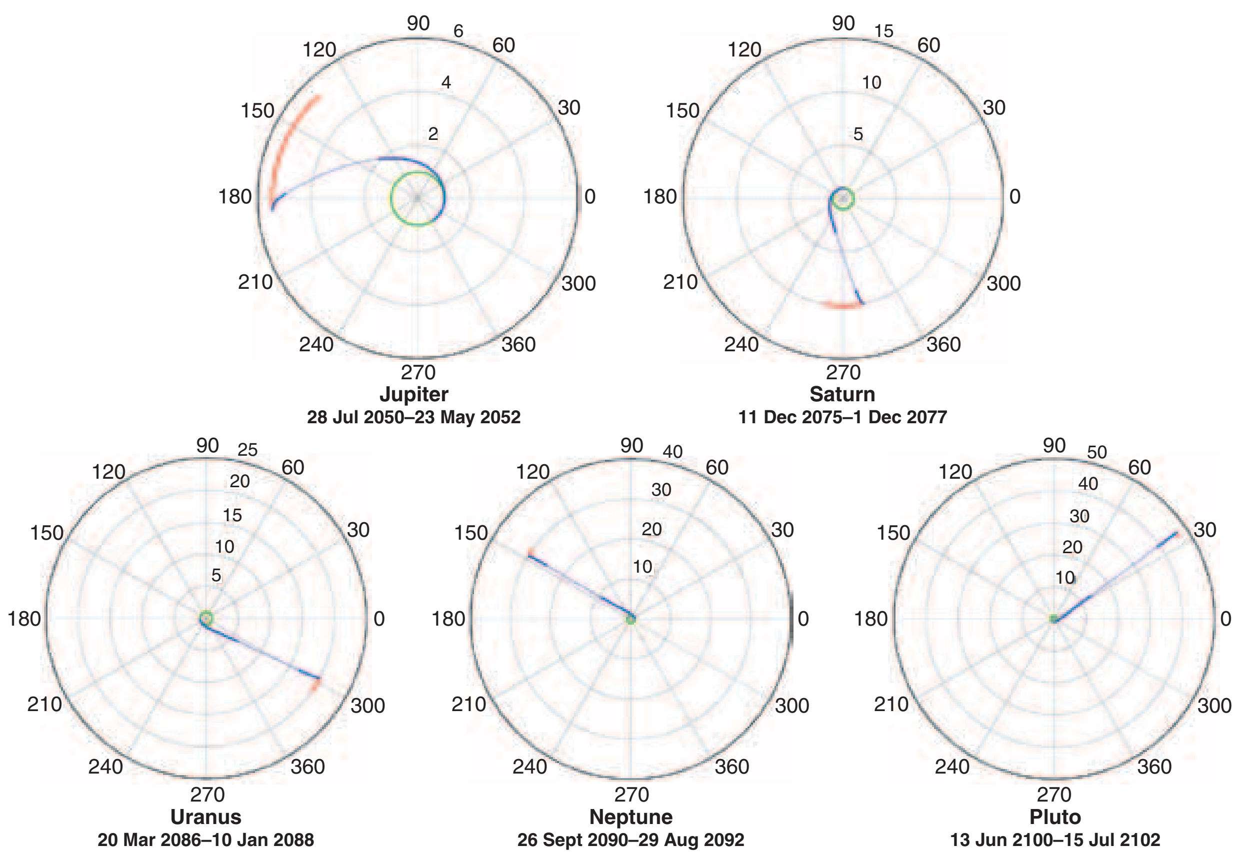

Figure 3. Outbound trajectories corresponding to the details of Fig. 2. Projections are into the plane of the ecliptic where 0° is the first point of Aries. Note the different radial scales and increasingly linear trajectories with larger heliocentric distance to the target.

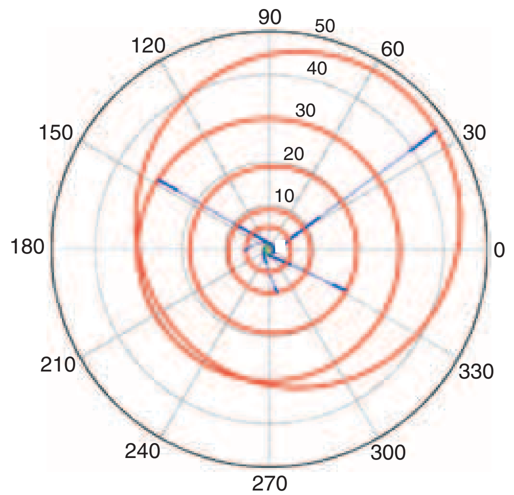

Figure 4. The various trajectories of Fig. 3 all given on the same radial scale.

planet in order to show trajectory details. In each case, the solid green line shows the Earth's orbit during the flyout (a solid line such as this takes more than 1 year), and the solid red line shows the motion of the target system during that time. The solid blue line shows the time during which the propulsion system is thrusting, first to accelerate and then to decelerate to the target. The dotted blue line shows coast periods. Figure 4 shows all of the trajectories to the same radial scale.

The striking point for all of these trajectories, and especially for the three outermost targets, is the lack of curvature. To date, planetary transfer trajectories make use of near-Hohmann-transfer orbits (minimum-energy solutions), albeit sometimes with intermediate planetary gravity assists. Propulsive maneuvers typically are used for gravitational capture at the target, rather than slowing down from faster-than-required transfer orbits. The "straight" trajectories are driven by the requirement of a fixed transit time; without the interplanetary deceleration period before reaching the target planet, the spacecraft in each case would escape from the solar system.

IN-SPACE PROPULSION

General

The key to distant targets in relatively short times is a low specific mass for the motive power of the transfer vehicle. For an efficient vehicle, we require the following:

$$\Delta v \approx gI_{sp} \approx \sqrt{\frac{\tau}{\alpha}} \ , \qquad (1)$$

where τ is the acceleration time to effect the speed change Δv, I_{sp} is the specific impulse (measured in seconds), g = 9.81 m·s^{-2}, and α is the specific mass of the propulsion system.[12, 13] The large specific impulses mean that chemical rockets will not suffice. The low values of α mean that a nuclear system of some sort is required.

Available Technology

It has been recognized that some type of magnetoplasma dynamic engine is required to operate at the tens to hundreds of MWe levels for crewed missions.[14] In the case of a crewed mission to the Neptune system, a round trip time of ~11 years was found for power levels of 10–100 MWe.

Typically, the systems studied with magnetoplasma dynamic thrusters have been in the 1–10 MWe range. For these, reliability and heat rejection have been the significant conceptual design issues (up to 10% of the input power can end up as heat that must be radiated away). Systems with up to 100 metric tons powered by nuclear electric propulsion systems of up to 200 MWe have been studied in a cursory fashion but have shown that at those levels, the required radiator mass begins to dominate, limiting further scaling of the designs and limiting roundtrip travel to Neptune and Pluto to 7–8 years or more.[15]

Parametric Case Study: Neptune

We have conducted a more detailed set of parametric trades for the case of a fast transfer to Neptune. The launch year of 2075 is sufficiently close to the 2090 date used above that the two cases are comparable (with an orbital period of 165 years, 15 years amounts only to a change of ~30° in Neptune's location in its orbit about the Sun).

The initial and final conditions are the same as before [i.e., at rest with respect to the locations of Earth at the beginning (C_3 = 0) and Neptune at the end just at the boundary of their spheres of influence]. An optimized trajectory is determined (for a constant thrust and constant mass flow rate) given a fixed mission duration of 730.5 days (i.e., 2 years). For this exercise, the propulsion system efficiency is taken as 100%, and the electric power level and launch date are determined so as to minimize the required propellant.

The problem is scaled in a way that yields results for 1 t of initial spacecraft mass. The power density is the optimum electric power needed per metric ton of initial mass. The optimum power density is evaluated as a function of jet exhaust speed over the range of 200–400 km·s^{-1} (20,394–40,789 s I_{sp}), which leads to a propellant ratio as a function of the exhaust speed. One may then apply specific values of specific mass of the power and propulsion system to the results to obtain the masses of the power and propulsion system per metric ton of initial mass. Subtracting the propellant and the power/propulsion system mass from the initial mass yields the net spacecraft mass per unit of initial mass.

