Low-Cost Propellant Launch to LEO from a Tethered Balloon – 'Propulsion Depots' Not 'Propellant Depots'

Brian H. Wilcox, Evan G. Schneider, David A. Vaughan, Jeffrey L. Hall, and Chi Yau (Tony) Yu
Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, CA 91109
818-354-4625
brian.h.wilcox @ jpl.nasa.gov

Abstract—As we have previously reported [1-4], it may be possible to launch payloads into low-Earth orbit (LEO) at a per-kilogram cost that is one to two orders of magnitude lower than current launch systems, using only a relatively small capital investment (comparable to a single large present-day launch). [1] [2] An attractive payload would be large quantities of high-performance chemical rocket propellant (e.g. Liquid Oxygen/Liquid Hydrogen (LO2/LH2)) that would greatly facilitate, if not enable, extensive exploration of the moon, Mars, and beyond.

The concept is to use small, mass-produced, two-stage, LO2/LH2, pressure-fed rockets (e.g., without pumps or other complex mechanisms). These small rockets can reach orbit with modest atmospheric drag losses because they are launched from very high altitude (e.g., 22 km). They would reach this altitude by being winched up a tether to a balloon that would be permanently stationed there. The drag losses on a rocket are strongly related to the ratio of the rocket launch mass to the mass of the atmospheric column that is displaced as the vehicle ascends from launch to orbit. By reducing the mass of this atmospheric column to a few percent of what it would be if launched from sea level, the mass of the rocket could be proportionately reduced while maintaining drag loss at an acceptably small level. The system concept is that one or more small rockets would be launched to rendezvous on every orbit of a propellant depot in LEO. There is only one orbital plane where a depot would pass over the launch site on every orbit – the equator. Fortunately, the U.S. has two small islands virtually on the equator in the mid-Pacific (Baker and Jarvis Islands). Launching one on every orbit, approximately 5,500 rockets would be launched every year, which is a manufacturing rate that would allow significantly reduced manufacturing costs, especially when combined with multiyear production contracts, giving a projected propellant cost in LEO of $400/kg or less. This paper provides new analysis and discussion of a configuration for the payload modules to eliminate the need for propellant transfer on-orbit. Instead of being a "propellant depot", they constitute a "propulsion depot", where propulsion modules would be available, to be discarded after use. The key observation here is that the only way cryo-propellant can get to orbit is by already being in a tank with a rocket engine. So careful system engineering could ensure that that same tank and engine would be useful to provide the needed rocket impulse for the final application. Long "arms" of these propulsion modules, docked side-by-side, could boost large payloads out of LEO for relatively low-cost human exploration of the solar system, for example.

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1. INTRODUCTION

Many mission studies of alternative approaches for extending human reach into the solar system have shown that the lofting of propellant mass into Earth orbit would be a dominant cost of any such effort [5]. The Saturn V moon rocket was typical of exploration missions in that 85% of the mass put into Earth orbit was propellant, as needed to leave Earth orbit, enter lunar orbit, land on the moon, return to lunar orbit, and depart back to Earth. Reducing the cost of lofting that propellant is key to the affordability of any sustainable exploration architecture, at least until the infrastructure is so advanced that extraterrestrial resources could provide the needed propellant at a lower effective cost. With current launch costs of ~$10,000 for every kilogram that is delivered to LEO, and with all the exploration architectures that are capable of captivating the imagination and support of the public and the Congress requiring one-to-several thousand tons of propellant mass to be lofted into LEO each year, the cost of propellant launch could easily consume the majority of the NASA budget (currently ~19 billion dollars (G$) per year). The pace of space exploration is almost completely limited by the rate at which propellant mass can be launched into LEO, since advanced exploration architectures envision reusable vehicles that could make multiple round-trips beyond LEO

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so long as they have sufficient propellant. Thus any affordable and sustainable exploration strategy would involve reducing the launch cost for propellant by a large factor.

Reducing the cost of launch into orbit has been extensively studied [6]. Unfortunately, credible proposals to achieve significant reductions in operational costs would have very large up-front investment costs. Such systems would include various combinations of new large reusable chemical rocket stages (with or without air-breathing 1st stages and/or runway launch and/or recovery), high speed "guns" that fire payloads through the atmosphere, and orbital towers or tethers that could be used as elevators to space. Advocates typically maintain that reduction of launch cost by a factor of 3 would require a capital investment at least equal to one year of the total NASA budget [6]. This is almost certainly unaffordable and unrealistic. This paper expands on previous analysis [1-4] to study a means to reduce propellant launch costs by a factor of about 30, using a capital investment that would be small (<<1G$) and that would be demonstrable within a few years. Furthermore, in the "propellant-rich" architecture envisioned here, all other space hardware could be made lower-performance (e.g., somewhat more massive but much lower cost per kilogram), and especially **more reusable** than it otherwise would be, since there would be plenty of propellant available to move it back-and-forth beyond LEO. Thus this approach may potentially reduce the overall cost of exploration by an order of magnitude or more when reuse of dry hardware allows 95-99% of the total mass put into LEO to be low-cost propellant.

As this paper describes concepts that are not yet approved as missions by NASA, all topics covered here are "for planning and discussion purposes only."

2. Summary of Previously-Reported Results

Think of the tethered balloon as a "flagpole" (Figure 1). The balloon supports a pulley that has the tether looped over it, and winches at the surface could lift a rocket launcher up to the stationary balloon, much as a flag is hoisted up a flagpole. A reloaded launcher goes up to the balloon from one winch as an empty launcher is lowered from the balloon to the other winch. By physically separating the two winches on the surface, the tether lines would not get tangled and the rocket could be dropped freely between the two tether lines for a few seconds before firing so that neither the balloon nor the tether are put at risk of being incinerated by the rocket. The flagpole architecture has the disadvantage that the tension in the tether acts on both sides of the pulley at the balloon, thereby doubling the required lift of the balloon. A superior alternative (described in detail in [2]) is to have a self-powered cable-car that winches itself up a single "large" tether (about 3 cm diameter), and then have a "small" tether to provide the required geometry at launch, and which also carries aircraft warnings such as radar reflectors, strobe lights, etc.

