Nuclear chemistry half life problems worksheet

Continue

Equation: $f(t) = a\left(\frac{1}{2}\right)^{\frac{t}{h}}$

a = initial amount
t = time
h = half-life time

f(t) = remains of material

The half-life of Po-214 is 0.001 seconds.
How much of a 10g sample is left after 0.003 seconds?

A: 10g

t: 0.003 secs

h: 0.001 secs

$f(0.003) = 10\left(\frac{1}{2}\right)^{\frac{0.003}{0.001}}$

$f(0.003) = 1.25$

Problem: A Nuclear reactor produces 20kg of uranium-232. If the half life of uranium is about 70 years, how loong will it take to decay to 0.1kg?

Solution: $N_0 = 20kg$ $\quad t_{\frac{1}{2}} = 70$ years

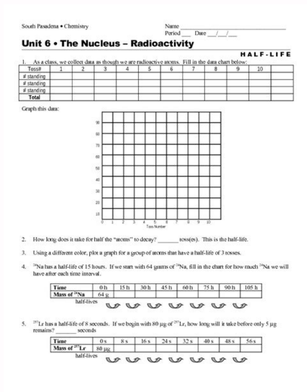
$N = 0.1kg$

$$t_{\frac{1}{2}} = \frac{t}{\log_{\frac{1}{2}}\left(\frac{N(t)}{N_0}\right)}$$

$$t = \left(t_{\frac{1}{2}}\right)\log_{\frac{1}{2}}\left(\frac{N(t)}{N_0}\right)$$

$$t = (70 \text{ years})\log_{\frac{1}{2}}\left(\frac{0.1kg}{20kg}\right)$$

$$= 535 \text{ years}$$



South Pasadena • Chemistry

Name _____
Period ___ Date ___/___/___

Unit 6 • The Nucleus – Radioactivity

HALF-LIFE

1. As a class, we collect data as though we are radioactive atoms. Fill in the data chart below.

Toss#	1	2	3	4	5	6	7	8	9	10	
# standing											
# standing											
# standing											
Total											

Graph this data:

2. How long does it take for half the "atoms" to decay? _____ toss(es). This is the half-life.
3. Using a different color, plot a graph for a group of atoms that have a half-life of 3 tosses.
4. ^{24}Na has a half-life of 15 hours. If we start with 64 grams of ^{24}Na, fill in the chart for how much ^{24}Na we will have after each time interval.

Time	0 h	15 h	30 h	45 h	60 h	75 h	90 h	105 h
Mass of ^{24}Na	64 g							

half-lives

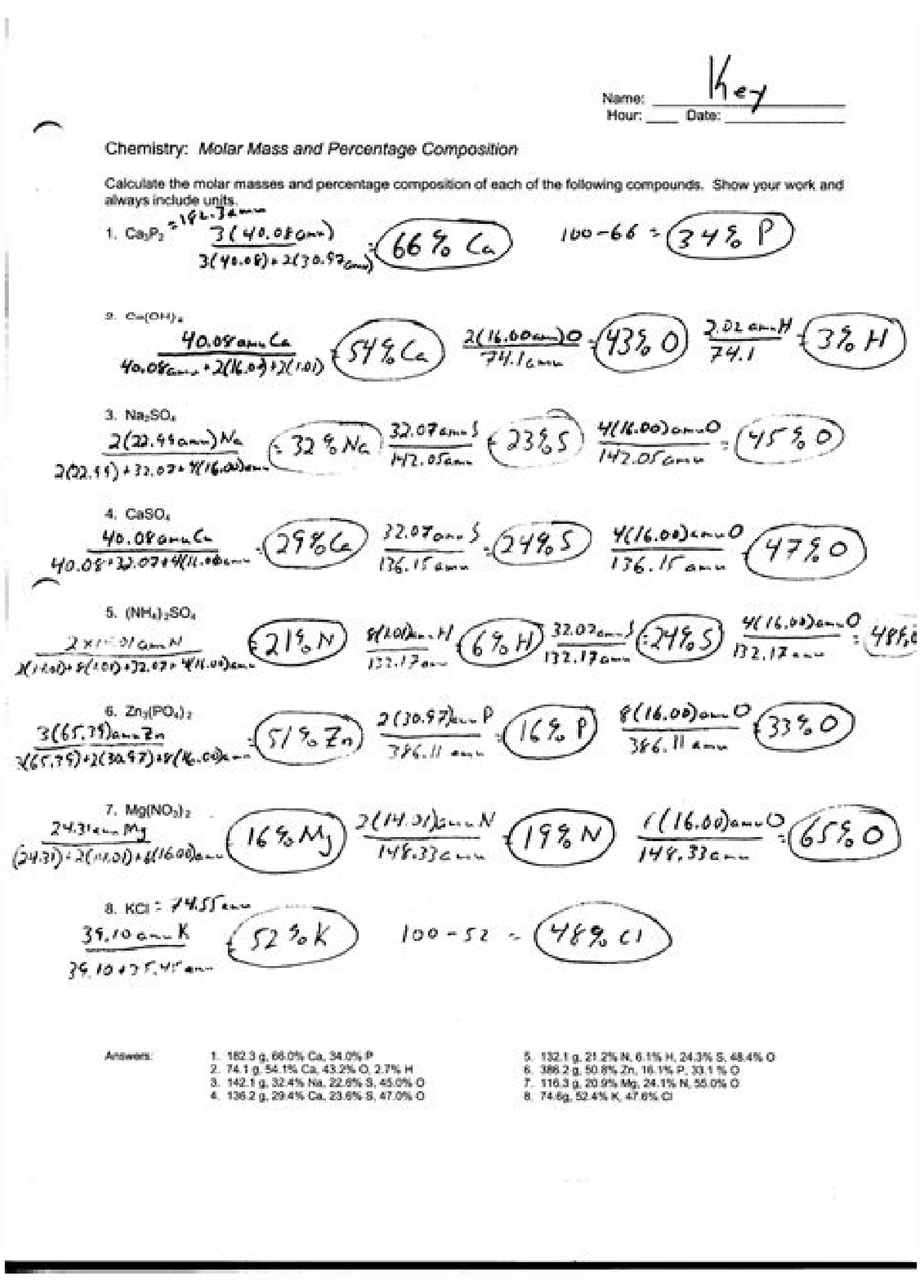
5. ^{191}Lr has a half-life of 8 seconds. If we begin with 80 µg of ^{191}Lr, how long will it take before only 5 µg remains? _____ seconds

Time	0 s	8 s	16 s	24 s	32 s	40 s	48 s	56 s
Mass of ^{191}Lr	80 µg							

half-lives

Table O
Symbols Used in Nuclear Chemistry

Name	Notation	Symbol
alpha particle	$^{4}_{2}He$ or $^{4}_{2}\alpha$	α
beta particle (electron)	$^{0}_{-1}e$ or $^{0}_{-1}\beta$	β^-
gamma radiation	$^{0}_{0}\gamma$	γ
neutron	$^{1}_{0}n$	n
proton	$^{1}_{1}H$ or $^{1}_{1}p$	p
positron	$^{0}_{+1}e$ or $^{0}_{+1}\beta$	β^+



Name: Key
Hour: ____ Date: _____

Chemistry: *Molar Mass and Percentage Composition*

Calculate the molar masses and percentage composition of each of the following compounds. Show your work and always include units.

1. Ca_3P_2 = 182.3 amu — 66% Ca, 100 − 66 = 34% P
2. $Ca(OH)_2$ — 54% Ca, 43% O, 3% H
3. Na_2SO_4 — 32% Na, 23% S, 45% O
4. $CaSO_4$ — 29% Ca, 24% S, 47% O
5. $(NH_4)_2SO_4$ — 21% N, 6% H, 24% S, 48% O
6. $Zn_3(PO_4)_2$ — 51% Zn, 16% P, 33% O
7. $Mg(NO_3)_2$ — 16% Mg, 19% N, 65% O
8. KCl = 74.55 amu — 52% K, 100 − 52 = 48% Cl

Answers:

1. 182.3 g, 66.0% Ca, 34.0% P
2. 74.1 g, 54.1% Ca, 43.2% O, 2.7% H
3. 142.1 g, 32.4% Na, 22.6% S, 45.0% O
4. 136.2 g, 29.4% Ca, 23.6% S, 47.0% O
5. 132.1 g, 21.2% N, 6.1% H, 24.3% S, 48.4% O
6. 386.2 g, 50.8% Zn, 16.1% P, 33.1 % O
7. 116.3 g, 20.9% Mg, 24.1% N, 55.0% O
8. 74.6g, 52.4% K, 47.6% Cl