Five values of specific mass were used: 10, $\sqrt{10}$, 1, $\sqrt{0.1}$, and 0.1 kg/kWe. The results show that the specific mass must be below ~1 kg/kWe in order to achieve a positive net mass. For a specific mass of 1 kg/kWe, the net mass passes through a maximum for a jet exhaust speed of ~220 km·s^{-1}. As the specific mass is reduced below 1, the optimum jet exhaust speed quickly increases such that the largest values of net mass occur at the top of the range considered, which is 400 km/s.

The 2075 launch window was chosen arbitrarily for the study. The launch date was optimized for each case. The dates range from 28 August 2075 for the 400 km·s^{-1} exhaust speed case to 17 August 2075 for the 200 km·s^{-1} case. The travel angles range from ~140° to 151°.

As a convenient example, we assume that the mass to be delivered is that of the completed International Space Station (ISS), namely 419.6 t.[16] Figure 5 shows the implied launch mass and propulsion system electric power for such a mission for the three lowest values of the assumed specific mass of the propulsion system, all as a function of the exhaust speed.

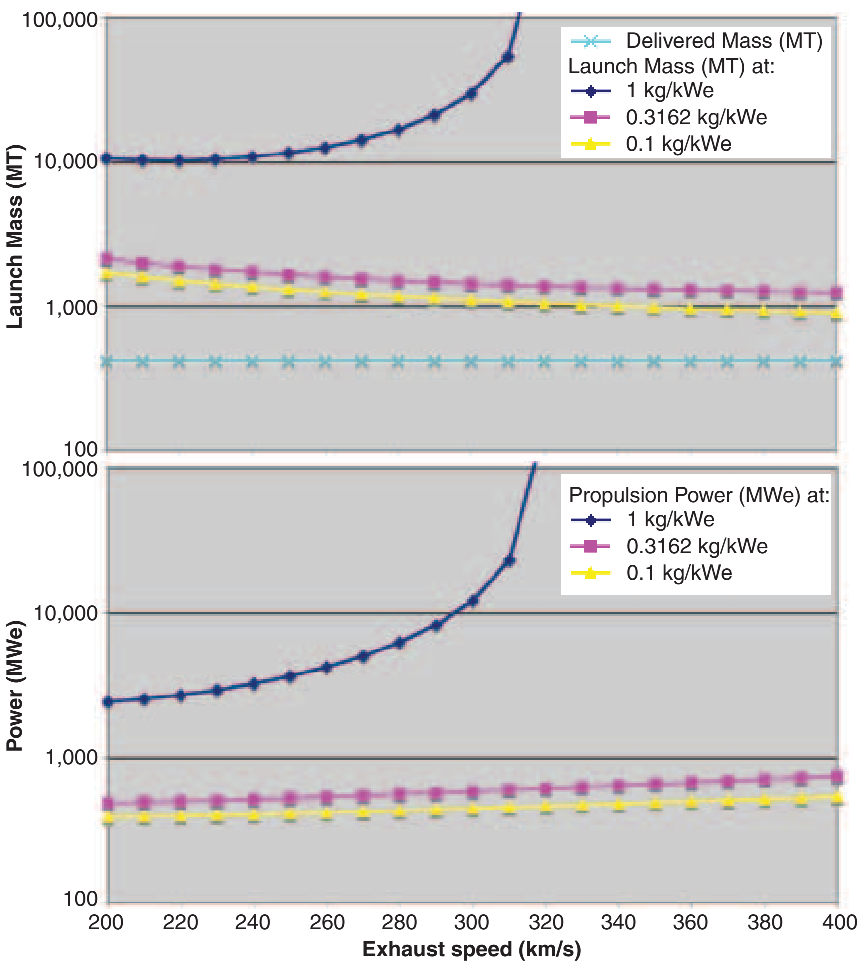

Figure 5. System requirements with a launch from $C_3 = 0$ for a 2-year optimized trip to Neptune (outbound only) as a function of propulsion system exhaust speed. Results are plotted for power system specific masses of 1, 0.3162, and 0.1 kg/kWe. The delivered mass at Neptune is assumed to be 419.6 t, excluding the power system mass and assuming that all propellant is used. The required launch mass (top) and the total (assumed electric) power (bottom) that must go into the exhaust beam are shown. The launch mass minima for the two lower specific power system masses occur for exhaust speeds >400 km/s.

RADIATION EXPOSURE

For prolonged deep-space travel, the two main radiation sources are solar energetic particles (SEPs) from the Sun and galactic cosmic rays (GCRs) from outside the heliosphere.[17] The two sources are anticorrelated with SEPs tending to occur more often near solar maximum. The GCR flux tends to minimize at solar maximum because the penetration of these particles into the heliosphere is impeded by the tangled interplanetary magnetic field lines associated with the intermixing of fast and slow solar wind.[18] The lower energies of SEPs make shielding possible, although challenging; higher-energy GCRs are more of a concern.[19]

GCR Variations

Longer-term variations of the GCR component also are associated with the longer-term variations in the solar cycle, notably those involving long minima in the overall number of sunspots. Pang and Yau[20] have considered variations in solar activity and their links with ^{14}C and ^{10}Be variations in various records going back as far as 3000 before the common era (BCE). These two radionuclides track each other very well and show significant maxima at the times of the various minima in the sunspot cycle (Wolf, ~1350 AD; Sporer, ~1500 AD; Maunder, ~1700 AD; and Dalton, ~1850 AD). Other maxima correspond to minima in ~1050 AD (Oort), ~910 AD, ~700 AD, ~500 AD, and ~300 AD. Another maximum in ~300 BCE has a comparable magnitude to that seen in the radionuclide data for the Maunder minimum period.

More recently, McCracken et al.[21] studied ^{10}Be data from the South Pole and the Greenland ice sheet to look at modulation of GCRs from 850 to 1958 AD. They found that the GCR flux reached similar maximum values during the Oort, Sporer, and latter part of the Maunder minimum, consistent with the results obtained by Pang and Yau.[20] They note that the low values of modulation found during the Sporer minimum may be consistent with the cosmic ray spectrum at Earth approaching that of the undisturbed interstellar medium.[22,23]

Although SEP events can potentially be more deadly, they also can be shielded against. The problem is the GCR flux,[24] which cannot be shielded against without unacceptable mass penalties: the shielding needs to have the equivalent depth near that of Earth's atmosphere.

The two questions are: *What is the "safe" level of exposure?* and *What is the exposure rate in space?* Braly and Heaton[25] examined the issue of exposure of unused film to radiation on Skylab and the amount of fogging that would occur in the film as well as the loss of transmissivity to a plate window of borosilicate glass. The solution was to provide a 4000-lb vault for film storage until use and a retractable cover for the window. However, the interesting point is the offhand remark about permissible radiation exposure levels with regard to the crew: 250 rem[26] to the skin and 25 rem to the blood-forming organs for a nominal 56-day stay. This level is for a 235-nautical-mile altitude, circular orbit, inclined 50° to the equator. The exposure is primarily attributable to the electrons of the Van Allen radiation belts, including those of the South Atlantic anomaly. The GCR background was calculated to be negligible at ~2×10^{-3} rad per day behind a 10 g/cm^2 Al shield (3.7 cm of Al).

The issue of radiation exposure for U.S. astronauts was raised again in the context of missions on the Russian *Mir* space station. In the NASA Aerospace Safety Advisory Panel (ASAP) Annual Report for 1998,[27] finding 14 reads: "In the ASAP Annual Report for 1997, the Panel expressed concern for high doses of radiation recorded by U.S. astronauts during extended Phase I missions in *Mir*. Subsequent and continuing review of this potential problem revalidates that unresolved concern. The current NASA limit for radiation exposure is 40 REM per year to the blood-forming organs, twice the limit for U.S. airline pilots and four times the limit for Navy nuclear operators (see also finding 23)."