Launching from the equator, the fuel depot would pass over the launch site on every orbit – a key to the volume manufacturing approach. Any non-equatorial launch site would pass through the orbital plane of a fuel depot only twice each day, but even then it would be rare for the fuel depot to happen to be passing over the launch site at those times as needed for direct rendezvous. Although ship-launch from the equator would be possible, it is fortunate that the U.S. has two territories south and southwest of Hawaii that are within a few kilometers of the equator: Baker and Jarvis Islands. This would allow the balloon to be tethered to buoys anchored to the shallow ocean bottom, and allow the resupply ship to conduct operations in the prevailing wind and ocean current "lee wake" of the island. A special benefit of launching from the equator is that it lies in the inter-tropical convergence zone, where powerful winds are almost non-existent. Indeed, because there are no Coriolis forces at the equator, there is no tendency for unstable air to organize into cyclones. Hurricanes do not occur on or cross the equator [7] and there are no jet streams [8], both of which make tethering a balloon at high altitude on the equator much easier than at other places. Another advantage of launching from an extremely remote location in the Pacific Ocean, far from normal shipping lanes, is that the most economical system would use rockets that are less than 100% reliable. The lowest-cost overall system might well have ~10% launch failure rate. "3-sigma" or greater reliability would not be needed, greatly reducing the overall

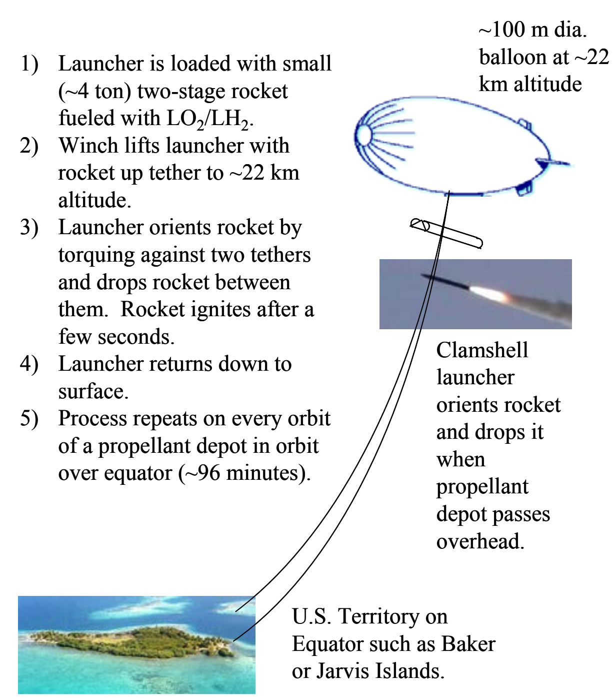

Figure 1: Concept for Low-Cost Propellant Launch to LEO from a Tethered Balloon

cost.

In [2] we showed, based on "textbook analysis," that a small LO_2/LH_2 pressure-fed rocket could be manufactured that would be capable of delivering ~200 kg of LO_2/LH_2 into LEO and that would have reasonable performance and expected cost. In this case, the textbook is *Modern Engineering for Design of Liquid-Propellant Rocket Engines*, by Dieter K. Huzel and David H. Huang; 1992 [9]. The expected manufacturing cost was minimized based on the assumed relative cost per kilogram of the pressurized propellant tanks and the rocket engines. We showed that the cost function has a broad optimum at an operating thrust chamber pressure of about 2 MPa (300 PSI), over a wide range of assumed thrust chamber material costs and chamber pressures. We concluded that an approximately cost-optimal second-stage engine could be built that has a specific impulse of 441 s, 41kN of thrust, a nozzle throat diameter of 11.6 cm, a nozzle exit diameter of 73.1 cm, and consumes about 9.5 kg/s of propellant. We speculated that the lowest-cost manufacturing approach would be to use the well-known "channel-wall" thrust chamber design [9, p96], where the entire thrust chamber, including throat and nozzle, is cast as a monolithic thick-walled copper-alloy structure, with many channels then machined into the outside of the casting to form passages for the cryogenic hydrogen that is both coolant and fuel. A structural backing would support the channel-wall liner, but that backing would never see high temperatures and so could be made out of inexpensive materials. Propellant pressure would be maintained by routing the coolant from the channels into a heat exchanger in the hydrogen tank, where a small fraction of its thermal energy would be used to boil the hydrogen. The hydrogen coolant would then be injected into the engine as fuel. The hydrogen gas boiled by the heat exchanger would pressurize both the hydrogen tank and the oxygen tank, with a relief valve to prevent over-pressurization. The tank pressure is assumed to be 20% higher than the thrust chamber pressure to ensure stable injection, following the guidelines given in [9, p115].

The production cost in volume manufacturing is estimated based on the experience of the automotive industry. The "learning curve" data that we used in [2] is summarized in Figure 2 [10]. A curve-fit to this data follows the traditional learning curve with free parameter 0.75 (e.g. the cost of unit 2^N is $(0.75)^N$ times the cost of the first unit). Perhaps the most important part of the volume-manufacturing approach is the "LOX Post" co-axial fuel injector, shown in Figure 3. The optimal injector design dimensions, derived in [2], are only a function of chamber pressure and so are common to both the first and second stages. As a result, this particular component would be needed in quantities of literally millions per year. As can be seen in Figure 2, even for complex products such as automobiles, the manufacturing cost per kilogram is only asymptotically greater than the bulk materials cost at these high production rates. This fuel injector is probably the most complex element of the launch

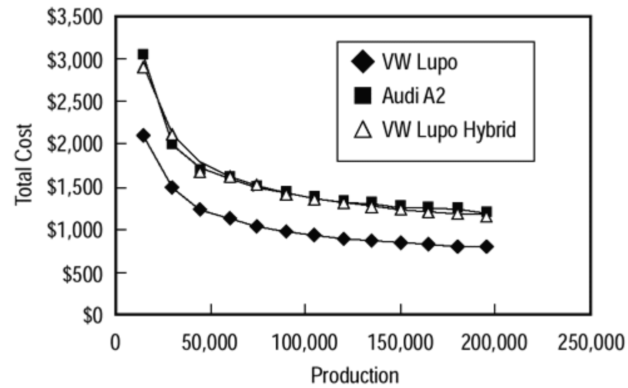

Figure 2: Manufacturing cost of automobiles in mass production (originally published in 2001 as Figure 7 in [10], reprinted with permission of The Minerals, Metals & Materials Society).

system, so it is fortunate that it would be manufactured in such large volumes.

We assumed in [1,2] that the rocket is controlled by two small head-end "vernier" rocket engines that are attached via gimbals to the payload assembly and would draw propellant from the payload, which is slightly oversized to account for this loss. The two main stages themselves would have no thrust-vector or throttle modulation control so as to keep their cost as low as possible. The first stage would ignite shortly after the rocket is dropped at the base of the balloon, with the second stage firing just after the first stage is spent. The rocket would be oriented so as to emerge quickly from the atmosphere and coast to the desired LEO orbital altitude, where the vernier thrusters complete the orbit injection. The discarded first stage would drop into the Pacific Ocean within a few hundred km of the launch site; the spent second stage almost reaches orbital velocity and burns up over the ocean after a partial orbit. The vernier control rockets would be used to complete the orbit injection of the payload and to accomplish rendezvous with the propellant depot as described in [1]. A relatively small amount of extra propellant would be carried by the payload module at launch

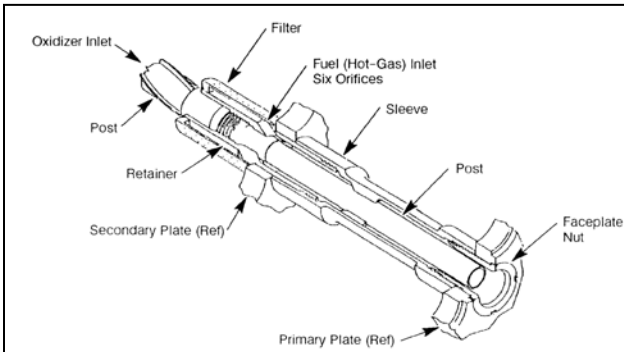

Figure 3: LO_2/LH_2 fuel injector, which would be manufactured in quantities of ~1.4M/y under this concept (originally published as Figure 4-61 in [9]; reprinted with permission of the American Institute of Aeronautics and Astronautics, Inc.).