Home Science Chemistry Lifestyle | Daily Life | News | The Sydney Morning HeraldWe're sorry, this feature is currently unavailable. We're working to restore it. Please try again later. Full PDF PackageDownload Full PDF PackageThis PaperA short summary of this paper25 Full PDFs related to this paperDownloadPDF Pack In order to continue enjoying our site, we ask that you confirm your identity as a human. Thank you very much for your cooperation. This year has been the 50th anniversary of the first man on the moon. That was the culmination of the space race between the US and Soviet Russia. That accomplishment was a whole lot more about "flags and footprints" and experimental flight test, than it was about science or real exploration.This article builds upon some earlier articles posted upon this site.It presents the latest version of my Mars mission outline plan,with an enlarged manned transport,and the latest sizing of 1-stage 2-way reusable chemical landers.These earlier articles are as follows:Why We Should Go Back (And Farther Still)Is there anything worthwhile to accomplish out there?Yes,definitely! In the longer term,there are those future off-world settlements and the associated future economies.I cannot tell you the details of how this might benefit us,because it has yet to be done.But it has always proven beneficial in prior centuries here on Earth. In the shorter term,there are the possibilities of space resource businesses,and of planetary protection against rogue asteroid and comet impacts.That second item is the most important of all:there is simply no better reason for continuing both unmanned and manned space programs than finding ways to protect the folks back home! It's not about winning some race,and it's not so very much about doing pure science just for the sake of knowledge.It's about real exploration of the unknown,something hard-wired into humans.In centuries past,this was exploration of the unknown parts of the Earth.Now it is about space and the deep ocean floor.This article is concerned with the real exploration of space. "Exploration" is a really an emotionally-loaded code word,something most people do not think about.What it truly means is you go there to find out "what all is there" (resources,including those you don't at first recognize),and "where exactly it is" (how hard to obtain,as well as how much is there).Then you have to stay a while to figure out how to use what you found, in order to cope with living in the local environment. All of that is part of "real exploration". Unless you do that correctly,there is no real possibility of future settlements and the associated future economies,or any of the benefits that would ultimately derive therefrom!There is no way to accomplish much of anything else,except just the "flags and footprints" act of going there and returning (which is the bulk of what Apollo itself really accomplished at the moon). Those who "get there first" do tend to do a little better in the long run,in terms of those benefits,provided that they do it "right" when they go.That is one crucial lesson from history.My Suggestions for the Near Term Establish a continuous human presence on the moon,the first item.Start small and expand it slowly over time. Do the lunar "exploration" thing right,this time. Send humans to Mars as the fulfillment of a dream centuries old,probably the second item.When we go,do the "exploration" thing right,from the very first landing.Further,it starts long before the first item (going to the moon) is "done" in any sense of that word. But,any vehicle capable of taking crews to Mars can also take a crew to near-Earth asteroids and comets.Visit those asteroids and comets and properly explore them,in order to learn how to defend against their impacting Earth,as well as "ground truth" for how to really do space mining. That's the third item,but it is just as easily done,and at least as important,as going to Mars. Maybe we do them at pretty much the same time.Ethically and Responsibly Addressing Known Risks For Spaceflight We are ethically-bound to address the known risks of manned spaceflight as best we can.There is a whole long list of safety risks associated with any sort of manned spaceflight.Three come to mind as the most truly credible risks:(1) reliability of,and escape from,spacecraft and booster rockets,(2) microgravity diseases,and (3) exposure to space radiation. The first one has cost us three American crews totaling 17 people dead (Apollo 1,shuttle Challenger,and shuttle Columbia). Each caused a year-or-more stand-down,and very expensive investigations,plus very expensive changes. The two shuttle losses were ultimately caused by bad management decisions valuing cost or schedule above safety.Apollo 1 was about a really-poor basic management attitude ("good enough for government work") combined with technical ignorance,because we had never done this sort of thing before. Those outcomes and their actual causes are why I claim "there is nothing as expensive as a dead crew,especially one dead from a bad management decision".Bear in mind that those expenses are both economic and political (which includes public opinion as well as DC politics). Making spaceflight more safe, from a reliability and escape standpoint, is now also something we already know how to address!This takes careful design allowing for failure modes,redundant systems, and copious verification testing.Mitigation efforts will never be perfect,but they can be quite good. Ethics requires that you treat this as a required constraint upon your designs. It means you always provide "a way out" for your crew at every step of the mission.It really is as simple (and as hard to do) as that!This very seriously constrains your overall mission architecture,as well as your detailed space vehicle designs. The other two have been long studied in low Earth orbit,where microgravity exposure is