This is followed by recommendation 14: "NASA should reduce the annual limit for radiation to the blood-forming organs by at least one half to not more than 20 REM."

The follow-up in finding 23 concerns exposure during extravehicular activity (EVA): "The greatest potential for overexposure of the crew to ionizing radiation exists during EVA operations. Furthermore, the magnitude of any overexposure cannot be predicted using current models." This is followed by the corresponding recommendation: "NASA should determine the most effective method of increasing EMU (extravehicular mobility unit) shielding without adversely affecting operability and then implement that shielding in the EMUs."

Both of these items come under the ISS program, for which the Panel notes (in part): "The governing principle universally accepted in the nuclear business, from weapons production to power generation to medical radiology, is 'As Low as Reasonably Achievable' (ALARA). To that end, the U.S. domestic airlines limit annual crew exposure to 20 REM, and the Naval Nuclear Propulsion Program limits crew and workers to 5 REM per year and no more than 3 REM per quarter. The ISS, on the other hand, allows an exposure of 40 REM per year."

Updated versions of problem exposures for deep-space missions are given by Cucinotta et al.,[28] who note:

> We show that the cancer risk uncertainty, which is defined as the ratio of the 95% confidence level (CL) to the point estimate, is about four-fold for lunar and Mars mission risk projections. For short-stay (<180 d) lunar missions, SPEs (solar proton events) present the most significant risk, but one that is mitigated effectively by shielding, especially for carbon composites structures with high hydrogen content. In contrast, for long-duration (>180 d) lunar or Mars missions, GCR risks may exceed radiation risk limits, with 95% CLs exceeding 10% fatal risk for males and females on a Mars mission. Shielding materials are marginally effective in reducing GCR cancer risks because of the penetrating nature of GCR and secondary radiation produced in tissue by relativistic particles. Currently, based on a significance test that accounts for radiobiology uncertainties in GCR risk projection, polyethylene or carbon composite shielding cannot be shown to significantly reduce risk compared to aluminum shielding.

They consider in detail cases of 600 deep-space days only for a Mars flyby mission and 1000 days for a Mars surface mission where 400 days of Mars surface days are added. Both SEPs and GCRs are included. The calculated doses for the two Mars flyby missions are summarized in Table 2.

Although polyethylene at 20 g·cm^{-2} shows some advantage over aluminum at the same areal density, there is no statistical significance in the probability distribution functions for uncertainties in projecting fatal cancer risk. There is such a difference for liquid hydrogen (LH_2), but its use as a shield introduces new system problems.

These results can be compared with those of over a decade earlier presented by Townsend et al.[29] using the HZETRN code. The results found included: (*i*) No shield

Table 2. Dose rates for deep-space missions to Mars.

Location	Dose rate, solar minimum (cSv per year)	Dose rate, solar maximum (cSv per year)
Deep space, 5 g·cm^{-2} Al	62.7	73.7
Deep space, 20 g·cm^{-2} Al	53.0	32.9

For units, see Ref. 26.

(119.2 cSv per year at 0-cm equivalent sphere) and (*ii*) 10.0 g/cm^2 (56.7 cSv per year at 0-cm equivalent sphere).

For long-term space missions, the issue is one of long-term, low-intensity doses and how the body deals with those as compared with acute doses, which is the basis of most of our current knowledge of effects. Divisions between subclinical exposure and clinical surveillance exposure (100 cSv) and clinical surveillance and effective therapy exposure (200 cSv) are well known for prompt exposures.[30] These numbers are, of course, significant for the design of "space storm shelters" for dealing with SEPs.

The 6-month average effective dose for the ISS is ~0.045 cSv per day; the 6-month average for Mars from the Mars Odyssey spacecraft is 0.13 cSv per day. Thus, the increase attributable to being outside of the magnetosphere is approximately a factor of three (personal communication, R. Maurer, APL, 5 Dec 2006). This number is basically in accord with the work cited above.[28] The real question is how such a chronic exposure (rather than the acute exposures associated with SEPs) affects humans over the long term. These various treatments average to a chronic dose level of 0.14 cSv per day. In considering missions to the outer solar system, GRC chronic dose will dominate SEP acute doses, and we can thus expect ~52 cSv per year. For a 5-year mission, the total accumulated chronic dose would be ~260 cSv (260 rem) and within the "therapy promising; guarded prognosis" range for an acute dose.[30] For comparison, ~400 cSv is the lethal dose for 50% of an acutely exposed population after 30 days.[31]

The shielding level of 20 g·cm^{-2} can be compared with the atmospheres of the terrestrial planets (Table 3).

The nominal structural mass of a vehicle is approximately equivalent to what one has at the surface of Mars. Triton's exosphere, not surprisingly, supplies no shielding at the surface. The thick atmosphere of Titan, even with its lunar-like weak gravity (approximately one-seventh that of Earth at the surface) shields that body much more effectively than the Earth's atmosphere shields the Earth's surface. The Earth's atmosphere and magnetic field together reduce the background dose to Earth's population to ~0.027 mrem per year from GCRs (of a total natural background of ~0.3 rem).

Initial studies on large space settlements[32] suggested that to limit exposure to ~0.25 rem per year from GCRs, a passive shield mass of ~400 g·cm^{-2} would be required. A limit of ~5 rem per year (as for radiation workers) would drop this by a factor of approximately three. As we will see from living-space estimates, this introduces a prohibitive shield mass.

SEPs and Apollo

The problem of radiation exposure in polar orbits close in to the Earth was discussed by Rust, who concentrated on the radiation hazard posed by energetic protons associated with some solar flare events[33]:

> During the Apollo program the radiation hazard was qualitatively the same (as during an ambitious space shuttle program), and NASA created a Solar Proton Alert Network to evaluate and warn of proton shower risks. Studies of the radiation hazards made in the 1960s for the Apollo program are still the most comprehensive available (at least at that time).[34–36] No way has yet been found to avoid proton showers, but so far astronauts have been at risk for only a few days each year. The risk from proton showers was small compared to the other risks that the astronauts faced. In the shuttle era, however, the relative importance of the hazard from proton showers will require reconsideration since there may be people in space almost continuously.

The salient point is that Shuttle crews launched into polar orbit from Vandenberg would be at far higher risk from radiation exposure than crews launched into low-inclination orbits from Cape Canaveral.

Rust continues: "I will describe the risks posed by major sudden solar flares, which start suddenly and can deliver in a few hours a disabling or even lethal dose of radiation. The largest doses can be two to three times the lethal level to a man. A lethal dose could be delivered over a period of several days, but at peak flux rates, a 1-hour exposure would cause nausea and possibly vomiting."

Rust also reproduces "suggested" exposure rates for Apollo astronauts, which are in the range of 50–100 rad per year (depending on tissue and location), hence 50–100 cGy. He notes that lengthy stays in geosynchronous or polar orbits entail risk that is "unacceptable because of flare protons." He continues: "It may be possible to design around hazards from trapped radiation, but better short-term forecasting of solar flares will be

Table 3. Areal atmospheric density at surfaces.[82]

Planet or moon	Surface pressure (mb)	Surface gravity (m·s^{-2})	Derived density (g·cm^{-2})
Venus	90,000	8.87	101,000
Earth	1,000	9.80	1,020
Mars	7–10	3.71	18.9
Titan	1,496	1.35	11,000
Triton	~1.5 × 10^{-5}	0.78	0.19

required to avoid aborting missions unnecessarily or exposing astronauts to a hazardous stream of protons."

Actual average doses of radiation to crew members have remained relatively low,[37] with the largest computed dose of 1.14 rad (1.14 cSv) (skin) occurring on Apollo 14. It is calculated that the solar proton event of 4–9 August 1972 would have caused a skin dose of 360 rad (360 cSv) to the astronauts if they had been in the command module at the time. (Apollo 17 flew 7–19 December 1972, and Apollo 16 flew 16–27 April 1972.)