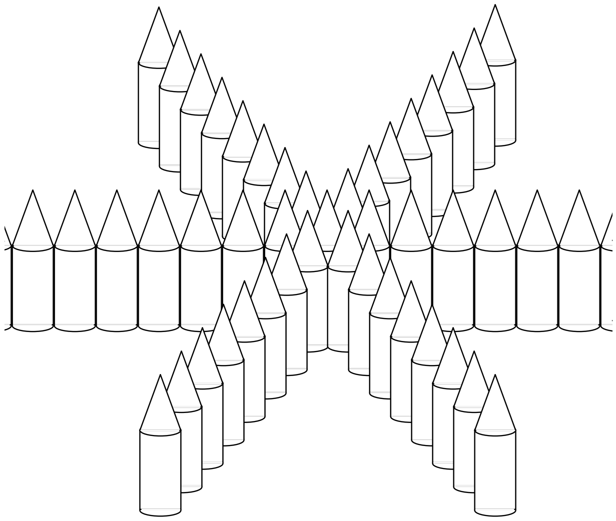

Figure 4: The Star Propellant Depot – a spin-stabilized, self-propelled array of small propellant modules.

to provide for these maneuvers. Only the nominal control propellant would be provided – off-nominal consumption would reduce the payload delivered to orbit. The cable-car/rocket-launcher would insulate the rocket to reduce boiloff (beyond that needed to bring the tanks up to working pressure) while the assembly is being winched up the tether; the tanks might also be topped off by the launcher just before it is dropped.

As described in [1], we envision the only permanent part of the propellant depot to be a robot that images arriving propellant modules (using visible light and thermal infrared radiation) so as to give radio commands for vernier thrusting to accomplish a precise rendezvous. On final approach, the robot would grapple the incoming propellant module, and secure it by clips or similar means to extend a row of identical propellant modules, forming an "arm" of the "star" propellant depot (Figure 4). The star depot, once complete, would be spun using synchronized thrusting of the many gimbaled vernier engines in the propellant modules, and accelerated by similar thrusting to transport itself to an inclined orbit, to the Earth-moon L1 libration point, to low lunar orbit, etc. With an assumed dry mass of 35 kg for each propellant module, and an assumed specific impulse of 420 s for each of the vernier thrusters, then of the propellant that would arrive in equatorial LEO, about 37% of the modules would arrive fully-fueled after transit to L1, or 29% would arrive after performing a propulsive plane-change maneuver to reach the 28.5^o LEO inclination required to rendezvous with hardware launched from the Kennedy Space Center in Florida (the spent modules would be discarded). From the L1 point (where the thermal equilibrium temperature allows indefinite storage of LH_2), it could be used to fuel or refuel vehicles making round-trips to the lunar surface or to Mars or beyond. To minimize propellant boil-off, star depots would be only delivered to the 28.5^o Earth orbit or to low lunar orbit as needed to promptly rendezvous with and refuel vehicles that would already be there. It is assumed that one propellant module would arrive in LEO every 96 minutes (the 90 minute orbital period, plus 6 minutes to catch up with the launch site moving with the rotating Earth). It is also assumed that one star depot would depart equatorial LEO every 2 weeks (at the launch windows to L1), implying that each arm of the star has 36 modules. This means that a star depot could deliver about 16 metric tons of propellant to L1 or 12.5 tons to the 28.5^o LEO orbit inclination every 2 weeks. All star depots that stay in LEO would be eventually de-orbited over the ocean. Propellant would be reserved so that all star depots leaving the Earth's gravity well (and the refueled Earth-departure stages) could eventually be mothballed to a stable high-Earth-orbit to reduce orbital and reentry debris and for use as a long-term resource. This would also avoid polluting the moon with the volatiles that escape from crashed vehicles, and the risk those impacts and their ejecta pose to surface astronauts and assets.

Each star would be formed around a "hub" module which is of a slightly different configuration than all the rest of the propellant modules. It would have six sets of clips around its circumference so that the six radial arms could be connected to it. It would have the same vernier thrusters and GPS-augmented inertial navigation and radio-commandable control system as the other modules so that it could be launched in the same way. However, it would have smaller propellant tanks so that the remaining mass and volume could be devoted to additional computing, communications and navigation equipment (e.g., a star tracker) so that it could function as the command computer for the depot, issuing thrust commands to individual propellant modules by radio. Further analysis may indicate that capillary forces are sufficient to collect the liquids in the relatively large surface-to-volume-ratio tanks of the small propellant modules, eliminating the need to spin the star depot, simplifying the plumbing and operations.

The propellant depot robot connects the modules together as they arrive at the star. The drains for liquid in each tank would be arranged so that either axial or rotational acceleration delivers the liquid to the drain. Similar to the LO_2/LH_2 upper stage engine on the Saturn V launch vehicle, a small acceleration may be required to keep the fuel settled at the drain ports to prevent ingestion of vapor when the engines are restarted. Prior to rotation of the star, and again similar to the Saturn V upper stage, this acceleration would be provided by directing the boil-off vents to the rear [11]. Each propellant module would have a small solar array, battery and electronics module at the rear (between the vernier thrusters) that would provide long-term power and command interface – forward of that is a thermal shield with foil wings that pop out at the time of payload separation to ensure that, when the back end of the star depot is pointed toward the sun, the heat load into the cryogenic payload would be minimized while the avionics stay within a reasonable temperature range. The payload tanks would be extremely well insulated, and appropriate surface coatings on the payload are optimized for indefinite propellant storage at L1.

The economics of manufacturing the rocket suggest that the cost would be low. The dry mass of stages 1 and 2 combined is about the same as the smallest car, and most of that mass is made of relatively thick aluminum panels that are easily rolled or stamped and welded together. Aerospace aluminum alloys typically command a 20% price premium over basic aluminum, but bulk aluminum only costs about $3.50/kg [10]. Nothing in stages 1 or 2 is remotely as complex as an automobile engine, transmission or dashboard. In our case, there are only a relatively few components in stages 1 and 2. The only moving parts in stages 1 and 2 are motorized ball valves that need to actuate only once. Based on manufacturing data such as that plotted in Figure 2, it seems quite possible that the manufacturing cost of stages 1 and 2 (combined) could be less than $10,000 - perhaps very much less. The current cost of liquid hydrogen and liquid oxygen, purchased in bulk, is under $2 /kg. So the total cost of the wet vehicle, less payload, is expected to be under $13,000.