inherent in everything we have done there,and radiation exposure is somewhat more than on Earth's surface,but less than outside the Van Allen radiation belts (and far less than inside the belts themselves).Microgravity Diseases Microgravity has proven to affect the human body in a variety of expected, and unexpected, ways.The longer one is exposed,the worse the various diseases become.Beyond the bone decalcification and muscle-weakening that we have long expected,there are also degradations of the heart and circulatory system,degradation of vision from eye geometry changes due to the fluid pressure redistribution,immune system degradations that we have yet to understand,and most recently genetic changes whose meanings are still a total mystery. No doubt more will be discovered,as that has been the trend. The longer exposed,the longer it takes to recover upon returning home,with full recovery actually still in doubt for some of the effects,despite diet,drugs,and exercise. The practical time limit seems to be only a bit more than a year.For that very reason,usual practices on the International Space Station (ISS) call for 6 months to a year's exposure at most,with 6 months the preferred limit. We do know that something near one full Earth gravity (one "gee") is therapeutic,precisely because that is what we evolved in.So,until we know better,any artificial spin gravity schemes need to supply very near one gee,in order to obtain the full Earthly benefits that we already know will work. Destinations outside of Earth-moon space are very much further away than the moon:one-way travel times range from near 6 months to multiple years.This is pretty much outside the preferred limit of microgravity exposure that we have already established on ISS. Mars is 6-to-9 months away one-way,and we do not know how therapeutic its lower gravity (38%) really is for the rigors of the return voyage.Other destinations are further away still,and all those we can actually land upon, are even lower gravity than Mars.That situation says quite clearly that we need to provide artificial gravity (no matter how inconvenient that might otherwise be !!!!) at something near one gee (until we actually know better !!) during these one-way transits to-and-from,in order to best preserve the health of the crews. Ethically,you simply cannot argue with that conclusion,no matter how inconvenient for design purposes,or for total mission cost purposes.That is the only "box on thinking" applied here.Supplying Artificial Gravity There is as yet no such thing as "Star Trek"-type artificial gravity.The only physics we have to serve that purpose is "centrifugal force".You must spin the vehicle,to generate "centrifugal force" as an equivalent to gravity.If the spin rate is low,then Coriolis forces (something everyone has experienced on a merry-go-round) become less important,and so fewer folks can tell the difference between this and real gravity,and there are fewer problems with disrupting the balance organs in the ear. The physics of spin say that the acceleration you feel is proportional to the radius of spin and to the square of the spin rate.The actual physics equation saysa = R w2where a is the acceleration,R the spin radius,and w the spin rate Another form expressed in gees,and not absolute acceleration units, isgees = 1.00 * [(R, m) / (55.89 m)] [(N, rpm) / (4 rpm)]2 Earthly experience with spin rates says that normal untrained and unacclimatized people can tolerate 3 to 4 rpm immediately,and for long-term exposures,without getting motion sick.People extensively trained might (or might not) tolerate higher spin rates in the 8-12 rpm class, without getting motion sick from long exposures.Still-higher spin rates (16+ rpm) induce blood pressure gradients head-to-toe in a standing individual, that are just unacceptable for long term exposures.Stand up,and you faint. 3-dimensional objects typically have 3 axes.About these axes these objects have properties called "mass moment of inertia" that relates to spin dynamics.Usually,higher moment of inertia correlates with a larger dimension along some axis perpendicular to the actual spin axis.These are typically proportional to mass,but proportional to the square of its distance from the center of gravity. There are two (and only two !!) stable spin modes for most objects: about the axis for highest moment of inertia (longest dimension), and about the axis for lowest moment of inertia (shortest dimension). The first case is exemplified by a baton twirler's spinning baton, and the second case is exemplified by a spinning bullet or artillery shell. There are no other stable modes of spin.See Figure 1.Figure 1 – Modes of Spin click for larger image Clearly, building a "spinning rifle bullet" 112 m in diameter at 4 rpm for one full gee at its outer girth is not so very feasible: this is just too big to afford at this time in history. But spinning a smaller-diameter "something" that is 112 m long, end-over-end at 4 rpm, for 1 gee at each end, would indeed be a feasible thing to attempt! That says select the baton-spin mode for practical designs. We already know a lot about the transient dynamics of spinning rigid objects, something important for spin-up and spin-down, as well as for applying any thrust while spinning. There would be no fundamental engineering development work to design a long, narrow spacecraft that spins end-over-end for artificial gravity. There would only be proving-out the specific design in tests before we use it. The most-often-proposed alternative is a cable-connected structure, because it is conceptually easy to reel-out long cables between two small objects. Cable-connected transient dynamics for spin-up and spin-down, and especially for applying thrust while spinning, are incredibly complex and still not very well-known. "You cannot push on a string", that is the complication! So there is a huge fundamental engineering development effort needed, beyond just proving-out the actual design to be used. What this really says is that the preferred near-term spacecraft design is a long and rigid, more-or-less cylindrical shape, to be spun end-over-end, baton-style. This will generate varying artificial gee from a maximum near the ends, to zero at the spin center. We know that microgravity vs gravity has no impact while prone sleeping, or else Earthly bed rest studies would not be a decent surrogate for some of the in-space microgravity effects. That means you can put the sleeping quarters in the low gravity section of the spacecraft near the spin center, and just put the daily workstations in the full-gravity sections of the spacecraft near the ends. See Figure 2.Figure 2 – Why Selecting Baton Spin Mode Is Wisest Choice click for larger image There are basically three radiation hazards to worry about: galactic cosmic rays (GCR), solar flare events (SFE), and the Van Allen radiation belts about the Earth (or similar belts around some of the outer planets). All three hazards are atomic or subatomic particles, just at different speeds and quantities. The threats they pose are location-dependent. GCR is a very slow drizzle of really-high-speed particles, moving at a large fraction of the speed of light. Particles that energetic are very difficult to shield against, because they penetrate deeply into shielding material, and quite often create "secondary showers" of other harmful radiation when they strike the atoms in the shield material. If the shielding atoms are low atomic weight, the secondary shower effect is greatly reduced. GCR comes from outside the solar system. Its quantity is affected by the solar wind, in turn affected by the sun's sunspot cycle, which is about 11 years long. The solar wind is stronger when sunspots are active, making GCR lower in the vicinity of the Earth-moon system at that time. From NASA's radiation effects website (ref. 1) I obtained these values that apply in the general vicinity of the Earth-moon system. GCR maximizes at about 60 REM per year when the sun is quiet, and minimizes at about 24 REM per year,   when sunspots are most active. To "calibrate" the effects of what may be unfamiliar units of radiation, the natural Earthly background radiation is about 0.3 REM per year (and up to 10 times higher in some locations), and a lethal dose would be 300 to 500 REM accumulated in a "short time", meaning hours to a week or so. (Just for information, 1 Sievert is 100 REM.) The NASA astronaut exposure standards are set at about twice the levels allowed for Earthly nuclear workers. Those NASA standards are no more than 50 REM per year, no more than 25 REM in any one month, and a career limit that varies with age and gender, but maxes-out at no more than 400 REM accumulated over an entire career. These career limits are predicated upon a single-handful percentage increase in the likelihood of late-in-life cancer.   Clearly, with a very modest shielding effect (to reduce worst-case 60 REM to an acceptable 50 REM annual), GCR is not the "killer" it is often portrayed to be. SFE (solar flare events) are different. They are much lower-speed particles, much easier to shield, but there is an incredibly-huge flood of them, when these events happen. They come in very-directional bursts from the sun, at rather erratic intervals. There are usually more of them during times of active sunspots, but they can indeed happen when the sun is quiet. They come at irregular intervals measured in durations of "several months apart". The intensity of a burst can vary wildly from only tens of REM received over a few hours, to tens of thousands of REM received over a few hours. The median dose would be multiple thousands of REM over a few hours. Obviously, for unshielded persons, the great bulk of events like this (those over about 300-500 REM) would be fatal doses, and it is an ugly, irreversible, and miserable death. There was a massively-fatal-level event in 1972 between the last two Apollo missions to the moon, and a low-intensity (non-fatal) event during one Apollo mission to the moon. We had chosen to ignore this SFE threat during Apollo because the short duration of the missions (at most 2 weeks) was small, compared to the typical interval (several months) between events. But, had a large event hit during an Apollo mission, the crew would have died in space in a matter of hours. As it turns out, this actual record shows that Apollo's "ignoring-the-risk-as-low-probability"-assumption was not a good assumption to make! That's 20-20 hindsight, but it is still a crucial lesson to learn! For an extended or permanent return to the moon, or going elsewhere, radiation shielding is obviously imperative! On Earth, we are protected from these SFE's (and the GCR) by both the Earth's magnetic field and its atmosphere. These are a very real threat anywhere outside the Earth's magnetic field! In low Earth orbit, we are protected only by the magnetic field, and the background exposure there is higher than down on Earth, but still much less than beyond the magnetic field. The Van Allen belts are concentrated regions of these same radiation particles trapped in the Earth's magnetic field. The intensity is lethal on a scale of days-to-weeks, but tolerable on a scale of hours-to-a-day-or-so. The inner boundary is not sharp, but this is generally considered to become a problem at about 900 miles orbit altitude, and extending many tens of thousands of miles out from the Earth. The exception is the "South Atlantic Anomaly", where the inner side of the Van Allen belt dips down locally to low Earth orbit altitude (100-300 miles). Satellites and spacecraft in high-inclination orbits inherently pass through the South Atlantic Anomaly every several orbits. The ISS does indeed encounter this threat, it being short "flashes" of exposure that accumulate over time, but these still fall well within the astronaut exposure standards (no more than 50 REM annually, no more than 25 REM in any one month). Their main effect is accumulation toward career limits. Spacecraft traveling to the moon or elsewhere must transit the Van Allen belts. Because of the potential for lethal exposure if you linger within them, such transits must be made quickly! Apollo did this correctly, transiting within only several hours. Given the state of today's electric propulsion technology, this rules out using electric propulsion for people to leave Earth orbit for the moon or elsewhere, because the spiral-out time is measured in multiple months. That would quickly accumulate to a lethal exposure, even with some shielding. Passive Shielding The same NASA radiation site has data regarding the shielding effects of typically-considered materials. Those are hydrogen, water, and aluminum. Mass of shielding above a unit exposed area turns out to be the "correlating variable", and 15-20 g/cm2 seems to be enough to generally address the worst SFE. Hydrogen has the lowest density, requiring the thickest layering, but also has the least secondary shower potential, when used against GCR. 211 to 282 cm of liquid hydrogen suffices. 15-20 cm of water is 15-20 gm/cm2, same shielding effect as a really thick layer of hydrogen. Water molecules are still light enough not to have much secondary shower risk. Aluminum would be the thinnest layer, but with the greater secondary shower effect. However, of the practical metals, its atoms are the lightest, and this secondary shower effect is deemed tolerable with it. 6-8 cm thick aluminum plate would be required. That is quite out-of-line with current spacecraft hull design practices: something nearer a millimeter. Other materials based on polymers, and even most rocket propellants, are light-enough atoms to be effective shielding with a low secondary shower risk, yet with densities roughly in the same ballpark as water, for a thinner layer thickness. So, any of these could be practical shielding materials! Because weight is critical, what you have to do is not simply add shielding weight to your design, but instead rearrange the distribution of masses you already otherwise need, so that they can also serve as radiation shielding. You will need meteoroid shielding and thermal insulation, and any manned craft will have water and wastewater on board, as part of the life support system. All spacecraft will need propellant for the next (and subsequent) burns. You use a combination of these, acting together. The real suggestion here is to use water, wastewater, and next-burn propellant tankage as shadow shields, in addition to the meteoroid protection and thermal insulation materials that the manned modules require anyway. It doesn't take much of this at all to cut the worst-case 60 REM/year GCR to under 50 REM/year. It takes only a little more to cut worst-case SFE to safe short-term exposure levels. If you cannot protect the whole manned interior, then the flight control station becomes first priority, so that maneuvers can be flown, regardless of the solar weather. Second priority would be the sleeping quarters, to reduce round-the-clock GCR exposure further. These seriously constrain spacecraft design. See Figure 3 for one possible way to do this, in an orbit-to-orbit transport design concept. This would also be a baton-spin vehicle for artificial gravity during the long transit. Plus, the habitation ("hab") design requires a lot of interior space for the mental health of the crew, something else we know is critical. Somewhere between 100 and 200 cubic meters per person is needed as a minimum, and at least some of it must be reconfigurable as desired by the crew. Spin-up is likely by electrically-powered flywheels in the center module. The vehicle is spun-up after departure, and de-spun before arrival. If a mid-course correction is needed, the vehicle could be de-spun for that, and spun back up for remainder of the transit.   