Cold War Example

For purposes of considering underground shielding, it is interesting to compare needs for a Moon or Mars habitat with the launch control center for a Titan II missile.[38]

The missile center was a three-story domed cylinder (12.8 m tall and 11.3 m in diameter). This gives a rough volume of ~1280 m^3. The top of the dome was 2.44 m below ground level and 0.46 m thick (reinforced concrete). Actual launch complexes also included a blast lock and the silo with the Titan II under a protective 670-t steel and concrete door designed to withstand a 300 psi overpressure from a nearby nuclear burst.

Density of lunar regolith varies from ~1.1 to 1.8 $g{\cdot}cm^{-3}$, and the Viking landers inferred similar regolith densities on Mars.[39] At 1.5 $g{\cdot}cm^{-3}$, 2.44 m of material provides 366 $g{\cdot}cm^{-2}$ of mass areal density. Concrete has a density of ~2.3 $g{\cdot}cm^{-2}$, so 0.46 m provides another 106 $g{\cdot}cm^{-2}$ for a total of ~470 $g{\cdot}cm^{-2}$ of shielding (this depends on the use of rebar in the concrete and what type of concrete density could be made from the local regolith). Hence, the Titan complexes may be a fairly good model for what would be needed to deal with the radiation background on bodies with no atmosphere.

Active Shields

Active shields, usually powered by magnetic fields, have been discussed since the time of early manned spaceflight after R. H. Levy's 1961 suggestion to use superconducting coils to produce a magnetic shield.[40, 41]

With respect to material shields, Kash[40, 41] notes that lighter elements are more effective per unit mass in stopping charged particles than heavier ones. He suggests that polyethylene (CH_2) would thus offer certain advantages (it also is a good neutron moderator for the same reason). Kash notes that magnetic and electric shields offer advantages over material shields in keeping secondary radiation low by deflecting particles rather than by stopping them. However, synchrotron radiation is an issue and requires secondary material shielding at very high incident energies. Kash does not look at this aspect of the magnetic shield problem. He gives the example of the purported mass savings by comparing a magnetic shield with a material shield of polyethylene to stop 1 GeV protons (3.33 m, which is equivalent to 0.306 kg/cm^2 for a material density of 0.92 $g{\cdot}cm^{-3}$).

Such active shields typically have been examined for SEP protection due to the significantly higher energies of GCRs and the limitations of the shielding volume of realizable topologies.[42] Electric fields also have been discussed, but the need for fail-safe power supplies and associated technologies has tended to make passive shielding the approach of choice.[32]

Although magnetic shields offer a potential mass savings, the systems aspects must be thought through very carefully because one cannot afford for such an active system to go down at precisely the wrong time. The issue is analogous to long-term, deep-space storage of LH_2 for a nuclear thermal system or of other cryogens such as argon or krypton for a nuclear electric system. The refrigeration units cannot fail or else there will be a catastrophic loss of the mission.

When examining long-term, mission-critical systems, the choice tends to favor passive systems because of the mass, complexity, and inherent risks in providing fail-safe status to the "better-performing" system. As a result, magnetic systems have not been developed, and to do appropriate trades, the mass of the power system for startup and renewal after faults, and the ability to deal with non-superconducting transient faults, also must be considered. Aside from risk issues, the addition of such required systems might already render a magnetic shield system inferior on a mass-performance basis to a simple passive, material shield.

Living Space and Life Support

Mass requirements for long space voyages of months or more are driven by the needs of the human crew. In addition to the radiation issue, consumables (e.g., breathable air, water, and food) and living space must be considered as well as mitigation for microgravity.

Large Space "Habitat"

The problem is illustrated by the rough numbers for an "O'Neill space colony."[32] For a large (population ~10,000) space colony, both artificial gravity and radiation reduction to a "reasonable" level are required. For an exposure level of ~0.50 rem per year (0.5 cSv per year) passive shielding of ~450 $g{\cdot}cm^{-2}$ = 4.5 $t{\cdot}m^{-2}$ is required. The researchers found that rotating the shield at the same rate as the colony to provide some semblance of artificial gravity was not possible because the stresses would result in its structural failure.[32] This shielding number thus directly translates into an overall mass requirement, although factors of up to approximately two in uncertainty depend on the structural material (LH_2, polyethylene, aluminum).

In the same study, an assessment of living requirements suggested that an area of 67 m^2 per person and a volume of 1740 m^3 per person are appropriate averages over the entire colony. Oxygen, food, and water are needed in the amount of ~4.5 kg·day^{-1}·person^{-1}, a number that could be reduced to ~2.9 with 70% of the oxygen and water recycled.[8]

Skylab

The U.S. Skylab was a "space station" converted from the third stage of a Saturn V (the S-IVB stage). Lofted into orbit by the first two stages of a Saturn V, the station was designed for three uses of 28–94 days by a crew of three. The lab had a design life of 600 days and a mass of ~79 t with an electrical capability of ~11 kWe. With a habitable volume of 361 m^3, the volume was 120 m^3 per crew member.[43] This can be compared with the total habitable volume of 71.5 m^3 on the Space Shuttle[44, 45] with a crew of up to eight people for up to 16 days.[46]

International Space Station

For comparison, once the ISS has been completely assembled in 2010, it will have 953 m^3 of pressurized living space with an electric power-generation capability of 110 kWe and will have an overall mass in low-Earth orbit (LEO) of 419.6 t. The June 2006 configuration had 186 t enclosing a habitable volume of 425 m^3 for a crew of three (142 m^3 per person); typical supplies are 3.63 t to support the three-person crew for 6 months[16] or a rate of effective consumption of 6.7 kg·day^{-1}·person^{-1} (180 days).

Project Boreas

These numbers also can be compared with those of the Project Boreas concept.[47] For a 10-person crew operating on the surface at Mars' northern pole for 1200 days, a base infrastructure of 92 t delivering 61 kWe of power is estimated.[48] Modular like the ISS, the Project Boreas Pole Station calls for seven modules, including a greenhouse and facilities for *in situ* resource utilization (ISRU). Total habitable area is estimated as being 289 m^2 (96 m^2, crew quarters; 45 m^2, multi-use; 50 m^2, crop growth module; 45 m^2, science module; 45 m^2, EVA module; 8 m^2, cache).[49] For 10 people, this is just less than 30 m^2 per person, less than one-half of the space colony number and approximately the minimum recommended size for long-term habitation.[48] Assuming "headroom" of 3 m, the implied living volume is 867 m^3 or 86.7 m^3 per person for a 10-member crew (Fig. 6).

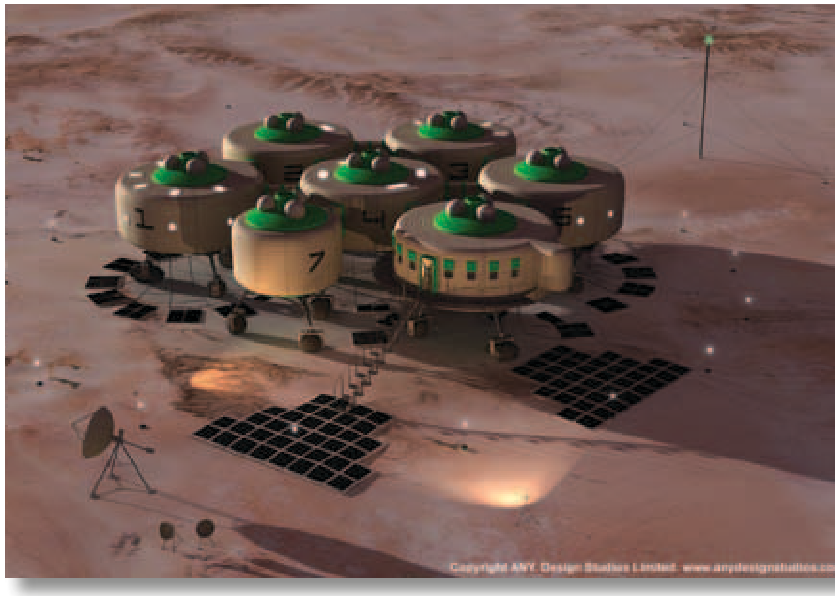

Figure 6. The Project Boreas Pole Station for Mars. This type of station could be used not only on Mars but for other outposts as well (e.g., on Callisto, Enceladus, Titan, Miranda, Triton, or Pluto). (Reproduced with permission from Mark Greene of ANY. Design Studios.)