The payload assembly, however, would have a relatively high specific cost. As discussed in [2], it is probably best to make this a highly-reliable piece of spacecraft-quality hardware. It represents the "brain" of the launch vehicle, having the inertial measurement unit and the avionics that trigger ignition and staging events, and control the vernier thrusters. Weight and reliability improvements on the payload assembly would pay handsome dividends in the overall system because the payloads have to change orbits while preserving as much propellant as possible and would be expected to operate for many weeks or months, performing on command. So it seems reasonable that the assumed 35 kg of dry mass in the payload assembly might have a specific cost of $1,000/kg, even manufactured in annual production volumes of 5,500. So we estimate that the completed rocket costs 13k$ for stages 1 and 2 (wet) and 35k$ for the payload assembly, for a total of about 50k$ per launch.

Launch operations would be managed by a medium-sized cargo-container-type ship that could be leased for 100k$/day (including crew, estimated based on quotations for a large oceanographic research vessel for a preliminary balloon deployment experiment at Baker Island [12]). Two such ships would be needed, one on-station and one going back and forth to port, loading in port, and providing shore leave for the crew. Launching every 96 minutes, the total cost of the rockets would be 750k$/day. So the total cost of ships and rockets would be about 350M$/y. If lithium-ion batteries with a life of 200 cycles are used to power the cable-car that runs up and down the tether, then even at current laptop battery prices that would only be 20M$/y. Even if the balloon and tether needed to be replaced every 100 days, that would only add a few M$ to the total cost, based on the cost of the NASA Ultra-Long Duration Balloon and the catalog price of the advanced rope that would be used (e.g., PBO). Replacing every 10 years a 200M$ on-orbit robot that manages the arrival and rendezvous of the payloads (as described in [1]) would add only 20M$/y to the system cost. Thus we expect the total system cost to be about 430M$/y, while launching 1100 tons of propellant into orbit, for a specific launch cost of 390$/kg (Figure 5). This is a reduction by a factor of almost 30 compared to current launch services. Note that the total annual system cost could be about the same as only a few present-day expendable launches, or less than the cost of a single projected heavy-lift launch vehicle. As discussed in [1], the system could be expanded by launching more than one rocket on every orbit of the propellant depot, increasing the amount of propellant delivered in integer multiples while further driving down the per-kilogram costs via mass production [10]. Perhaps most importantly, these savings could be achieved with very low capital investment. Certainly a vendor who configures a factory to manufacture the rockets or the payload assemblies would need to be assured of a multi-year contract with an appropriate early-termination clause, but given that, there is no particular reason that the government should make a large up-front investment, or to contract on anything but a fixed-price basis. Presumably the government would have to pay to develop proof-of-principle rocket/payload prototypes. One obvious procurement strategy is to get multiple prototypes built by competing prospective full-production bidders in a "shoot-out".

Figure 6 shows the proposed launch configuration. The overall length of the vehicle at launch would be about 12 m (fitting in a standard 40' shipping container). As described in [3] we derived detailed design parameters for the required rocket engines for the 2-stage launch vehicle. We adopted that 2nd stage engine design as a baseline, and considered the

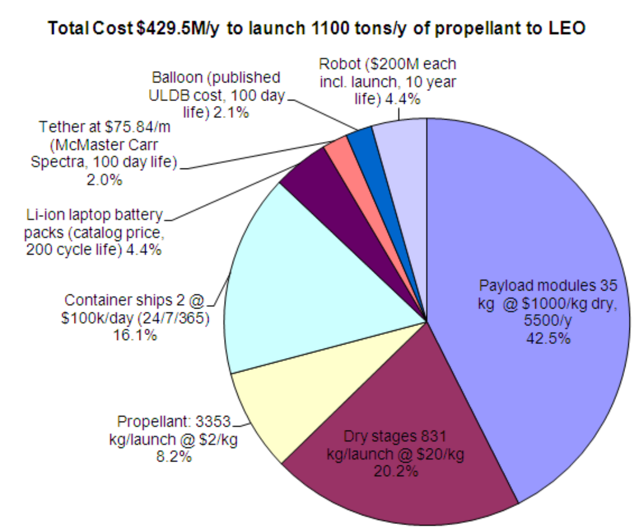

Figure 5: Estimated cost of propellant launch from a tethered balloon - $390/kg.

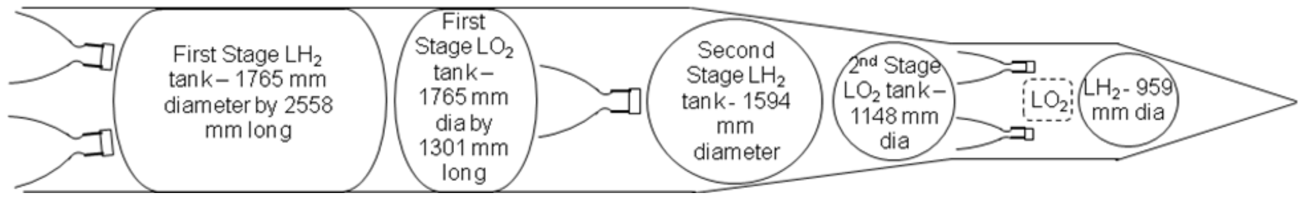

Figure 6: Proposed Launch configuration: overall length is ~12 m, 1.8 m diameter, 4439 kg at launch; both first stage tanks have wall thickness 6.9 mm; second stage hydrogen tank wall thickness is 2.5 mm and oxygen tank thickness 1.3 mm; payload assembly hydrogen tank wall thickness is 0.9 mm and oxygen tank thickness 0.5 mm. There would be a single 41 kN engine for the second stage with cluster of 4 identical engines for the first stage.

case where the first stage consists of a cluster of four engines identical to the single 2nd stage engine. Each of the four would be canted so that their thrust vector passes through the nominal center-of-mass of the launch stack, so that variations in performance between the engines wouldn't have an undue effect on the control authority requirements. The manufacturing volume of this engine would be almost 30,000 per year, making relevant the use of the manufacturing cost data shown in Figure 2. The only other somewhat complex components (beyond the fuel injector described previously) needed for each engine would be precision, highly-polished cryogenic ball valves (although low-leakage would not be particularly crucial on the launch stages because the propellant would be in the stage for a very short period). Again the high production volume should make the cost similar to that of modern automotive components having similar precision and reliability.

It is presumed that there are large cut-outs in the skin adjacent to the two head-end vernier engines so that they could gimbal out at possibly large radial and tangential angles to provide transverse thrust and roll control during ascent. Fortunately, the low atmospheric density and dynamic pressures associated with launch from high altitude would make this strategy viable.

Spherical tanks would be used in the second stage and the payload assembly to reduce mass. The tank mass per unit volume for a spherical tank is only 75% that of a long cylindrical tank, at any given tank pressure and wall stress. Cylindrical tanks would be used in the first stage at the same diameter as the cluster of four engine nozzles to minimize the frontal area and hence aerodynamic drag. The tank mass in the payload would only be about 9.5 kg of the allocated 35 kg for the dry payload assembly. The low mass of the spherical tanks means that the thrust chamber pressure of the vernier thrusters could be the as high as the other engines (2 MPa), so all fuel injectors are identical and the LO_2/LH_2 injection velocity ratio would not be a concern, as discussed in [9, p116].