Note in the figure how the arrival propellant and the water and wastewater tankage has been arranged around the manned core to provide extra shadow shielding, for really effective radiation protection. The manned core modules are presumed insulated by polymeric layers that also serve as meteor shielding (while adding to the radiation protection, without being driven by that issue). The pressure shell on the inside of this insulation should be unobstructed by mounted equipment, so that easy and rapid access for patching of holes is possible. There is not time to move stuff when a compartment is depressurizing! Ethics! At departure, the vehicle can be propelled by a different propellant and engine choice, since departure is a short event. The arrival propellant is likely a storable to prevent evaporation losses. Storable return propellant tankage sets can be sent ahead unmanned, for docking in orbit at the destination.   There is an emergency return capsule (actually two capsules) mounted at the center module, each one enough for the entire crew. ("Bailout" at Mars presumes a rescue capability already exists there, so we need redundant engines instead.) Emergency bailout, upon a failed burn for returning to Earth orbit, is the main function of this capsule. Routinely, it could return a crew to Earth from the spaceship, once it is parked safely in Earth orbit. This kind of orbit-to-orbit transport design could serve to take men to Mars or to the near-Earth asteroids and comets. For Mars, the lander craft could be sent ahead unmanned to Mars orbit, and none are needed to visit asteroids. But you cannot send return propellant ahead on an asteroid mission. By refueling and re-supplying in Earth orbit, such a manned hab design could easily be used for multiple missions, once built. Care must be taken in its design and material selection to support many thousands of cycles of use. Thus the craft could safely serve for a century or more, updated with better propellants and engines as the years go by. There I went and wrote a basic "how-to" document for practical and ethical interplanetary spaceship design! Figure 3 – Using Otherwise-Required Materials To Also Serve As Radiation Shielding click for larger image These first few sections so far have been reprised (with edits) from "Just Mooning Around",posted 7-14-19.Everything that follows is new.Mars Mission Outline 2019:Overall The new version uses a larger orbit-to-orbit transport,and recovers the solar-electric tugs that preposition unmanned assets at Mars for the manned mission (2016 did not).It uses similar (but larger) landers as the 2016 version,and it still jettisons the Earth departure stage without recovery. That last could be addressed by fitting the departure stage with a second propulsion system, possibly electric,and putting it into a 2-year-period orbit after stage-off.Then it could be captured into Earth orbit for reuse.That recovery possibility is beyond scope here in the 2019 version. Consider it to be a "future update". Main point here:if one does spin gravity in a baton-spin mode,the resulting transit vehicle is ill-adapted for a direct entry at Mars,or a direct entry at Earth.Such a design is far better-adapted as an orbit-to-orbit transport,with any Mars lander function relegated to a separate vehicle,sent separately.Long-life reusability also points toward an orbit-to-orbit transport design,free of entry heat shield requirements.It means we base our exploration forays onto the surface of Mars from low Mars orbit. The resulting mission architecture requires that both the landers and the Earth return propellant get sent ahead unmanned to parking orbit about Mars,with the manned orbit-to-orbit transport arriving afterward,and rendezvousing in Mars orbit with those items.This powerful concept is not unlike the Lunar Orbit Rendezvous architecture that made it possible to mount each Apollo landing mission with only one Saturn 5 booster.See Figure 4 for the overall mission architecture.Figure 4 – Overall Mars Mission Architecture Requiring Mars Orbit Rendezvous click for larger image The landers themselves are envisioned as one-stage reusable articles that make multiple flights, based out of low Mars orbit. Sending 3 landers ahead with their propellant supply allows one lander to make a landing with only part of the human crew, with a second lander in reserve as a rescue craft. Thus, there is a "way out" even during the landings, unlike with Apollo! Because of storability concerns, the wisest choice is that the lander propellant and engine design be the same as the transport propellant and engine design. This maximizes the interchangeability of engine hardware and propellant supplies, in the event that there are mishaps from which to recover, without aid from Earth. It also simplifies the overall design and hardware development and prove-out. The presence of a third lander allows one lander to become unserviceable, while still maintaining the reserve rescue lander capability, without which landings so far from Earth become too risky to ethically attempt. This is shown in Figure 5, including the velocity requirements for the lander design. The initially-sized version of the lander design concept was used in the 2016 posting, and came from one of the options explored in another posting titled "Reusable Chemical Mars Landing Boats Are Feasible", dated 31 August 2013. These landers are resized somewhat for this posting.Figure 5 – Surface Landing Forays Based Out Of Low Mars Orbit click for larger image Note that for a rescue possibility to exist, some of the crew must stay in the transport in low Mars orbit, while others descend to the surface in a lander. Because we do not know how therapeutic Mars's 0.38 gee gravity might be for the surface crew, I suggest we spin the transport for artificial gravity while it is in orbit, de-spinning for lander departures and arrivals. Thus everybody stays fully healthy no matter what, while we alternate crews on the surface. Now, overall, it is worst-case 9 months to and from Mars, and in any case, 13 months at Mars waiting for the orbital "window" to open for the voyage home. That last is simply inherent from the choice of min-energy Hohmann transfer orbits. That leaves a long time for the crew to explore on Mars. That plus the possibility that the initial landing site might not prove to be desirable, makes it wise to plan for multiple landings, at possibly-multiple sites. Basing exploration forays from low Mars orbit is what makes multiple landings at multiple sites possible at all! No other mission architecture can provide this capability. It is that orbit-based architecture allowing for multiple landings which lets us alternate roles for the crew, so that all of them get to spend time on the surface of Mars (unlike what was possible with Apollo). With a mission crew of 6, that means we could send down alternating crews of 3 in the lander, while the other 3 do science from orbit and provide the critical watchdog rescue capability with the other two landers (two for the reliability of redundancy). It is already known that odd numbered crews fare better in hazardous situations, there being no possibility of the stalemate of ties, in decision-making. Given the existence of the rescue capability from low Mars orbit, we can address lander reliability in two ways, thus increasing the odds of success, and also the odds of still saving the lander crew, if things go seriously wrong. (We are ethically bound to do this!) First, the lander must use redundant engines, so that if one fails, the remaining engine (or engines) can still perform the mission.   Second, the crew piloting cabin could be rigged as an abort-to-surface (or abort-to-orbit) capsule, in the event that too many redundant engines fail, or that there is some overall catastrophic failure of the lander. The minimum number of landings is two, one for each half of the crew. Allowing some time for reconnaissance-from-orbit prior to the first attempt, and for preparations for returning to earth, we can plan on 12 months total for the landings, splitting the remaining month between those other two needs in orbit about Mars. That does cover up to two possible landing sites in the one voyage to Mars! The surface crew will live inside the lander on the surface. That means it must carry them, their exploration gear, and up to 6 months of life support supplies, on each trip. More exploration gear could be carried to the surface if we shorten the stay for each lander. If four trips will be made, that's 3 months each (not 6), and one can trade away life support supplies for extra exploration gear carried down. That could cover up to four possible landing sites in the one trip to Mars, and each crew of 3 making 2 trips, all with the same overall resources sent to Mars, excepting the total lander propellant supply. Continuing that logic, if 6 trips are planned, that's 2 months each, each crew of 3 making 3 trips, and a higher weight of exploration gear relative to life support supplies. That's up to 6 separate sites that could be explored in the one voyage to Mars! Or, 12 trips of 1 month each, which is up to 12 sites explored. Since the lander propellant is sent ahead by SEP, it is rather easy to afford such a capability. The biggest mass ratio-effective burn for the lander is the ascent burn, which can be at significantly-reduced payload, since wastes can be left on the surface along with some exploration gear, while the weight of a plethora of samples is far less than the weight of gear and supplies during the less-demanding descent.   That makes the overall 5.22 km/s delta vee far more affordable with an overall realistic mass ratio and storable propellant specific impulse (Isp).   Those considerations very dramatically impact and constrain the design of the lander.Sending Assets Ahead Unmanned The unmanned transfers can be done more efficiently (lower total mass to be launched) with solar electric propulsion (SEP). The manned transport uses short-burn chemical rocket propulsion to avoid long spiral-out/spiral-in times. (An SEP-based transport would give the crew a lethal radiation dose spiraling-out through the Van Allen belts on departure from Earth, and again spiraling-in through the belts on return to Earth.) At least approximately 0.1 gee vehicle acceleration is required to qualify as a gravity loss-free "short burn". This prepositioning of assets at Mars using SEP was also a part of my 2016 Mars mission posting. The differences here are that I recover the SEP "tugs" for reuse on future missions, and that I use a larger "hab" for the orbit-to-orbit transport. Earth Departure of Manned Transport The Earth departure can be done with higher-performing LOX-LH2 tankage and engines on one end, that are staged off after the burn. To recover these, a higher aphelion orbit with a 2 year period is required, plus some sort of propulsion to return to Earth orbit. This could be electric, or some storable propellant rockets. (Expecting LOX-LH2 cryogens not to evaporate over a 2 year period is just nonsense!) I did not include that here, but it is required for more reusability. That's a future growth item.Velocity Requirements for the Mission The orbital mechanics of min-energy Hohmann transfer determine the minimum velocity requirements for the manned (and unmanned) vehicles, as well as the one-way travel time. Shorter flights require more energy, which is more propellant and tankage that must be sent to low Earth orbit and assembled. The basic velocity requirements for the manned orbital transport are shown in Figure 6. These take the form of unfactored orbital mechanics values serving as the mass ratio-effective values for vehicle design. This is allowable because all these chemical rocket propulsion burns are "short" and exoatmospheric. The resulting mass-ratio-effective design values are given in Figure 7.Figure 6 – Orbital Velocity Requirements For The Orbit-to-Orbit Manned Transport click for larger image For only Mars arrival with the manned transport, there is a need for a rendezvous propellant allowance. It is necessary to adjust orbital position to coincide with the assets sent ahead. As a wild guess, add another 0.2 km/s delta vee to the value shown in Figure 7 as the mass ratio-effective value for design. Figure 7 – Design Velocity Requirements For The Orbit-To-Orbit Manned Transport click for larger image For the assets sent ahead with SEP, design velocity requirements are much more problematic. There are no drag losses, but the gravity losses are huge, since the burns are months long! For a rough rule-of-thumb estimate, just use twice the values in Figure 7. That is what I did here. Propulsion Estimates No particular existing chemical rocket engine's characteristics were used. Ballistic estimates were made "from scratch" using shortcut methods. For both the transport and Earth-departure engines, it was assumed that no gas used to drive pumps was dumped overboard, meaning 100% of the hot gas generated went through the propulsion nozzle. This requires an efficient engine operating cycle. Estimates were made from 1000-psia data for chamber characteristic velocity and gas specific heat ratio, using standard ideal-gas compressible flow methods to develop vacuum thrust coefficient (to include the effects of a nozzle kinetic energy efficiency reflecting streamline divergence). The c* and r "constants" vary with chamber pressure in a way that conforms to empirical ballistic methods I have long used successfully. This gets us to specific impulse (and thus effective exhaust velocity) for vehicle mass ratio determinations with the rocket equation dV = Vex ln(Wig/Wbo). The actual design thrust level is driven by vehicle mass and the min 0.1 gee acceleration requirement, which sizes throat (and exit areas) via the thrust/throat area/thrust coefficient equation F = CF Pc At. That leads to real engine dimensions. For not-quite-the-highest-tech in engine design technology, a good "wild guess" for engine weight would be thrust/50, both in force units, figured at 1 gee Earth gravity for the weight. Assuming redundant engines for safety and reliability, these rockets won't be simultaneously run at full thrust. For vacuum-only operation, there is no need for really high chamber pressure, and there is no need to worry about backpressure-induced separation effects, because there isn't any backpressure. 