For Boreas, the estimated need is 10.8 t of dehydrated food and 2 liters of water per day per person (2.9 kg of hydrated food per person per day).[50] Part of the breathable air, water for hygiene, and methane for a rover for the Martian pole is envisioned to come from ISRU—effectively "living off the land"—at a level of ~26 kg·day^{-1}, excluding additional shielding against radiation. With highly efficient recycling, life-support consumables are estimated as being 4.3 kg·day^{-1}·person^{-1}, rising to ~4.8 kg·day^{-1}·person^{-1}.[51] The Project Boreas study group estimates the total amount of water required from *in situ* resources to be ~1.6 kg·day^{-1}·person^{-1} with additional supplies needed for feedstock for a propellant plant to provide propellant for an exploration rover.

Derived Requirements

Expendables

Although deep-space voyages to the outer solar system will of necessity require extremely efficient recycling, adequate margins also must be supplied because adequate supplies are mission critical for a human crew. The "correct" number is somewhere between 3 and 7 kg·day^{-1}·person^{-1} based on various extrapolations and the ISS experience noted above. To be prudent and somewhat constrained, we adopt 5.5 kg·day^{-1}·person^{-1} for the required expendables. This translates into 2.0 t·year^{-1}·person^{-1} or 60 t for a crew of six for 5 years: 2 years in transit in each direction and 1 year maximum at the destination. For a crew of 10, the total increases to 100 t.

Living Space

For an eventual crew of six on the ISS with a total habitable volume of 953 m^3, the per-person volume is 159 m^3, approximately double that of

Boreas but less than one-tenth of that for a space colony. If we adopt a spherical geometry and 200 m^3 per crew member, then for a crew of six, the living space would be 6.6 m in radius; for a crew of 10, this would have to increase to 7.8 m in radius to maintain the same per capita allocation. This size is approximately twice that of the spaceship Discovery 1 in the movie *2001: A Space Odyssey*.[52]

Shielding

For these radii, the corresponding surface areas are 764 m^2 and 547 m^2, respectively. At 2.7 $g \cdot cm^{-3}$, a shielding of as much as 450 $g \cdot cm^{-2}$ implies an aluminum thickness of 1.67 m. The corresponding Al mass for the two geometries is 4240 t and 3145 t for the 10-crew and six-crew cases, respectively. Reducing the shielding to a near-minimum case of 20 $g \cdot cm^{-2}$ (0.074 m of Al) yields 154 t and 110 t, respectively. Hence, the difference in shielding mass for the conservative versus high exposure case amounts to a factor of ~27 in the mass of the shielding alone.

Table 4 summarizes the various examples cited and the derived requirements.

ECONOMICS

> *The real price of everything, what everything really costs to the man who wants to acquire it, is the toil and trouble of acquiring it.... It was not by gold or by silver, but by labour, that all the wealth of the world was originally purchased; and its value, to those who possess it, and who want to exchange it for some new productions, is precisely equal to the quantity of labour which it can enable them to purchase or command.*
>
> Adam Smith[53]

Such a program as envisioned here would not be a simple perturbation on a national—or international—budget. The cost would be sufficiently large, and the outlook for even potential profits so long-term, that the technical completion is only a small part of such exploration. In fact, the problem of how to implement human exploration of the entire solar system already exhibits hints of the economic solution to Fermi's paradox.[54]

Science Push: Antarctic Analogue

The most relevant analogy is that of the establishment of the Amundsen–Scott station at the South Pole in Antarctica.[55] The overall setting was the International Geophysical Year[56] but in the context of the cold war. Operation Deep Freeze I used 1800 volunteers and 20,000 measurement tons (~22,600 m^3) of cargo in the austral summer of 1955/1956 to establish Little America V and the staging base at McMurdo. During the following winter, 93 men stayed in the U.S. bases to prepare for setting up South Pole Station and Byrd Station, both in the interior of the continent, the following year under Deep Freeze II. That year, 3400 men, 12 ships, and multiple aircraft were used to establish the bases. After 64 aircraft sorties to the pole dropping 730 tons of equipment and supplies, 18 men spent the austral winter of 1957 at the pole. Before this time, only 10 humans had reached the South Pole: five who survived in the Amundsen expedition that reached the pole on 14 Dec 1911 and five who perished in the Scott expedition that reached the pole 17 Jan 1912.

The Scott expedition had cost £30,000, and Paul Siple, the first scientific commander of South Pole Station, later estimated that it had cost approxi-

Table 4. Living-space and life-support summary.

Outpost	Population	Mass (t)	Power	Outpost occupation time	Expendables per capita ($kg \cdot day^{-1}$)	Living space per capita (m^3)	Shielding
O'Neill colony	10,000	10.6×10^6[a]	Not given	Indefinite	4.5	1,740	450 $g \cdot cm^{-2}$
Skylab	3	79	11 kWe	600 days	Not available	120	Not available
Space Shuttle	8 (max.)	110[b]	21 kWe	16 days for each flight (max.)	Not available	9 (min.)	Not available
ISS (station complete)	6	419.6	110 kWe	Indefinite	6.7	159	Not available
Project Boreas	10	92	61 kWe	1200 days	4.8	87	Not available
ARGOSY	6	110[c]	Not calculated	5 years	5.5	200	20 $g \cdot cm^{-2}$ Al

[a]9.9×10^6 t is radiation shield mass derived from lunar regolith.
[b]Orbiter gross takeoff weight with payload to orbit (nominal).
[c]Radiation shielding only.

mately $1 million per man to put the first crew at the South Pole Station established by the International Geophysical Year.[55] Correcting for inflation, these amounts can be estimated as approximately £2,104,350[57] or $774,400 per person[58] and $7,160,000 per person,[59] respectively, in 2006 dollars. Hence, the cost for setting up a base was ~10 times that for a simple expedition.

Technology Pull: Post-Nova Studies

If you build it, [they] will come.

From the movie *Field of Dreams*[60]

Extremely heavy lift launch vehicles (EHLLVs, which we have dubbed "Supernovas") were studied in the early 1960s.[8] Detailed cost estimates were prepared for vehicles with payload capabilities of 0.5–2.0 million pounds (227–908 t) to LEO. The vehicles studied were mostly recoverable and fueled with liquid oxygen (LOX) and LH_2.[61] Most designs, such as the Rombus, were to be capable of ~10^6 lb. to LEO. The estimated cost of that vehicle was approximately $4 billion (1964 dollars) to develop and a projected $25 million per flight for multiple flights.[62] For comparison, actuals for the Saturn V are a development cost of $7.4396 billion (1966 dollars) and a per-launch price of $431 million (1967 dollars). Using the price deflators from the aerospace industry for defense goods and services (1968 = 22.9, 1987 = 85.62[63]; 1987 = 72.3, 2003 = 109.964) gives an inflation factor of 5.68. Using inflation rates of Department of Defense goods and services from 1968 to 2003, we can estimate an inflation factor of ~6.4 to 2007 (5.68 from 1968 to 2003 and 3% per annum thereafter). The Saturn V development cost would escalate to approximately $48 billion and the per-launch cost to approximately $2.7 billion. Using the square root scaling of Ref. 65 for initial investment cost, going from a Saturn V to a 1000 t to LEO capability (a factor of approximately eight) would increase the initial development cost by a factor of $8^{1/2}$ to approximately $136 billion.

For the Rombus vehicle numbers estimated by Wade[62], and using the gross domestic product (GDP) deflator index of ~1.127 from 1964 to 1968, the Rombus development cost would be approximately $29 billion and the per-launch cost approximately $180 million (a 2007 cost of $180 kg^{-1} for 1000 t to LEO, the type of costs that have been hoped for but never attained). The Saturn V comparables would be $3 billion per launch of ~125 t or $24,000 kg^{-1}, approximately a factor of two higher than the "canonical" $10,000 per kg.