The results of a simple spreadsheet-based dynamic simulation of the launch are shown in Figure 7. Aerodynamic lift and drag are modeled (as was done in [1]) using published wind-tunnel data for a cylindrical body with a sharp conical nose at Mach 3.12 and Reynold's numbers of 8×10^6 and 14×10^6 [13]. For the current model, a $\sin(2\alpha)$ function of angle of attack α was fit to the lift-to-drag ratio of the published measurements, giving multiplicative coefficient 17.96 (e.g. the lift-to-drag ratio is $17.96 \sin(2\alpha)$). The drag coefficient is similarly fit to the data as $1-0.8\cos(2\alpha)$. Note that the vehicle speed would be very low except when the angle of attack is small (e.g., <0.1 radians), so this simple model (based only on frontal area) should be adequate. The Mach number of the published data reflects a speed of about 1 km/s, which roughly corresponds to the peaks of lift and drag in Figure 7. The Reynolds numbers for the published data are also in rough agreement with the flight conditions near the peaks of lift and drag. Atmospheric density is modeled as an exponential with scale height 7.3 km, based on a sea-level density of 1.293 kg/m^3. This gives reasonable agreement with empirical data over the regime of interest represented in Figure 7.

The initial conditions of the launch simulation results shown

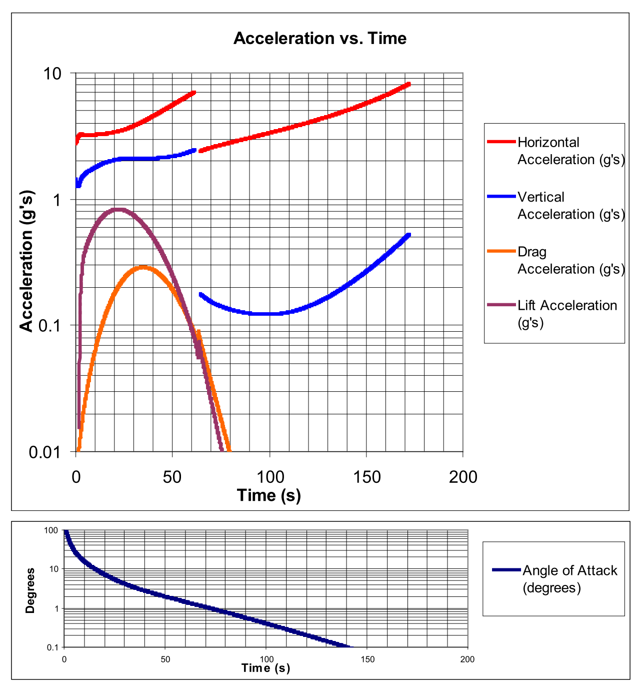

Figure 7: Acceleration and angle of attack versus time during launch. Lift and drag accelerations are expressed in air-relative-velocity-centered coordinates (not gravity-relative) but their resolved effects are included in the horizontal and vertical acceleration plots.

in Figure 7 are that the rocket is dropped from an altitude of 22 km 6 seconds before ignition, at which time it has an elevation of 40.2 degrees above the horizontal. (The launch altitude is optimized based on wind speed data [14] as discussed in [2].) A GPS-augmented inertial navigation system is assumed to command the vernier thrusters to control the angle-of-attack to follow the lower curve in Fig. 7, aligning the vehicle axis with a ballistic trajectory after an exponential decay. (This control law is representative but not optimized.) The launch mass of the rocket is 4439 kg. The first stage consumes 2,329 kg of propellant in 61 seconds, accelerating the vehicle to a horizontal velocity of 2,587 m/s and a vertical velocity of 1,157 m/s at a point 68 km downrange and an altitude of 53 km. The inert mass of the first stage (582 kg) is jettisoned, and the second stage ignites a few seconds later. The second stage consumes 1,024 kg of propellant in 108 seconds, accelerating the vehicle to a horizontal velocity of 7,196 m/s and a vertical velocity of 1,339 m/s at a point 555 km downrange with an altitude of 187 km. The inert mass of the second stage (249 kg) is jettisoned, and the vehicle coasts for another 147 seconds up to an altitude of 278 km (150 nautical miles). When the horizontal velocity is added to the original speed of the launch site (463 m/s with respect to the center of the Earth) the inertial velocity is 7,659 m/s, compared to a required circular orbit velocity of 7,748 m/s. The vernier thrusters add the necessary 89 m/s of delta-V to circularize the orbit at 278 km altitude, and then to rendezvous with the propellant depot by radio command as described above. The drag and gravity losses of the first stage are 384 m/s. The drag and gravity losses of the second stage are 189 m/s. The spent second stage burns up over the ocean.

In [3] we concluded that, when aerodynamic torques are considered (not just thrust misalignment torques), the benefits of possible spin-stabilization disappear. Per the launch simulation, control propellant usage is minimized when the center of mass starts out about 29 cm behind the center of pressure (which could be pre-set by attaching small trim tabs to the structure). As the propellant from stage 1 is expelled, the center of mass moves forward. At 24 seconds after ignition, the center of mass passes through the center of pressure. By first stage burnout, the center of mass has moved 126 cm ahead of the center of pressure. The first peak in aerodynamic torque occurs 10 seconds after ignition at 4.8 kNm of torque. After the center of mass passes through the center of pressure, the second peak in aerodynamic torque occurs 42 seconds after ignition, with 5.0 kNm of torque (in the opposite direction). Without spin stabilization, about 13 kg of propellant would be expelled by the vernier thrusters to counter this torque over the 1st stage burn. We concluded in [3] that each vernier engine would need to deliver about 800 N of thrust. Since the two vernier thrusters would be located at least 3 meters forward of the center of mass during the first stage burn, the maximum counterbalancing torque that the vernier thrusters could exert is about 5 kNm, or about the same as would be required. To further reduce the propellant consumed for control during the aerodynamic portion of the flight, one possibility would be to affix small canard fins on the exterior of the nozzles of the vernier thrusters. We speculated in [3] that it might be good to attach small fins to the vernier rockets that protrude out of the side of the vehicle, deflecting the air stream and providing control authority without firing the engines. The mass impact of these small fins would be traded against the mass savings in tankage needed to carry the offset propellant.

We have assumed that each vernier engine has a specific impulse of 420 s, slightly lower than the main engines because of the somewhat adverse surface-to-volume ratio of the small-diameter thrust chamber. However, we assume that the design of the vernier engines is based on the same methodology given in [9]. Because it is a regenerative engine (e.g., the coolant for the thrust chamber is dumped back in as fuel), it would retain a relatively high specific impulse. To generate ~800 N of thrust would require a propellant flow rate of about 0.2 kg/s. As previously mentioned, the propellant injectors in the vernier engines are expected to be the same as those in the first and second stage, and to operate at the same chamber pressure. The 41 kN second stage engine derived in [2] has 52 injectors, meaning that each injector would account for about 790 N of thrust. So each vernier engine could achieve approximately the desired thrust and propellant flow rate using exactly one injector as shown in Figure 3.