6-7 mbar on Mars is also effectively no backpressure at all, so the lander engines can be the same vacuum design as the transport engines. Reflecting those considerations, I assumed 1000 psia at max thrust, typical operation at 500 psia, and min throttled-down pressure 200 psia. Others may disagree, but that is what I did. The higher the Pc, the higher the c*, and thus the higher the Isp. But so also the higher is the weight of the engine. The data I got for the NTO-MMH storable transit engines are given in Figure 8. The data I got for the LOX-LH2 Earth departure engines are given in Figure 9. For both I assumed an expansion bell equivalent to a constant 15 degree half-angle conical bell, leading to a kinetic energy efficiency of 0.983 for the nozzle efficiency. Any real-world curved bell will have an average half angle not far at all from that value; it will be slightly shorter than the equivalent conical bell, and just about the same efficiency. Figure 8 – Ballistic Estimates For Storable-Propellant Transit (and Lander) Engines click for larger imageFigure 9 — Ballistic Estimates For Cryo-Propellant Earth Departure Engines click for larger image The solar electric propulsion is more problematical in its characteristics, it being currently available only in small sizes, with scaleup efforts underway at both Ad Astra and NASA. What is important for vehicle design purposes would be thrust/weight for the actual electric thruster equipment, its operating specific impulse, its electric power/thrust requirement, and the type and phase of its propellant (liquids or solids are easier to store at lower total tankage weight than gases).   Add to that the producible electric power/panel area, the weight/panel area, and miscellaneous equipment weight (if any), for the solar power supply equipment, and for autonomous robotic vehicle guidance. The size of the thruster's thrust relative to the full vehicle weight should probably fall near what the current small thrusters on satellites use: something near or above 0.001 gee. Here are the values for the putative system I "chose", it being something that does not yet exist, but likely could be made to exist near-term. Bear in mind the available solar power at Mars is half that at Earth (Mars actually sizes the panels). The value shown for electric power/area of solar panel is for near-Earth space, turned to face the sun directly. This data represents a Hall-effect device on iodine.SEP Items Data Table click for larger image The solar photovoltaic power per unit area was estimated as the solar constant at Earth (in space 1353 W/m2) multiplied by a 20% conversion efficiency of sunlight power to electric power. That represents a high-tech space-industry type of solar cell. The weight was estimated from reported data for the Alta Devices Alta 5x1 2J and Alta 5x1 1J satellite solar panel devices. The miscellaneous equipment is not structure, that is in the weight/area figure for the panels. It is the mass of the autonomous guidance equipment, including things like star trackers, computers, communications, and accelerometers. Space Hab for the Crew: Characteristics I based these guesses off the Bigelow Aerospace B-330 space station module design as seen on the internet (ref. 2). This is the big commercial product, not the simple, small BEAM unit attached to ISS for testing and evaluation by NASA. These are nominally 15.7 m long and 20 metric tons. They are somewhat inflatable, and feature a core equipment and framing structure around which the inflated hull is unobstructed.   There is a meter of layers of micrometeoroid shield that also serve as thermal insulation, and as low-molecular-weight radiation shielding.   Each module contains some 330 cubic meters of interior space. The hard core protrudes on one end, providing a place for solar panels. The modules of the orbit-to-orbit transport cannot be exactly these B-330 modules, but can be something rather similar! Docking multiple modules end-to-end creates the baton-shaped vehicle this mission design needs. The modules must have external features of some type that allow tankage to be attached around the outer periphery, and internal fold-out decks as part of the core. The center module must be very stout, and contain big electrically-driven flywheels for vehicle spin-up and spin-down, plus places to dock space capsules. It would seem wiser to put big solar panels on the center module, with the docked capsules, and the flywheels inside, where spin forces are zero-to-minimal. It is likely to be hard shell, not an inflatable, for strength. That module is also likely to be quite heavy. As a wild guess, call it 16 m long and 40 tons. The others can be nominal 16 m long, and nearer 20 tons, reflecting inflatable pressure shell along almost the entire length, plus the features for attaching external tankage.   Call the internal volume 350 m3 each as a best guess, excluding what the hard core occupies. Counting the center module, some 7 modules each 16 m long docked end-to-end is 112 m long, for 1 full gee at each end if spun at only 4 rpm. That basic structure would total 160 metric tons, using the guesses in the previous paragraph. To that one must add masses for crew and 2 space suits each, their personal effects, and personal equipment (call it 0.5 metric ton per person as a guess), and for fully-expendable supplies of food, water, and oxygen (call it 0.75 metric tons per person per month, knowing that these are just "reasonable guesses"). Crew and supplies must fit within the vehicle, which has (for the 6 modules not filled with flywheels and heavy equipment) some 2100 m3 volume. If one assumes half the volume is packed supplies, and also a crew of 6, that leaves some 175 m3 per person as living space available. That's about like 3 large living rooms in a typical middle-class house. That seems adequate at first glance, if it is well distributed, and some part of it is reconfigurable at some level.   The crew weight allowance is 3 metric tons, and the packed supplies mass is about 4.5 tons per mission month. If the mission is 31 months long (9 months transit, 13 months at Mars, 9 months return), that's about 140 tons of supplies, with no margin for error. So call it a nominal 150 tons. This presumes no recycling or growing-of-food in space or on Mars. It's a worst-case deal, but we can do this "right now".   So, the empty hab section is estimated at 160 tons. It gets loaded with about 150 tons of supplies, allowing for 7.5% safety factor on supply mass, and loaded with about 3 tons of crew with their suits, equipment, and personal effects. Fully loaded, that's 313 tons. That would be crew of 6, and supplies for a 31 month mission plus a small margin. See Figure 10. Figure 11 shows an image of the spreadsheet where these numbers were calculated. Yellow highlighting denotes inputs. Some selected outputs are highlighted blue. Figure 10 – The Estimates for the Hab Section Structure of the Orbit-to-Orbit Transport click for larger imageFigure 9 — Ballistic Estimates For Cryo-Propellant Earth Departure Engines click for larger image Assumed depleted at a constant rate, the supplies total 150 tons at departure, 109.5 tons at Mars arrival, 51.0 tons at Mars departure, and not-zero at 10.5 tons at Earth arrival, assuming the safety margin is not consumed. This presumes wastes are dumped overboard with no recycling at all!   This dumping reduces the effective mass of the hab section, at each mission segment, a benefit to propellant required. We can already do somewhat better than that with recycled water, but this is a worst case estimate! Yet this open-cycle assumption gets the smaller propellant supply for return to Earth. "Efficiency" is not always beneficial: that is too often presumed erroneously! Jettisoned mass reduces next-burn propellant requirements. That's just physics you cannot fight!Sizing the Manned Transport and Its Return Propellant The fundamental notion for sizing propellant supplies for the four events (Earth departure, Mars arrival, Mars departure, and Earth arrival) is that the mass of the loaded, crewed hab, plus the mass of all propellant tankage, plus the mass of the engines, is the ignition mass. That minus the mass of propellant burned from that tankage is the burnout mass. That produces a mass ratio for the burn, and the delta-vee it will produce, which must meet or exceed the requirement for that burn. This is subject to the constraint that we want 0.1 gee or thereabouts as a min vehicle acceleration at each burn. To do this, one must estimate the ratio of propellant to loaded tank mass for the added tankage. This has to reflect a long, slim tank geometry for docking multiple tanks around the periphery of the hab, and it must account for the mass of the docking structures needed to achieve that result. As a guess, I am assuming that the empty tank inert mass (with all those fittings) is 5% of the loaded tank mass, so that the contained propellant is 95% of the loaded tank mass. To that end, I used a series of calculation blocks in a spreadsheet worksheet to run the calculations. Again, inputs are highlighted yellow, and significant outputs are highlighted blue. Figures 12, 13, and 14 show the results. Bear in mind that the loaded tank mass for the Mars and Earth arrival burns must be part of the "payload" for the Earth and Mars departure burns, respectively. They are unique in this way. That means the dead-head payload is the appropriate hab mass plus the mass of the next burn's loaded tanks. The current burn's tanks must push this (plus the added engine mass) to the required delta-vee for that burn. Added engine mass is handled by an iteratively-applied tankage scale-up factor just slightly over unity.Figure 12 – Part 1 of Orbital Transport Propulsion Sizing click for larger image As it turns out, finding the propellant tankage mass to push the hab to the required delta-vee is not an excruciating iterative process. You first find the mass ratio MR that is required from the required mass ratio-effective delta-vee, and the propulsion's effective exhaust velocity, by the rocket equation. Ignoring the mass of the engines themselves, it turns out to be closed-form to find the loaded tankage mass Wtf from that mass ratio, and the total "dead head" mass to be pushed in that burn. For both departures, the "dead head" mass is the appropriate loaded hab mass plus the loaded mass of the corresponding arrival tankage. For both arrivals, the "dead head" mass is just the loaded hab mass. This can be corrected at the 1 or 2% level for total engine mass later, to ensure fully meeting the delta-vee requirements, simply by scaling up the loaded tank mass Wtf with a factor applied iteratively until delta-vee produced meets the requirement.   Wtf = Wdead (MR – 1)/(1 – MR f) where f = Wt/Wtf and Wt is dry tank massFigure 13 – Part 2 of Orbital Transport Propulsion Sizing click for larger imageFigure 14 – Part 3 of Orbital Transport Propulsion Sizing click for larger image That's the orbital transport rough-out design for Mars. It can get there to low Mars orbit from low Earth orbit where it was assembled. It can rendezvous with its Earth return propellant, the Mars landers, and the Mars lander propellant supply, all three of which were sent ahead by electric propulsion.   The nonrecoverable items are the Earth departure stage and the empty Mars departure tanks. The empty Mars arrival tanks are left in Mars orbit. Everything else about this design is recovered in low Earth orbit. Note that this ship is 1413 metric tons, as assembled and loaded in low Earth orbit, ready to go to low Mars orbit. Its use requires that some 997.26 metric tons of loaded propellant tanks be sent ahead to Mars for the return propellant supply. In order to actually make landings on Mars as staged out of low Mars orbit, the landers and their propellant supply must also be sent ahead to low Mars orbit. With much bigger propellant tankage, this same design could take men to a near-Earth asteroid. For such missions, landers are not needed, and there is no practical opportunity to pre-position return propellant, except many years ahead. Those missions are far more difficult. Analysis of one is not attempted in this posting.Sizing the Lander and Lander Propellant Supply The lander payload is its crew, their suits and personal equipment, plus an amount of life support supplies that depends upon how long the crew will live in the lander on the surface, each landing. The de-orbit burn for a surface-grazing ellipse is a trivial 50 m/s delta-vee. Most of the deceleration is aerodynamic drag, effectively terminating at end-of-hypersonics at Mach 3, just about 1 km/s velocity, but at a low altitude because of the high ballistic coefficient. That altitude is only about 5 km! From there, deceleration is by retropropulsion alone, with a large "kitty" to cover hover and divert requirements. Assuming 1 km/s velocity at 5 km altitude, along a straight slant trajectory at 45 degrees, the average deceleration level required is 70 m/s2, or 7.211 gees, which with the lander mass, sets the required engine thrust level for landing. That is a rough ride, about twice the rigors of return from low Earth orbit, and justifying all by itself the maintenance of full crew health with artificial spin gravity! The lander is a one-stage reusable "landing boat" intended to make multiple flights, each fueled from a propellant supply sent with it to low Mars orbit. Factored, the mass ratio-effective descent delta-vee is just about 1.5 km/s. Propellant is storable NTO-MMH, to preclude evaporation losses and massive energy requirements to prevent freezing or boiling. The ascent must account for small but non-zero gravity and drag losses (about 2% of velocity), and a "kitty" for rendezvous maneuvers. That mass-ratio-effective delta vee is just about 3.62 km/s. The payload requirements for crew, equipment, and supplies as a function of surface duration are given in Figure 15, along with a crude estimate of the "larger-than-minimum" vehicle inert weight fraction that is appropriate to the necessary structural robustness, and to the equipment required to function as a reusable entry-capable vehicle, and as a surface habitat. Conceptually, the lander is sketched in Figure 16. Some of its backshell panels double as cargo load/unload ramps. Most of the cargo space can be isolated and pressurized as