Ehricke and D'Vincint[66] attempted to estimate the costs associated with a large post-Saturn vehicle, dubbed NEXUS, using LOX/LH_2 engines to take 10^6 lb. to LEO. A refined version of the analysis, summarized by Ehricke,[67] focuses on trades between orbital vehicle assembly mode and launch into orbit for immediate use, or direct flight mode and interorbital space vehicle for deep-space transport. The author notes: "This brief study shows that, because of the expense of orbital operations, reduction of orbital assembly operations and, hence, of the number of successful launches required, is an added important reason for developing large ELVs [Earth launch vehicles] and improving the economy of routine missions to the moon and beyond." Given our experience with the ISS (80 flights with five vehicle types required for station completion and maintenance during that time[16]), this concern has not been retired, especially considering the nominal 12-year assembly time (1998–2010). In the afterword of Ehricke's paper, Koelle[68] notes that this incorporation of labor (and infrastructure) costs into on-orbit assembly of large structures is a key point that had not previously been appreciated.

Deep-Space Markets

As with future considerations of the development of the Space Transportation System, the assumption behind the idea of EHLLVs was that the development of an efficient transportation system would enable a market. The history of the last 50 years has shown that economic advantages of space have accrued from Earth monitoring of weather systems for prediction of storms and of resource assay as well as for communications. Although mining of raw materials from asteroids has been discussed in terms of market value of platinum-group and other metals,[69] mining bulk materials from asteroids is almost as bad as transport of iron ore from Mars—either scenario makes "selling coal to Newcastle"[70] a bargain.

Other National Efforts

In terms of scale of effort, the development of the Panama Canal, a 34-year effort begun by the French and completed by the United States, is perhaps a good comparison. Here, the difference is that there were easily recognizable strategic and commercially profitable advantages. Nonetheless, by the time the canal opened in 1914, the effort had entailed an expenditure of approximately $639 million and required 80,000 workers, ~30,000 of whom died in the construction.[71] In 2007 dollars, this amounts to a total investment of approximately $10.7 billion. This can be compared with the approximately $27 billion for the Manhattan Project in 2007 dollars[72] and $92.8 billion for the Apollo program (inflated from $25.4 billion in 1973 by a factor of 3.655[73]).

Potential EHLLV Costs

In any event, it is difficult to imagine the development cost (FY2007$) of a 1000 t-to-LEO Supernova to be less than approximately \$50 billion and less than approximately \$1 billion per launch, even for a reusable booster (\$1000 per kg). It also is worth noting that both from a performance as well as from a pollution perspective, LOX/LH_2 will likely be the propellant combination of choice, drawing on the all-cryogenic launch vehicle pioneered by Boeing in the Delta-IV with the RS-68 engine. If Supernova vehicles are developed, larger engines will be required and will be the significant cost in the overall development effort.[74]

Rough Order-of-Magnitude Program Costs

Bringing down costs without an obvious deep-space market driver is a "chicken and egg" problem. Until transport and infrastructure costs decrease, it will be difficult to develop commercial deep-space markets for anything, but only the materialization of such markets can drive the transportation need to the volume required to bring the prices down. This is true for automobiles and airplanes as well as for spacecraft and associated transportation systems.[75]

For the types of systems envisioned in the calculations of Fig. 5, ~1700 t of mass is sufficient to cross from Earth to Neptune in ~2 years (assuming 0.1 kg/kWe and an exhaust speed of 200 km·s^{-1}). The mass of ~420 t is intended to carry food for a crew of six for 5 years, little more shielding than that provided by the structural mass of the ship, and a Boreas-style station that could be used as a science staging base on Triton and left there for any future expeditions.

If a previous robotic expedition was able to demonstrate that ISRU production of propellant for the return is feasible, then two launches of ~1000 t each to LEO would suffice to begin the expedition. If propellant production is not feasible, then as much as ~1500 t would need to be transported to the destination, requiring some 6000 t to begin with. For the five expeditions of Table 1, the implied LEO mass would be ~30,000 t or 30 EHLLV launches at 1000 t each with assembly required in orbit. We could optimistically envision approximately \$50 billion for the EHLLV development and \$30 billion for 30 launches.

For an Apollo program cost of approximately \$100 billion, and a guess that the program size scales as the distance and the cost as the square root of the program size, and discounting to 10% to account for a learning curve, then in going from the Moon (~0.3×10^{-2} AU) to Pluto near aphelion (~50 AU), we could guess that eventual runout cost might go as approximately $\$100B \times \sqrt{50/0.3} \approx \$1.3T$ plus an additional \$100 billion for the EHLLV development and costs. To provide an "upper limit," we could apply this same methodology to each of the targets of Table 1 (scaling by the orbital semi-major axis of each planet) in turn and derive a grand program cost estimate of:

$$\$100B \times \begin{pmatrix} \sqrt{\frac{5.2}{0.26}} + \sqrt{\frac{9.5}{0.26}} + \sqrt{\frac{19.2}{0.26}} \\ + \sqrt{\frac{30}{0.26}} + \sqrt{\frac{39}{0.26}} \end{pmatrix} \approx \$4T\,. \quad (2)$$

The GDP for the United States was approximately \$13 trillion in 2006; that of the world as a whole was approximately \$48 trillion, with the high-income nations accounting for approximately \$36 trillion.[76]

The current NASA budget is approximately \$16 billion, with approximately \$3 billion going to human exploration (the Exploration Missions System Directorate).[77] If this amount of funding were increased at the same rate as the real GDP increase (e-folding time of ~28 years[4]), then over the next 93 years, the amount of funds available would be approximately \$2.2 trillion or only approximately half of the estimated/guessed cost. This further strengthens the suggestion that such an effort would have to be truly international in scope. Put differently, \$4 trillion is approximately 1.5 times the U.S. cost of World War II in 2006 dollars.[59]

CONCLUSIONS

"Space," it says, "is big. Really big...."

Douglas Adams[78]

The deleterious effects of GCRs ultimately limit possibilities of the human exploration of the solar system. Unless huge (approximately factor of 10) mass penalties are paid, round-trip human voyages are likely limited to ~5 years even if artificial gravity can limit the health effects of microgravity. The technical limits are set by propulsion specific mass and specific impulse. Exploration across the solar system will require specific impulses in excess of 20,000 s at specific masses of no more than ~0.1 kg/kWe, in turn implying a system based on nuclear energy.

A 5-year round-trip mission will require ~10 t per person of expendable supplies with a likely crew of at least six people and an extremely reliable vehicle with an extremely dedicated and stable crew. Infrastructure capable of putting tens of thousands of metric tons of materials into LEO will be required as well. Such a project is potentially achievable at the cost of at least 10% of the current world GDP. With current investment in human space activity in the United States, even with growth projected on the basis of the growth of the overall U.S. economy, a dedicated, international effort will likely be required if the entire solar system is to have an initial reconnaissance by human crews by the beginning of the 22nd century.

Human missions beyond the asteroid belt to the outer portion of the solar system are literally a monumental undertaking. This challenge can be met, but only for a substantial cost. Robotic missions, including sample returns to venues as distant as Pluto at aphelion, although less capable, appear to be more easily and cheaply accomplished than missions with human crews. However, developing reliable, decades-long robotic missions is in itself a formidable technical challenge.

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REFERENCES AND NOTES

[1]NASA, *The Vision for Space Exploration*, NASA NP-2004-01-334-HQ, pp. 2–22 (2004).

[2]National Research Council, *New Frontiers in the Solar System: An Integrated Exploration Strategy*, National Academies Press, Washington, DC (2003).

[3]McNutt, R. L. Jr., "Space Exploration in the 21st Century," in *Proc. 10th Int. Workshop on Combustion and Propulsion: In-Space Propulsion*, Lerici, Italy (2006).

[4]McNutt, R. L. Jr., *Solar System Exploration: A Vision for the Next Hundred Years*, IAC-04-IAA.3.8.1.02, 55th International Astronautical Congress, Vancouver, Canada, http://www.lpi.usra.edu/opag/mcnutt_IAApaper.pdf (2004).