Another issue addressed in [3] is possible ice build-up on the tether. The equatorial climate of the launch site has high humidity at low altitudes, but the tether spans the hot, humid conditions at the surface and the frigid, dry conditions at the balloon. There would be a transition zone where sustained ice buildup on the tether is possible. Unchecked, this could drag down the balloon. Fortunately, our concept involves a cable-car that runs both up and down the tether every 96 minutes. This cable-car would flex the tether around wheels, which would shatter any ice buildup. The cable-car could also carry special implements to remove excess ice. In our concept, there would also be a stationary, "lightweight" tether that carries aircraft warnings and also pulls the main tether off-vertical to facilitate safe rocket launch. This stationary tether might also need to have some means for clearing ice build-up. One possibility would be to have many small solar-powered cable cars that each carries an aircraft warning (strobe light and radar reflector). Each small cable-car would move up and down its assigned section of the tether, clearing the ice.

The significant aerodynamic forces due to winds at the launch altitude would necessitate use of a superpressurized balloon that could maintain the balloon shape and avoid fatigue-induced failure of balloon material that is flagging in the wind. There are two basic design options: a spherical balloon that would use a high strength film plus fabric laminate material, or a pumpkin balloon, which is a tendon reinforced polyethylene film balloon. Fabric plus film laminates are commonly used materials for blimps and aerostats, while the pumpkin balloon is being developed by NASA for ultra long duration scientific balloon flights at very high stratospheric altitudes (~36 km). At the present time, it is unclear which design option would be best for the rocket launch application. The spherical fabric balloon is likely to be more robust and have a longer lifetime, but with the probable disadvantages of being more expensive to build and more massive than a pumpkin balloon. Prototypes would need to be built and tested to quantify the cost, mass and lifetime issues, after which a final balloon design option could be selected.

Table 1 summarizes the mass and performance of each stage of the proposed small rockets.

In [1] we assumed that the "hub" of the "star" propellant depot (Figure 4) would include a cryocooler that reduces or eliminates boil-off of the cryogenic propellant stored in the depot. In [2] we presented refined analysis indicating that each payload module would need its own cryocooler for boil-off control, since the radiator area requirements would be too great for the hub module alone. Boil-off of the cryogenic payload while in LEO would be a major concern. As described in [2], the equilibrium temperature behind a sun-shield at the Earth-moon L1 libration point is only about 40K, so it should be possible to maintain the liquid hydrogen indefinitely at ~20K with a single-stage cryocooler operating between the hydrogen and oxygen tanks. The LO_2 tank, operating at ~89K, could be passively-cooled by radiating into space at the L1 point. But our operational concept calls for the star propellant depot to remain in LEO for about two weeks during its construction and prior to its self-propelled transit to L1. So boil-off during this two-week interval would be a major concern.

Once on-orbit, the payload assembly would partition into a "hot" assembly and a "cold" assembly. The hot assembly includes the solar array, batteries, electronics, and vernier engines. The cold assembly includes the cryogenic tanks. Separating the hot assembly (kept pointed always at the sun) and the cold assembly (pointed away from the sun) is a thermal barrier. This consists of multi-layer blanket material, at least some of which is "spring-loaded" to expand beyond the nominal skin of the rocket fairing so that it completely shadows the cold assembly from the sunlight. This expanded sunshield may deploy at the time of separation of the payload from the second stage - perhaps the second stage could pull off the aerodynamic skin of the payload as part of separation, revealing the thermal shield

ISP=441s	Dry mass (kg)	Propellant (kg)	Stack at ignition (kg)	Horizontal ΔV (m/s)	Drag/gravity/control losses (m/s)
Stage 1	582	2329	4439	2587	631
Stage 2	249	1024	1528	4609	189
Payload assembly	35	220	255	89	248
Launch site velocity				463	
Total	866	3573	4439	7748	

Table 1: Summary masses and delta-Vs

for spring deployment. It may be desirable to "fan-out" the foil elements so that they have extra space between layers so extra radiative cooling could occur out the sides of the blanket, similar to what is planned for the James Webb telescope sunshield. Within the cold assembly, the liquid hydrogen tank (at ~20K) needs to be thermally isolated from the liquid oxygen tank (at ~89K).

Reference [15] provides test data showing that multilayer insulating blankets leak heat between a cryogenic side and a room-temperature side at the rate of about $1W/m^2$. Assuming that stainless steel flex lines are used for the cryogenic fluids, reference [16] indicates that stainless steel has a thermal conductivity of about 2 W/m-K at 20K. A 25 mm diameter stainless steel flex line with 1 mm wall thickness has a metal cross-section of 80 mm^2, and so with a thermal gradient of 100K/m it leaks heat at only 0.016 W. The heat of vaporization of LH_2 is about 450kJ/kg. So if the thermal blanket around the hydrogen tank has an area of 5 m^2 and a leak rate of $1W/m^2$, and if all the stainless flex lines and mechanical linkages have a leak of 2 W, then the total heat leak into the tank is 7 W and the boiloff rate (without cryocooler) would be 0.056 kg per hour. This would boil off 19 kg in 14 days - about 65% of all the hydrogen. One possibility is to over-size the hydrogen tank and simply allow this much hydrogen to boil away. Since hydrogen is a small fraction of the total payload, this is not much of a mass penalty (~10% of payload mass).

It is clear that valves need to prevent the cryogenic liquids and gasses from transiting down the stainless steel flex lines between the cold and hot assemblies - otherwise our estimate of the heat leak of the flex lines would be far too low. This means that the valves, presumably highly-polished metallic ball valves, need to be on the "cold side" and mechanically actuated through linkages from the "hot side". This could be accomplished with cables or pushrods. Based on the thermal conduction of the metallic flex lines, we estimate that the heat leak through such mechanical linkages would be negligible.

Reference [17] describes an extremely detailed boil-off analysis of several interplanetary spacecraft, of which the proposed Mars Sample Return (MSR) Earth Return Vehicle (ERV) (configuration "M3") is most similar to our application. That vehicle would carry 5148 m/s of LH_2/LO_2 propellant, would be solar powered, and operate in the relatively "hot" environment of Mars orbit for a long period of time. They assume the Passive Orbit Disconnect Strut (PODS) technology, which would provide stiff load paths during launch that would be separated on-orbit to reduce the heat leak to "1/10th that of state-of-the-art struts," with the penalty of reducing the natural frequency by a factor of about 3 (the thermal conductivity is about proportional to the square of the frequency). Despite the fact that the proposed MSR ERV spacecraft described in this document would carry >10 times as much LH_2 as the vehicle considered here, analysis showed it only had a heat leak of

~0.8W during interplanetary cruise at 1 A.U, requiring about 300W of power from the main bus for cooling. In equatorial Mars orbit, where the conical shield axis could be aligned with the polar axis to continuously shield both from the sun and from Mars, the bus power requirement for cooling was estimated to be only about 400W. Since the radiative temperature of Mars is not that much different from Earth (at least compared to 20K) we can infer that it would be possible that we could maintain zero boil-off in LEO with substantially less than 400W of bus power devoted to a cryocooler. This suggests that a similar alignment between the conical shield axis and the polar axis of the Earth might be worth considering, despite the changes that would required to the mating of adjacent payload modules in forming the "star" and the necessity of gimbaling the solar arrays. Perhaps the solar array function could be combined with possible aerodynamic fins associated with the gimbaled vernier thrusters, as speculated earlier. Presumably the conical heat shield would be open toward the north star in the northern winter and be pointed oppositely in northern summer to optimize the view angles of the solar array.