living space, once unloaded. The piloting cabin is the abort capsule, something somewhat similar to a crew Dragon from Spacex. This thing is NOT a minimalist lander the way the Apollo LM was. Figure 15 – Payload Requirements Vs. Surface Duration click for larger imageFigure 16 – Conceptual Sketch of Reusable "Landing Boat" click for larger image The ascent payload is smaller, since most (but not all) the supplies are used up (and wastes left behind) at ascent liftoff. There is a generous allowance for Mars samples to be returned to the orbital transport. This has to be taken into account in calculating the actual vehicle masses, since the two delta-vees are handled at two different payload fractions, in the one vehicle design. That process is inherently iterative, as shown by the data given in Figure 17. Figure 17 – Iterative Determination of Lander Characteristics vs Surface Duration click for larger image In order to determine these numbers, one guess a value for the max lander mass, which is ignition-at-descent (Wig-des). The inert fraction times this gives the vehicle inert mass Win. The ascent and descent payloads are determined vs mission surface duration separately. The mass ratios already determined are used to estimate propellant masses. The ascent propellant mass Wp-asc is determined first as (MR-asc – 1)(Wpay-asc + Win), then the descent propellant mass Wp-des as (MR-des – 1)(Wpay-des + Win + Wp-asc), treating the ascent propellant as part of the effective "payload" during descent. The descent payload plus both propellant masses plus inert mass sum to the result for descent ignition mass. The input guess for descent ignition mass is then adjusted iteratively, until it converges to the result for descent ignition mass. This is done by simple trial and error in the spreadsheet. There is such a result computed for each of 4 possible surface durations that divide evenly into the 12 months available. These results are then the inputs for a characterization of the lander sizing as a function of design surface duration. For the selected 2-month duration (corresponding to 6 total lander flights), those results are given in Figure 18. These show ascent and descent weight statements, confirmation of delta-vee capability, and characterization of vehicle mass fractions, plus the propellant supply required to cover the appropriate number of flights. Similar tables exist in the spreadsheet for the other 3 durations, but those are not shown here.   Figure 18 – Lander Design Characteristics for 6 Flights of 2 Month Surface Duration Each click for larger image Figures 19 and 20 show the trade-off of vehicle sizes and propellant supply sizes versus surface duration options. The selected design is near the "knee" in the curve of number-of-flights vs surface duration, at 2 month duration for 6 flights. For shorter duration, the required propellant supply is significantly larger. For longer duration, the required propellant supply is smaller, but not so significantly smaller. The lander size itself is significantly affected by the design surface duration, being larger at longer duration. The 2-month duration selected limits this affect, without so significantly penalizing the payload fraction (which ranges from about 2 to about 3%). The selected 2-month duration is also near the "knee" in that curve. Longer durations do not improve this as much as was gained going from 1 month-12 flights to the selected 2 month-6 flights option. Figure 19 – Number of Landings and Required Propellant Supply Vs. Surface Duration click for larger imageFigure 20 – Lander Size and Payload Fraction Vs. Surface Duration click for larger image For this selected design (6 two-month surface stays), three landers fueled and loaded with supplies, less crew, suits, and personal equipment, each massing 376.5 metric tons, must be sent to Mars along with some 1764 tons of propellant to support all 6 flights. If 95% of the tank weight is propellant, the mass of loaded tankage to be sent is some 1856.8 metric tons. If sent as tanks docked to each of the 3 landers, that's a 376.5 ton lander plus 619 tons of loaded propellant tanks. The "smart" thing to do from a reliability / self-rescue standpoint is to send the transport return propellant with those same three landers, so that if one is lost, the transport can still return safely by drawing the shortfall instead from the remaining lander supplies. That return propellant was determined above to be 997.26 metric tons of loaded tanks. Divided by 3, that's an additional 332.4 metric tons of Earth return propellant tankage sent to Mars with each lander. That makes each lander plus propellant tanks a 1327.9 metric ton item to be moved by solar electric propulsion from low Earth orbit one-way to low Mars orbit. Each such is thus quite comparable to the departure mass of the manned orbital transport. That would not be true at the other surface durations. There are 6 landings to be made, and three such landers sent to Mars. Distributed evenly, that is two flights per lander minimum, and 6 maximum. Bear in mind that only one lander is sent to the surface at a time, carrying a crew of 3, while the other three crew do science in orbit, while acting as the safety rescue "watchdog", with at least one functional lander, even if the other one fails. The worst case is that all 6 flights are made with one lander. Thus the lander design must allow for at least 6 flights per vehicle, justifying in part the higher inert mass fraction used in this design rough-out. Landers get left in low Mars orbit at mission's end, when the transport departs for Earth. Subsequent missions might utilize these assets, and reduce the sent mass to only more lander propellant. That possibility argues for much more than 6 flights per vehicle, in turn a really good argument for the very robust inert mass fraction of 20% used here. Alternatively, they could be landed robotically.Common Engine Design for Transport and Lander? The lander mass is 378 metric tons at ignition, and 241 at touchdown, as just determined above. The average is 309.5 metric tons. Also as determined above, the average deceleration required is 70 m/s2. That translates to 21,665 KN of retropropulsion thrust required to safely land (nominally 22,000 KN). This is totaled for multiple engines. Less may be used for ascent, as such high gee capability is not required for that. Something nearer 2 gees at ascent ignition mass 236.3 metric tons (4726 KN thrust) is more appropriate. As described above, something near 1170 KN thrust from multiple engines is the minimum required for the orbit-to-orbit transport.   This was set by the min 0.1 vehicle gee capability at max vehicle mass, and still resulted in only large fractional gee capability at min vehicle mass. This thrust level selection could be doubled or tripled (or more) with relative impunity.   A worksheet page was set up in the spreadsheet to explore how this could be done, in two steps. The results are shown in Figure 21, which indicate the possibility of using some number of 3600 KN max thrust engines, throttleable from 20 to 100%. In the first step, I input factored thrust requirements, plus a number of engines, and a max number of inoperative engines. The thrust requirement for the lander descent is based on slowing the average descent mass (as a constant) from 1 km/s to zero, in a slant path length of 7.1 km, using the oversimplified kinematic equation V2 = 2 a s. This is a very high-gee descent! Reducing that requires not just supersonic retropropulsion, but hypersonic retropropulsion (starting retropropulsion earlier in the entry sequence). It is an inevitable consequence of the high ballistic coefficient producing very low altitudes (on Mars) for end-of-hypersonic deceleration. This is an area for further design work! The thrust requirement for the lander ascent is its Earth weight, factored-up just slightly, to accommodate flight tests on Earth. That's "overkill" for Mars with its lower gravity. The thrust requirement for the orbital transport is based on its Mars departure mass (largest of the masses under storable propulsion) and a min 0.1 gee vehicle acceleration requirement. This is arbitrarily factored-up by 3 to achieve commonality, without exceeding max gees ~ 2 at Earth arrival. That initial result indicated that something like 3600 KN max thrust per engine would be suitable, with 9 engines in the lander operating at part throttle in descent, and 4 engines operating at part throttle in ascent, able to lose up to 3 engines either way, and still function within limits. This was explored further, looking at vehicle gees and engine throttle percentages, in the second step. Up to 3 of these lander engines could cease operation during ascent or descent. The remainder could supply adequate thrust at 100% throttle or less, without waiting for lightoff of any inactive engines. That's an important safety consideration, which ethics demands! Two of these same engines would be adequate to push the orbital transport at part throttle, with only one operating engine still able to supply much more than the demanded minimum thrust.Figure 21 – Determination of Size and Distribution of a Common NTO-MMH Engine Design click for larger image In all cases, engines operate between 20 and 100% throttle setting, and appropriate gee limits are not exceeded. Min transport vehicle gee requirement (0.1 gee) is exceeded. For descent, the lander retropulsion operates between about 6 and about 9 gees. This event is only about 14-15 seconds long!!! "At the last second" to actually land, some 8 of the 9 engines must be shut down to reduce thrust to nearer Mars weight of the lander (about 749 KN to 872 KN, depending upon how much propellant was burned) at touchdown, with the remaining active engine operating at about 21-24% thrust setting. This single-engine point is the riskiest aspect of the landing, but it is mitigated by the facts that (1) this engine is already operating, and (2) it need only continue to operate at reduced thrust for a second or two. On ascent with a reduced number of engines, this is 1.2 to 3.6 gees for the lander at full thrust, far more than is needed to depart against Mars gravity (only 0.38 gee). Active throttling reduces that some. The transport operates between 0.3 and 1.8 gees during the return to Earth. This exceeds the min acceleration requirement, but not the maximum. A 3600 KN engine design for this NTO-MMH common engine would resemble the notional sketch in Figure 22. If the Earth departure stage at 1350 KN uses 5 engines, each would be approximately 1350 KN max thrust capability operating at 20% thrust. Up to 4 could be non-functional, and still easily meet the overall min departure thrust requirement, without exceeding 100% throttle. Higher vehicle acceleration than 0.1 gee is easily obtained, but even with all 5 engines at full thrust, it is still only fractional gee. Such a 1350 KN LOX-LH2 engine would resemble the notional sketch in Figure 23.Figure 22 – Sketch of Proposed 3600 KN NTO-MMH Common Engine (one of 2 transport, 9 lander) click for larger imageFigure 23 – Sketch of Proposed 1350 KN LOX-LH2 Earth Departure Engine (one of 5 on the departure stage) click for larger image This item is the most speculative, because (1) it uses the most assumed data, and (2) this kind of solar electric propulsion has yet to be scaled up to such sizes to push masses this large. To cover the gravity losses (both planetary and solar), I simply doubled the required orbital delta-vee data. I simply assumed the average characteristics of small Hall effect thrusters operating on iodine could be scaled way up by simple clustering, at the same thrust/weight and thrust/power ratios. And, I just assumed the characteristics of satellite-sized solar panels could be scaled up to the low-hundred kilowatt range at the same power/area and weight/area ratios. My approach was a self-contained solar-electric propulsion (SEP) "tug", that incorporates the clustered thruster unit, the solar panels to power it, sized for reduced sunlight at Mars, a robot guidance package, and a low-pressure "tank" to contain the easily-sublimated and inexpensive iodine propellant. I used published data for two Busek Hall-effect thrusters, and for a couple of Alta Devices satellite solar panels, for these estimates. This SEP "tug" is coupled to a dead-head payload for the trip from Earth orbit to Mars orbit, using all of its 120 clustered SEP thrusters to achieve a milli-gee of vehicle acceleration capability at Earth departure. That payload is one (of the three) Mars landers (fully fueled and supplied), plus a 1/3 share of the total lander propellant supply, and plus a 1/3 share of the manned orbital transport's Earth return propellant supply. This dead head payload is over 1300 metric tons. For the return trip (these "tugs" are fully reusable), there is no dead-head payload, only the "tug" and its iodine tank, still containing just enough iodine propellant to get home. During the trip home, only one SEP thruster in the cluster need be used to achieve near a milli-gee of vehicle acceleration at Mars departure. That leaves many "spares in case the one fails", insuring utter reliability. (Outbound, the cluster is large enough that the loss of a few thrusters is no significant percentage loss of thrust.) The size of one such thruster (200 mN, mN meaning milli-Newtons) falls within the range of thrusters produced today. This produces adequate acceleration of the unladen vehicle. The scaleup is by clustering, not by increasing the size of the thrust in such a device. The clustering-together of 120 of these units produces some 24,000 mN, needed to move the laden vehicle at adequate acceleration. The resulting SEP "tug" design is depicted in the sketch of Figure 24. I used a big two-stage spreadsheet worksheet to iteratively size this "tug" system, examining the 4 "burns" individually. The second stage of this process fully defines the characteristics of the "tug" and its estimated performance. This is the tabular data in the partial spreadsheet image shown in Figure 25. Hopefully, this rough-sizing is "overkill", due to my just-assumed doubling of the orbital delta-vee requirements. The intent here is to slowly spiral-out of low Earth orbit to escape, and continue an accelerating spiral about the sun to an appropriate midpoint, then use a decelerating spiral about the sun toward capture at Mars. From there, it follows a decelerating spiral-in to low Mars orbit. The return uses the same spiraling processes, just unladen of dead-head payload, and at far-lower thrust and propellant requirements. Figure 24 – Depiction of the SEP "Tug" Design Sizing Rough-Out click for larger imageFigure 25 – Partial Spreadsheet Image Showing "Tug" Characteristics and Performance click for larger image Of all the items analyzed, this is the easiest and most straightforward, because there is one and only one burn (the Earth departure burn). Then this stage is jettisoned. The stage layout concept and sized data were already determined as part of the orbital transport propulsion sizing above. These data were given as part of Figures 12, 13, and 14 above, plus part of the common engine discussion just above, with sized engine dimensions in Figure 23. Just to summarize, the departure stage has 5 LOX-LH2 engines each designed for 1350 KN thrust, weighing an estimated total of 5.139 metric tons. The stage comprises LOX and LH2 tankage whose combined dry weight is 41.906 metric tons. The total propellant load is some 796.210 metric tons. Thus the loaded stage itself is some 843.255 metric tons. This stage pushes a fully loaded and crewed hab plus Mars arrival propellant tankage that totals some 569.810 metric tons of dead-head payload. Total orbital transport vehicle mass, at Earth departure ignition, is thus some 1413.065 metric tons. This was shown in Figure 14 above, including weight statements and performance. Not considered here is reuse of the Earth departure stage. Its engine sizing would be fine, but it needs larger tanks and propellant to accomplish 2 burns. The first is to put the orbital transport onto a Hohmann transfer ellipse trajectory. After releasing the transport, it burns a second time to enter an ellipse about the sun with an exactly two-year period. That way the Earth is there when it reaches perihelion, thus making recovery feasible at all. It is just not reasonable to expect that cryogens like LOX and especially LH2 will not completely evaporate away over a 2 year interval. Therefore, the reusable form of the stage must also incorporate a second propulsion system storable over long periods. This added propulsion provides the delta-vee to return to Earth orbit from the 2-year solar orbit perihelion conditions.   