[5]Stern, S. A., "The New Horizons Pluto Kuiper Belt Mission: An Overview with Historical Context," *Space Sci. Rev.* **140**, 3–21 (2007).

[6]Fountain, G. H., Kusnierkiewicz, D. Y., Hersman, C. B., Herder, T. S., Coughlin, T. B., et al., "The New Horizons Spacecraft," *Space Sci. Rev.* **140**, 23–47 (2007).

[7]Jet Propulsion Laboratory, *Solar System Dynamics*, http://ssd.jpl.nasa.gov/ (accessed 4 Sept 2007).

[8]McNutt, R. L. Jr., "ARGOSY: ARchitecture for Going to the Outer Solar SYstem," *Johns Hopkins APL Tech. Dig.* **27**(3), 261–273 (2006).

[9]In a nuclear electric propulsion system, a nuclear-fission reactor, nominally using highly enriched uranium and using fast neutrons, produces heat, which is then converted to electricity. The electricity is then used to power some form of electric thruster, which provides quasi-continuous but low thrust at high I_{sp}. Hence, high-speed changes can be accomplished with relatively small amounts of propellant mass. Such systems have been analyzed for decades but have never flown.

[10]C_3 is the energy per mass of a spacecraft in excess of that required to reach escape speed from the Earth. It is typically expressed in units of km^2/s^2. Hence a C_3 of zero corresponds to a spacecraft just at escape speed, the speed required for the spacecraft speed to reach zero at infinite separation from the Earth if there were no other source of gravitational field.

[11]The mass ratio is the ratio of the total mass of a rocket or spacecraft to its empty mass (i.e., mass with no remaining propellant after the total speed change capability of the propulsion system has been exhausted). The rocket equation, also referred to as the Tsiolkovsky equation, is derived from Newton's second law of motion and shows that the total speed change that a rocket can accomplish in gravity-free space is equal to the rocket engine exhaust speed multiplied by the natural logarithm of the mass ratio.

[12]Shepherd, L. R., "Performance Criteria of Nuclear Space Propulsion Systems," *J. Br. Interplanet. Soc.* **52**, 328–335, (1999).

[13]Stuhlinger, E., *Ion Propulsion for Space Flight*, McGraw-Hill, New York (1964).

[14]Noca, M., and Polk, J. E., *Ion Thrusters and LFAs for Outer Planet Exploration*, Paper 228, AAF Sixth International Symposium on Propulsion for Space Transportation of the XXIst Century, Versailles, France, http://hdl.handle.net/2014/8607 (2002).

[15]Sercel, J., and Krauthamer, S., *Multimegawatt Nuclear Electric Propulsion—First Order System Design and Performance Evaluation*, AIAA-1986-1202, AIAA Space Systems Technology Conference, San Diego, CA (1986).

[16]NASA, *Reference Guide to the International Space Station*, NASA SP-2006-557 (2006).

[17]National Research Council, *Space Radiation Hazards and the Vision for Space Exploration: Report of a Workshop*, National Academies Press, Washington, DC (2006).

[18]National Research Council, *Exploration of the Outer Heliosphere and the Local Interstellar Medium: A Workshop Report*, National Academies Press, Washington, DC (2004).

[19]Parker, E. N., "Shielding Space Travelers," *Sci. Am.* **294**, 40–47 (2006).

[20]Pang, K. D., and Yau, K. K., "Ancient Observations Link Changes in Sun's Brightness and Earth's Climate," *EOS Trans. Am. Geophys. Union* **83**, 481 (2002).

[21]McCracken, K. G., McDonald, F. B., Beer, J., Raisbeck, G., and Riou, F., "A Phenomenological Study of the Long-Term Cosmic Ray Modulation, 850-1958 AD," *J. Geophys. Res.* **109**, doi: 10.1029/2004JA010685 (2004).

[22]McCracken, K. G., Beer, J., and McDonald, F. B., "Variations in the Cosmic Radiation, 1890-1986, and the Solar and Terrestrial Implications," *Adv. Space Res.* **34**, 397–406 (2004).

[23]Beer, J., Raisbeck, G. M., and Yiou, F., "Time Variations of 10Be and Solar Activity," in *The Sun in Time*, C. P. Sonnett, M. S. Giampapa, and M. S. Mathews (eds.) University of Arizona Press, Tucson, pp. 343–359 (1991).

[24]Wilson, J. W., Shinn, J. L., Tripathi, R. K., Singleterry, R. C., Clowdsley, M. S., et al., "Issues in Deep Space Radiation Protection," *Acta Astron.* **49**, 289–312 (2001).

[25]Braly, J. E., and Heaton, T. R., *Radiation Problems Associated with Skylab*, NASA 72N17702 (1972).

[26]"rem" (alt. "REM") stands for radiation equivalent in man and is a biologically effective amount of radiation dose. Recent attempts to provide canonical MKS (meter/kilogram/second) units have tended to complicate rather than simplify the literature. This is attributable, in part, to the fact that the main units [gray (Gy) and sievert (Sv)] are comparable to lethal radiation doses; hence, centigrays (cGy) and centisieverts (cSv) have come into usage. Thus, this supplants the "radiation adsorbed dose" or rad, which is equal to 100 erg deposited per gram or 0.01 Gy. The centigray and rad are thus equal units of radiation dose. The relative biological effectiveness of the absorbed dose depends on the type of radiation. The biological absorbed dose in sieverts (rem) equals the absorbed dose in grays (rad) multiplied by the relative biological effectiveness. The time during which the dosage is delivered is also an issue. An acute dose (radiation delivered more rapidly than cells can begin to respond to the damage) of ~400 rem (4 Sv) has been cited as lethal to 50% of the exposed population in >30 and <60 days. For consistency's sake and for ease of comparison with historical data, we have used a combination of centigrays and centisieverts as well as rad and rem because the dose amounts are equal in these units. Average background radiation exposure in the United States is ~0.3 rem = 0.3 cSv per year.

[27]NASA Aerospace Safety Advisory Panel, *Annual Report for 1998*, NASA, Washington, DC (1999).

[28]Cucinotta, F. A., Kim, M.-H. Y., and Ren, L., *Managing Lunar and Mars Mission Radiation Risks. Part I: Cancer Risks, Uncertainties, and Shielding Effectiveness*, NASA/TP-2005-213164 (2005).

[29]Townsend, L. W., Cucinotta, F. A., and Wilson, J. W. "Interplanetary Crew Exposure Estimates for Galactic Cosmic Rays," *Radiat. Res.* **129**, 48–52 (1992).

[30]Glasstone, S., and Dolan, P. J., *The Effects of Nuclear Weapons*, 3rd Ed., U.S. Department of Defense and Energy Research and Development Administration, Washington, DC (1977).

[31]Cember, H., *Introduction to Health Physics*, Vol. 105, 1st Ed., Pergamon, New York (1969).

[32]Johnson, R. D., and Holbrow, C., *Space Settlements: A Design Study*, NASA SP-413 (1977).

[33]Rust, D. M., "Solar Flares, Proton Showers, and the Space Shuttle," *Science* **216**, 939–946 (1982).

[34]Donnelly, R. F., *Solar-Terrestrial Predictions Proceedings*, Government Printing Office, Washington, DC (1979).

[35]Reetz, A. Jr. (ed.), *Second Symposium on Protection Against Radiations in Space*, NASA SP-71 (1965).

[36]Warman, E. A. (ed.), *Proceedings of the National Symposium on Natural and Manmade Radiation in Space*, Technical Memorandum TMX-2440, NASA, Washington, DC (1972).

[37]Bailey, J. V., "Radiation Protection and Instrumentation," Section 2, Chap. 3, in *Biomedical Results of Apollo*, NASA SP-368 (2002).

[38]Stumpf, D. K., *Titan II: A History of a Cold War Missile Program*, University of Arkansas Press, Fayetteville, AR (2000).

[39]Moore, H. J., Hutton, R. E., Scott, R. F., Spitzer, C. R., and Shorthill, R. W., "Surface Materials of the Viking Landing Sites," *J. Geophys. Res.* **82**, 4497–4523 (1977).