The overall conclusion is that a cryocooler may not be necessary to manage boil-off in LEO, if the hydrogen tank could be somewhat oversized, but a small cryocooler is desirable anyway to manage boil-off for indefinite periods at L1, as described in [2]. It seems likely that this cryocooler, operating between the hydrogen and oxygen tanks, could perform a useful function during the loiter period in LEO, but this would not be absolutely required for the integrated system to be effective.

3. LONG ARMS OF PROPULSION MODULES

The net result of this architecture is that a large amount – thousands of tons – of high-performance propellant could be delivered to Earth orbit for a modest amount of money – one billion dollars or so. A slightly different propellant module configuration (Figure 8) from that envisioned in [1-4] (and refining the cartoon configuration of Figure 6) would allow the modules to be linked together into very long "arms". These long arms would have the advantage that they could be used to provide delta-V for payloads directly, without propellant transfer. Since propellant could not reach any orbit depot without being in a tank and without having a rocket engine, we conclude that with proper system engineering, no refuelable tanks or extra engines really need to be part of any depot or mission concept. These long arms are designed as trusses to withstand the large bending moments introduced when the propulsion modules on the ends of the arms are firing. In this concept, the modules are linked together with "S-hook" tension bands at the bottom and tubes at the top that transmit large compressive forces through the spherical ~1-meter diameter, 360 PSI hydrogen tanks. The compressive tubes could be docked together using permanent magnets and alignment cones to ensure they mate together securely. When modules at the end of a long arm fire, the bending moment would be transmitted to the payload by compression across the row of hydrogen tanks at the top and tension in the S-hook bands at the bottom. Diagonal compression is taken across the oxygen tanks, which would be arranged as long cylinders in pairs on either side and below the hydrogen tanks. A lightweight, strong, and thermally insulating tension band (e.g., kevlar, vectran) would connect at one of the S-hooks, go around a saddle at the top of one oxygen tank and around the back of the hydrogen tank, around another saddle on the oxygen tank on the other side, and down to the other S-hook. This would create a very stiff and strong truss using as compressive members only the existing, relatively heavy tanks with only lightweight, structurally-efficient, and thermally-insulating tensile structure (shown in red in the left schematic of Figure 8) connecting them. The thin tension band cable limits the heat leak between the hydrogen and oxygen tanks. Staging of spent modules at the ends of an arm would be accomplished by energizing an electromagnet around the permanent magnets that keep the hydrogen tanks together, momentarily causing the electromagnets to repel. The spent end of the arm would rotate slowly around the S-hook, until the S-hook passively disconnects after about 90 degrees of rotation to release the

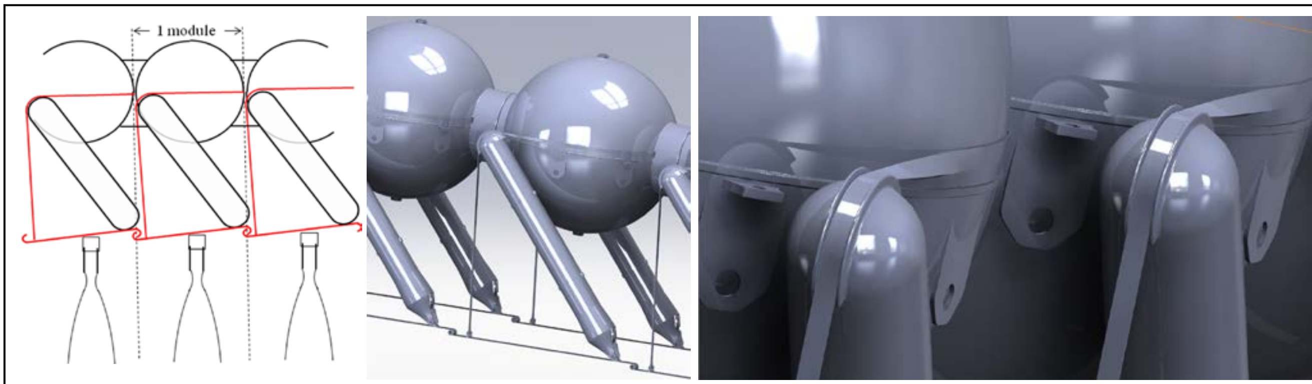

Figure 8: Details of mechanical truss elements of "long arms" of propulsion modules. Spherical hydrogen tanks support compression forces along top of long arm, while "S-hook" bands support tension along the bottom. Pairs of cylindrical oxygen tanks support compressive shear forces, with tension bands distributing force to back of hydrogen tanks. For clarity, other components in renderings such as gimbaled vernier rockets, solar arrays, sunshield, etc. are not shown.

spent modules after each burn. It would be operationally helpful if each module could autonomously dock with its neighbor in an inverse maneuver, eliminating the need for the "depot robot" at the original star depot hub in LEO. In that case, the hub module would contain the optical telescope needed to command the rendezvous, as well as a star tracker (perhaps using the same optics) needed to navigate to an Earth-moon libration point.

A possible criticism of this approach is that the propellant mass fraction of these modules would be relatively low, so that higher performance could be achieved if the propellant were transferred into a lighter-weight tank/engine assembly. This is true in principle, but the practical benefits of such an approach would not be significant. Figure 9 shows the results from the rocket equation, where rockets of different propellant mass fraction and specific impulse are used to leave LEO to achieve Earth escape velocity. On the horizontal axis is plotted the propellant mass fraction for the rocket. On the vertical axis is plotted the required Initial Mass in LEO (IMLEO) as a multiplier on the payload mass that achieves escape velocity (along with the spent rocket mass). Different curves are plotted for selected values of the specific impulse. For moderate specific impulse (e.g., 280 s, characteristic of solid rockets) the IMLEO becomes a prohibitively large multiplier on the final payload as the propellant mass fraction drops below ~85%. However, as higher-performance propellant is used, the effect of propellant mass fraction becomes much less pronounced. We are assuming cyrogenic hydrogen and oxygen, with our small vernier engines assumed to have an Isp of 420 s, with a propellant mass fraction of 85%. Note that, for high values of Isp (e.g. >400 s) that neither Isp nor PMF are particularly important in terms of IMLEO. There is no great benefit in attempting heroic measures to improve either the Isp (to 450 s, for example) or the PMF for these propulsion modules. However, the value of using liquid hydrogen as a propellant is clear, as compared to lower-performance fuels (e.g. Isp's of 350 s or lower). The fact that the proposed long arms of propulsion modules would have a high Isp (>400) and offer almost "continuous staging" opportunities more than outweighs the relatively low propellant mass fraction.