Being unmanned, there is no reason this second propulsion system could not be solar-electric using iodine. The stage then executes a spiral-in to low Earth orbit after capture. The alternative is storable propellants like the NTO-MMH. Being out of scope here at this time, these designs have not been explored. Consider that as a future upgrade.Totaling Up the Mission and Its Launch Requirements This mission to Mars requires a fleet of 4 vehicles to be sent from Earth orbit to Mars orbit. One of these (the manned vehicle) returns to Earth. The other three are unmanned assets sent ahead earlier by electric propulsion, for the crew to utilize when they arrive by conventional rocket propulsion. The three unmanned vehicles are identical, comprising a dead-head payload and a reusable solar-electric "tug" that returns to Earth for reuse, after delivery of the dead-head payload into orbit at Mars.   That dead-head payload payload is the same for each of these vehicles: an uncrewed but loaded and fueled reusable Mars landing boat, plus 1/3 of the total Mars lander propellant supply, plus 1/3 of the crewed vehicle's Earth return propellant supply. That dead-head payload is 1327.9 metric tons for each of these 3 vehicles. Each of these three unmanned vehicles totals some 2413.5 metric tons as assembled in Earth orbit, that being the dead-head payload plus the fueled SEP "tug". The crewed vehicle (the orbit-to-orbit transport) comprises the crewed and loaded hab section, plus the loaded Mars arrival propellant tankage, plus the expendable Earth departure stage that uses cryogenic propellants. (All the other rocket propulsion uses the same storable propellants, and the SEP "tugs" use sublimable iodine to keep the iodine "tank" weight down.) Ready to depart Earth orbit, the transport and departure stage total some 1413.065 metric tons. The grand total that must be assembled in orbit for the fleet of 4 ships is some 8653.6 metric tons. For that, you get 6 landings at up to 6 different places on Mars, all in the one manned trip to Mars. That's 1442.3 tons to support each of the 6 landings, essentially. These are 2-month max stays at each landing site. You get all this, plus a "way out" or a self-rescue capability built into the mission at every step, plus a fully-healthy crew with radiation shielding and artificial gravity during the transits, and in low Mars orbit. That's a lot of benefit for the cost.Getting Landers To Low Earth Orbit The selected lander design is just about 378 metric tons, crewed, loaded and fueled. Less crew (and their suits and gear), that's just about 376.5 metric tons. Just about 294 tons of that lander weight is propellant. So, a loaded, crewless, empty-of-propellant lander is just about 82.5 metric tons. Remove the supplies, but leave the surface equipment and rover aboard, and this is about 77 tons. Completely unladen, the lander is about 75.6 tons. I looked at SLS (150 metric tons to LEO, guessing $1,000M per launch), Spacex's "Starship" (100 metric tons to LEO, guessing $150M per launch), Spacex's Falcon-Heavy (63 metric tons to LEO flown expendably, about $85M per launch), ULA's Atlas-V (20 metric tons to LEO at about $85M per launch), and Spacex's Falcon-9 (20 metric tons to LEO flown expendably, and $63M per launch). The loaded unfueled lander mass of 75.6 metric tons is out of reach of Falcon Heavy, much less Atlas V or Falcon 9, even if an 8-meter payload diameter could be flown on any of them. NASA's SLS might possibly launch it dry of propellant, maybe even two of them at once, although it has yet to fly. That would be 2 or 3 flights of SLS at $2-3B to put 3 landers into orbit, unladen of propellant. It would be 3 flights of "Starship" at $450M total. The most cost-effective of those two options is "Starship". 3 "Starships" deliver 3 landers loaded but unfueled. At 294 tons of propellant per lander, and 100 tons per "Starship", some 9 "Starship" tanker flights would be required to fuel them fully up. At 150 tons per SLS, some 6 SLS flights would be required to fuel them up fully. At about 60 tons per flight, some 5 Falcon Heavy flights could be those tankers per lander,   for some 15 Falcon-Heavy flights to fuel the 3 landers up. At 20 tons per flight, it would require some 45 flights of Falcon-9 or Atlas-V to fuel the 3 landers in orbit. The most cost-effective way to deliver these bulk liquid propellant supplies turns out to be 9 "Starship" flights, with 15 Falcon-Heavy flights a rather close second. If "Starship", the transfer crew need not be sent up separately.Getting Earth Return and Lander Propellant Supplies to LEO and Docked Remember, we must send to Mars each lander loaded and fueled, plus 1/3 of its Mars landing propellant supply, plus 1/3 of the transport's Earth return propellant supply. These propellant supplies are pre-loaded tanks. They are 1764.1 tons for the lander operations, 541.3 tons for the transport's Mars departure, and 455.9 tons for the transport's Earth arrival. That totals some 2761.3 metric tons of propellant, which must be in tanks, at about 95% propellant and 5% tank inert. Unconstrained by other considerations, I chose to break this up into nominal 60-ton loaded tanks. The lander supply is 31 of these, the Mars departure supply is 10 of these, and the Earth arrival supply is 8 of these. That's a total of some 49 tanks to deliver to LEO, at 60 metric tons each. The most cost-effective way to do this was 49 flights of Falcon-Heavy, flown expendably. We will need a docking crew on-orbit for about a week max to assemble the docked cluster for each of the landers. This can be a crew of 2 to 4 in a Crew Dragon atop a Falcon-9. This probably will not happen in parallel for the 3 landers, but serially. So plan on 3 manned Falcon-9 launches to support these assemblies.   Getting the Transport to LEO, Loaded, and Assembled The orbit-to-orbit transport goes up as separate modules (without supplies) to be docked in orbit. There are six 20-ton modules and one 40-ton center modules, complete with solar wings that must unfold. All the listed boosters could launch the 20-ton modules, only Falcon-Heavy, "Starship", or SLS could launch the 40-ton module. The most cost-effective means was a tie: 2 flights of "Starship" or 3 flights (expendable) of Falcon-Heavy deliver these 7 modules to LEO. There is about 150 tons of supplies, crew suits, and crew personal equipment to deliver to the transport and load inside (152 exactly, per these admittedly-uncertain estimates).   This is separable into lots deliverable by any of the boosters listed. From a cost-effectiveness viewpoint, this was another tie: 2 flights of "Starship", or 3 expendable flights of Falcon-Heavy. This is going to require a temporary docking and loading crew of perhaps 4 to 6 astronauts for a week or so in orbit. If we send them up in two Crew Dragon capsules atop Falcon-9 boosters, they can come home in one, and leave the other Crew Dragon docked to the transport as one of its emergency return escape craft. Add 2 Falcon-9 flights for the transport assembly crew unless "Starship" is used instead.Getting the SEP "Tugs" to LEO and Fueled The SEP "tug" hardware, empty of the solid iodine fuel, are not heavy at all. This crude estimate says they are 14.42 tons each, and there are 3 of them. That includes the folded solar panels, the big thruster array, the guidance package, and the empty tank which doubles as the vehicle core structure, about which dead-head payload gets docked. Any of the listed boosters can get an empty tug to LEO. The most cost-effective means is 3 Falcon-9 launches, possibly flown recoverable, but the expendable price was used here. The iodine thruster fuel is a sublimable solid, which can be sent up in portions that fit the various boosters, determining the number of flights. For the three tugs together, we need 3213.54 metric tons of iodine sent to LEO. (Most of this, by far, gets used sending payload to Mars. Only a few tons with only 1 thruster firing is needed to return to Earth.) Any of the listed boosters can do this job. The most cost-effective means is by "Starship", with Falcon-Heavy a close second. That would be 33 "Starship" flights, or 54 Falcon-Heavy flights flown expendably. It will take a crew of 4-6 astronauts to load the iodine fuel and unfold the solar arrays, plus some checkout. We probably do not do all 3 vehicles in parallel, but serially. If by "Starship", that vehicle can carry the crew. If by Falcon-Heavy, a separate Falcon-9 launch is needed to send this crew up for a week or two in orbit as the payloads arrive, which is a huge Falcon-Heavy flight rate! "Starship" with payload and loading crew aboard is thus the preferred way, by far. Getting the Earth Departure Stage to LEO and Fueled This is assumed an empty stage delivered as one piece of hardware at 47 metric tons, plus 796.2 metric tons of LOX-LH2 propellants delivered as bulk liquid. Bulk liquids can be delivered in multiple payloads by any of the listed boosters, but requires special tankage and a human crew to do the transfers. The most cost effective way to deliver the empty stage is by a single Falcon-Heavy, possibly flown recoverably, but priced expendably for this analysis. The most cost-effective means to deliver bulk propellant is 8 "Starship" flights, followed fairly closely by 14 Falcon-Heavy flights. These require crews, which can be aboard the "Starship" flights. They would have to come up in some 14 Falcon-9 launches with Crew Dragon if Falcon-Heavies were the propellant ferries. By far, the preferred approach is 8 crewed "Starship" flights.Getting the Crew Onto the Transport for the Mission The Mars mission crew is only 6 people. This is one Falcon-9 Crew Dragon flight to send them up. Their Crew Dragon docks with the transport to be its second (and redundant) emergency escape capsule. If not covered earlier, make this 2 flights so there are two Crew Dragons as escape capsules.Totaling Up Mission Launch Requirements & Guessing Costs I totaled-up the launch costs for this mission. On the assumption that launch costs are 20% of overall program costs, that puts this mission in a rather modest cost category, despite the large tonnages. That is precisely because it does NOT use SLS to launch anything, at a billion dollars per flight (if not more)! See Figure 26 for a summary of the launch requirements and costs. The basis for comparison is the infamous "90 Day Report", based on mounting essentially "Apollo-on-steroids-plus" as executed by the long-favored contractors, to send a crew of 4-to-6 to one site on Mars, in the one trip. Figure 26 – Rough-Guessed Costs From Estimated Launch Requirements click for larger image This makes the comparison to the "90 Day Report" even more stark.This mission as planned has a "way out" or a self-rescue capability at every step,plus inherently designed-in artificial gravity and radiation protection (to include solar flare events). The likelihood of this crew returning alive and healthy is actually quite high.In comparison,with the "90 Day Report" mission,that likelihood is rather low,because it does not offer those characteristics. What this mission accomplishes is up to 6 different sites explored in the one manned trip to Mars.With the "90 Day Report" mission design,only one site gets explored. This mission leaves considerable usable assets at Mars for future missions to utilize.That would include the reusable landers,either in low Mars orbit,or on the surface if landed robotically.Plus, there might be some leftover propellant,probably in Mars orbit.The "90 Day Report" mission leaves few (if any) usable assets on Mars for future missions to utilize:maybe a surface habitat structure and a rover or two,and possibly a nuclear power supply item. See Figure 27 for a listing of what this mission accomplishes,compared to that of the "90 Day Report".Figure 27 – Mission Accomplishments and Characteristics Summary and Comparison click for larger image The first gross indicator is program cost for the one trip to Mars, divided by the number of sites explored while the mission is there. For my mission design, cost per site ranges from $11.7B/site to at most $70.3B/site, depending upon whether the minimum 1 or maximum 6 sites get explored. That is factor 6.4 to 38.5 times better cost per site than that of the "90 Day Report". The second gross indicator is the likelihood of getting the crew back alive and healthy. Because of the features demanded by ethics, and designed-in from the start, this mission plan can truthfully claim a high likelihood of accomplishing this.   The "90 Day Report" mission plan cannot truthfully claim that. For one thing, there is no rescue for a crew stranded on Mars. For another, there is a high likelihood of a solar flare event during a 31 month mission, and almost zero chance of surviving that event with no radiation shelter. And yet another: there are two 9-month transits in zero-gee, separated by a 13 month stay on 38% gee Mars, with undetermined