[40]Kash, S. W., "Minimum Structural Mass for a Magnetic Radiation Shield," *AIAA J.* **1**, 1439–1441 (1963).

[41]Kash, S. W., "Magnetic Space Shields," in *Advances in Plasma Dynamics*, T. P. Anderson and R. W. Springer (eds.), Northwestern University Press, Evanston, IL, pp. 135–166 (1964).

[42]Shepherd, S. G., and Kress, B. T., "Comment on 'Applications for Deployed High Temperature Superconducting Coils in Spacecraft Engineering: A Review and Analysis' by J. C. Cocks et al.," *J. Br. Interplanet. Soc.* **60**, 129–132, 2007.

[43]Wade, M., "Skylab," *E. Astronautica*, http://www.astronautix.com/project/skylab.htm (accessed 4 Sept 2007).

[44]Wade, M., "Challenger," *E. Astronautica*, http://www.astronautix.com/craft/chaenger.htm (accessed 4 Sept 2007).

[45]Wade, M., "Atlantis," *E. Astronautica*, http://www.astronautix.com/craft/atlantis.htm (accessed 4 Sept 2007).

[46]Wade, M., "Columbia," *E. Astronautica*, http://www.astronautix.com/craft/columbia.htm (accessed 4 Sept 2007).

[47]Cockrell, C. S. (ed.), *Project Boreas: A Station for the Martian Geographic North Pole*, The British Interplanetary Society, London (2006).

[48]Parkinson, B., and Wright, P., "Systems Modeling and Systems Trade for Pole Station," in *Project Boreas: A Station for the Martian Geographic North Pole*, C. S. Cockrell (ed.), The British Interplanetary Society, London, pp. 24–31 (2006).

[49]Greene, M., "Base Design for Pole Station," in *Project Boreas: A Station for the Martian Geographic North Pole*, C. S. Cockrell (ed.), The British Interplanetary Society, London, pp. 32–48 (2006).

[50]Maret, W. "Considerations for Optimal and Functional Nutrition at Pole Station," in *Project Boreas: A Station for the Martian Geographic North Pole*, C. S. Cockrell (ed.), The British Interplanetary Society, London, pp. 172–176 (2006).

[51]Parkinson, B., Cockrell, C. S., and Clarke, J. D. A., "In-Situ Resource Use (ISRU) at Pole Station," in *Project Boreas: A Station for the Martian Geographic North Pole*, C. S. Cockrell (ed.), The British Interplanetary Society, London, pp. 57–61 (2006).

[52]Ordway, F. I., "2001: A Space Odyssey," *Spaceflight* **21**, 110–117 (1970).

[53]Smith, A., *The Wealth of Nations, Books I–III*, Penguin Books, Baltimore (1976).

[54]Hosek, W. R., "Economics and the Fermi Paradox," *J. Br. Interplanet. Soc.* **60**, 137–141 (2007).

[55]Belanger, D. O., Deep Freeze: *The United States, the International Geophysical Year, and the Origins of Antarctica's Age of Science*, 1st Ed., University Press of Colorado, Boulder, CO (2006).

[56]Stoneley, R. "The International Geophysical Year," *Nature* **188**, 529–532 (1960).

[57]Officer, L. H., "Purchasing Power of British Pounds from 1264 to 2007," *Measuring Worth*, http://www.measuringworth.com/ppoweruk (accessed 3 Sept 2007).

[58]Officer, L. H., "Dollar–Pound Exchange Rate from 1791," *Measuring Worth*, http://www.measuringworth.org/exchangepound (accessed 3 Sept 2007).

[59]Officer L. H., and Williamson, S. H., "Purchasing Power of Money in the United States from 1774 to 2007," *Measuring Worth*, http://www.measuringworth.com/ppowerus (accessed 3 Sept 2007).

[60]Wikipedia contributors, "AFI's 100 Years...100 Movie Quotes," Wikipedia, http://en.wikipedia.org/wiki/AFI%27s_100_Years..._100_Movie_Quotes (accessed 3 Sept 2007).

[61]Bono, P., and Sapp, T. P., *Operational Plan and Cost Formulation*, Vol. VI, Douglas Aircraft Company Missile & Space Systems Division SM-42969 (1963).

[62]Wade, M., "Rombus," *E. Astronautica*, http://www.astronautix.com/lvs/rombus.htm (accessed 3 Sept 2007).

[63]Aerospace Industries Association, *Aerospace Facts and Figures 1998/1999*, http://www.aia-aerospace.org/stats/facts_figures/ff_98_99/facts_figures9899.cfm (accessed 3 Sept 2007).

[64]Aerospace Industries Association, *Aerospace Facts and Figures 2004/2005*, http://www.aia-aerospace.org/stats/facts_figures/ff_04_05/facts_figures0405.cfm (accessed 3 Sept 2007).

[65]Koelle, H. H., *Handbook of Astronautical Engineering*, McGraw-Hill, New York (1961).

[66]Ehricke, K. A., and D'Vincint, F., *NEXUS—Concept of a Large Reusable Launch Vehicle*, Paper 63-277, AIAA Summer Meeting, Los Angeles, CA, http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19630011856_1963011856.pdf (1963).

[67]Ehricke, K. A., "Economy of Large Launch Vehicles Including Orbital Labor Cost," *J. Spacecraft* **1**, 611–619 (1964).

[68]Koelle, H. H., "Discussion by H. H. Koelle," afterword to Ehricke, K. A., "Economy of Large Launch Vehicles Including Orbital Labor Cost," *J. Spacecraft* **1**, 619 (1964).

[69]Parkinson, B., "Planetary Exploration—Towards New Worlds or New Possibilities," *J. Br. Interplanet. Soc.* **54**, 229–235 (2001).

[70]Wikipedia contributors, "Selling Coal to Newcastle," *Wikipedia*, http://en.wikipedia.org/wiki/Selling_coal_to_newcastle (accessed 3 Sept 2007).

[71]Canal Museum website, http://canalmuseum.com (accessed 3 Sept 2007).

[72]Schwartz, S. I. (ed.), *Atomic Audit: The Costs and Consequences of U.S. Nuclear Weapons Since 1940*, Brookings Institution Press, Washington, DC (1998).

[73]U.S. Congress, House, Subcommittee on Manned Space Flight of the Committee on Science and Astronautics, *1974 NASA Authorization, Hearings on H.R. 4567, 93/2, Part 2*, 93rd Cong., 1st sess., p. 1271 (1973).

[74]Sackheim, R. L., "Overview of United States Space Transportation Technology and Associated Space Transportation Systems," *J. Propul. Power* **22**, 1310–1333 (2006).

[75]Woodcock, G., "High-Power Solar Electric Propulsion for Human Exploration Missions," *J. Br. Interplanet. Soc.* **59**, 230–238 (2006).

[76]The World Bank, *Data & Statistics*, http://www.worldbank.org/data/databytopic/gdp.html (accessed 9 Sept 2007).

[77]NASA, *NASA FY 2007 Budget Request Summary*, www.nasa.gov/pdf/142459main_FY07_summary.pdf (accessed 9 Sept 2007).

[78]Adams, D., *The Ultimate Hitchhiker's Guide to the Galaxy*, Random House, New York (2002).

[79]Wikimedia contributors, "The Transneptunians Size Albedo Color," Wikimedia Commons, http://commons.wikimedia.org/wiki/File:TheTransneptunians_Size_Albedo_Color.svg (accessed 3 Sept 2007).

[80]Wikimedia contributors, "The Transneptunians 73AU," Wikimedia Commons, http://commons.wikimedia.org/wiki/File:TheTransneptunians_73AU.svg (accessed 3 Sept 2007).

[81]Wikimedia contributors, "Eris Orbit," Wikimedia Commons, http://commons.wikimedia.org/wiki/File:Eris_Orbit.svg (accessed 3 Sept 2007).

[82]Cox, A. N., *Allen's Astrophysical Quantities*, 4th Ed., American Institute of Physics, New York (2000).

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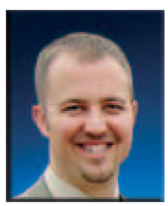

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