therapeutic effect, if any. Should an emergency free return at Earth arrival be required, that is a high-gee event (likely 10+ gees). A crew weakened by microgravity diseases is unlikely to survive this. Now remember, spaceflight history clearly demonstrates that there is nothing as expensive (economically and politically) as a dead crew. Especially one dead from a bad management decision. My mission design raises crew survival probability, the “90 Day Report” mission design does not; that survival probability is quite low, if one is truthful about it. In order to get both high “bang for the buck” and a high likelihood of getting a crew back healthy, I had to think way outside the usual boxes. One of those boxes is “nothing can look much different than what we already did during Apollo, shuttle, and ISS”. Another is “no mission can be affordable if there must be a high tonnage launched”. A third is “you simply must do direct entry at Mars to save launched tonnage”. A fourth is “you must use SLS no matter what in order to launch this mission”. All proved to be false constraints on thinking. The only one that is true is the one I used: crew survivability above all, driven by basic ethics. In a nutshell: “provide a way out or a self-rescue capability at every single step”. That drove me to orbital-based exploration and a manned orbit-to-orbit transport design. The main possible weakness of my mission design is the low payload fraction of my one-stage reusable landers: around 2%. A one-shot two-stage design would have a far higher payload fraction (perhaps 5-6% if you include the safety-required abort capsule, only higher if you fail this safety requirement), resulting in a smaller mass sent to Mars for each lander. But I would have to send more of them (8) to maintain a rescue capability and a spare, and still visit as many as 6 sites. This I leave to others to explore.Final Comments In terms of both cost and safety, the comparison of this mission plan to that of the “90 Day Report” demonstrates the unattractiveness of the usual way NASA did things in the past. There is far more “bang for the buck” and an enormously-higher probability of getting the crew back alive and healthy in my plan. Not only that, my program cost is far, far lower. The astute reader will observe that I have selected a lot of Spacex hardware as the most cost-effective means to launch and assemble this mission. That begs a comparison to the Spacex plan just to send multiple “Starships” to Mars by direct entry from the interplanetary trajectory. According to the presentations released, that would be about 6 “Starships” initially landed on Mars, with probably one or at most two of them eventually returning to Earth, if the local propellant production works, and it can supply them fully and quickly enough. It is as yet unclear whether 5 or 6 “Starship” tanker flights are required for refueling each interplanetary “Starship” in LEO for the journey to Mars. So somewhere between 36 and 42 total “Starship” launches are required to support their mission. Using $150M per launch, and launch costs equal 20% of program cost, that’s $5.4-6.3B launch cost, and $27.0-31.5B program cost, to put their mission onto Mars. That program cost scaleup is real, even for them, because they are counting on others to supply the local propellant production hardware, local rover vehicle capabilities, and local life support capabilities (cannot live in the landed “Starship” forever !!), not to mention local electric power. They have their hands full just developing the ““Starship” vehicle. That’s comparable to my costs, and (like me) way below the costs in the “90 Day Report”. The differences are many, however. They explore only 1 site, period. If the local propellant production fails to meet expectations, nobody comes home. They say they will supply radiation sheltering, but not artificial gravity. They are counting on Mars’s 0.38 gee being “therapeutic enough”, when in point of fact, nobody yet knows that to be true. My mission takes none of those risks and explores up to 6 sites. There is no aborting or bailing-out during the “Starship” direct entry at Mars. There is no aborting or bailing out during the landing and touchdown. They have yet to address soil bearing loads vs landing pad size for Mars, or rough field landing hazards such as slope, local roughness and big boulders. There is no bailout or abort during the return ascent. There is no bailout or abort for the direct entry at Earth return. There is no bailout or abort during the Earth landing and touchdown. A failure during any one of these events is inevitably a loss of the vehicle and everybody aboard. My mission takes none of those risks. Yeah, you can save the money using “Starship” as the transit vehicle (by about a factor of 2-3 over my plan). But you are also very much more likely to kill one of your crews if you do (which also very likely would put a stop to the ongoing mission). Ethics-driven spaceflight design “from the get-go” seems the more prudent course, especially when you consider the consequences of killing a crew. References #1. NASA radiation website titled Spaceflight Radiation Health Program at JSC (no cited reference newer than 1992). #2. From Bigelow Aerospace website as of 3-7-17

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