4. EXAMPLE: A HUMAN MARS MISSION

In our approach, the propellant would arrive on-orbit in tanks and with appropriate rocket engines affixed, so that no transfer to different tanks and/or rockets would be needed. Instead, "long-arm" clusters of these propulsion modules could be used to perform any needed in-space Delta-V. For example, they could be accumulated at the Earth-moon L1 libration point, and then used to accelerate a human Mars spacecraft out of L1 with a gravity-assist around the backside of the moon (using a delta-V of ~350 m/s), to perform a trans-Mars injection maneuver at the top of the Earth's atmosphere (~550 m/s, starting at ~escape velocity) and then to perform a "stop and drop" maneuver at Mars, where the vehicle would stop from hyperbolic approach at the top of the atmosphere (~5900 m/s) and perform a subsonic "drop" through the Mars atmosphere all the way to the surface. This propellant-intensive approach "solves" all the current conundrums associated with heavy-payload landing on Mars [18], since subsonic retro-propulsion is assumed to be understood. As previously discussed, these propellant modules would have relatively high specific impulse – 420 seconds, and modest propellant mass fraction – 85%. However, the disadvantage of the modest propellant mass fraction would be overcome by the ease of staging away spent tanks by jettisoning banks of small modules after each burn. A human mission to Mars could be accomplished with this system in a way that overcomes many of the existing technology challenges. If two 25-ton ELV payloads are brought by star-arms from LEO to Earth-moon L1, they could be mated together and outfitted with new, very long star arms for the journey to Mars. In this example, about 664 tons of propellant modules are arranged as two rings of star-arms, each containing ~94 modules, emanating radially away at the base of the lower payload. These rings would be offset so that the ends of the upper arms could thrust without interference from the lower arms. About 24.7% of the propulsion modules would be used for Trans-Mars Injection (TMI) by way of a moon-Earth gravity assist. The propulsion modules would then be re-configured as radiation protection during cruise, presumably arrayed along the sides of the habitats, and perhaps the two 25-ton payloads would be separated and spun on a tether for artificial gravity, even as they would be shielded by 520 tons of propulsion modules. At Mars, another 41.5% of the initial propellant load would be used in an all-propulsive maneuver to inject into low Mars orbit (LMO). The 50-ton payload would then perform a "stop and drop" maneuver – propulsively stopping at 125 km altitude (using 28.2% of the original propellant), and then doing a constant-velocity subsonic descent for 500 seconds to the Martian surface (using the remaining 5.6% of the propulsion modules). This would eliminate all concerns over aeroassisted entry of

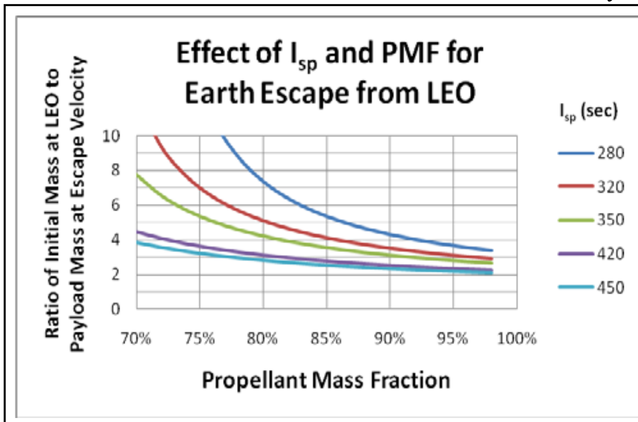

Figure 9: Effect of specific impulse and payload mass fraction on Earth escape from LEO, demonstrating diminishing returns on improvements in PMF with high Isp.

massive payloads, which is presently a completely unsolved problem.

A second 50-ton payload module would presumably travel together with the first for use as an Earth-return system. Departing L1 with 627 tons of propellant modules, this system would remain in low Mars orbit while the crew is on the surface, and provide propulsive departure back to Earth and all-propulsive capture back to the Earth-moon L1 point. The returning habitat would have 24.6 tons of propellant modules with it to provide radiation shielding during the Mars-Earth cruise. This transit habitat, along with whatever infrastructure had been left on Mars, would then be available for re-use on subsequent missions (e.g., every 26 months). Thus four 25-ton ELV launches to LEO, along with several thousand tons of propulsion modules launched from a tethered balloon, could become a 50-ton crewed surface exploration and ascent infrastructure and another 50-ton LMO Earth-return habitat. Travelling together en-route to Mars, either 50-ton habitat could function as a "lifeboat" for the other.

Here we have assumed that the payload modules are limited by the current launch infrastructure to 25 tons and 5-meter diameters; if a new heavy-lift launcher becomes available the situation would improve, since all the "long arms" could be arranged as a single ring around the base of the larger (perhaps 10 m diameter) payload, and overhead required to dock multiple 25-ton payloads together may not be required. We have also assumed that all the propellant modules would be affixed to the payload at the time of departure from a libration point. This would not really be necessary, since propulsion modules could thrust as a group and later rendezvous with a payload, e.g., during transit to Mars, after conducting separate TMI burns.

5. Exploration Architecture Implications

A "propellant-rich" space exploration architecture would have many advantages over the halting approach used to explore space since Apollo:

1. Most or all critical mission events could be performed entirely using well-understood propulsive maneuvers, instead of by aerocapture, aerobraking, or other risky approaches. This would preserve expensive flight hardware such as habitats in such a way that they could be re-used.
2. The large amount of propellant carried would offer additional radiation protection for human crew and avionics. If the propellant is carried in relatively heavy tanks, that would add to the protection.
3. If each propulsion module is self-contained, with no propellant transfer to other systems, then each module would only need to have a modest total burn life (~1000 seconds), after which it would be discarded or used as radiation shielding until the next critical event. The task of re-certifying multi-use rockets as "flight-worthy" would become irrelevant. Only payloads such as habitats would be re-used, not rockets.
4. Low-cost propellant would allow use of dry hardware that is somewhat heavier and perhaps less expensive to develop and deploy than ultra-lightweighted hardware that would be needed if launch of the associated propellant were much more expensive. This lower-cost dry hardware may also be more reusable than the lightest possible version that would meet the requirements. Reuse of such dry hardware, even as few as 3-5 times, could greatly reduce the cost of space missions.

These benefits are applicable to human exploration missions to libration points or other planetary bodies, to robotic sample return missions, or to the launch of large astronomical instruments such as telescopes and interferometers outside the Earth's gravity well.

6. Summary and Conclusions

This paper describes a system of small rockets that would be launched from an equatorially-tethered balloon at an altitude of ~22 km. The purpose of this paper is to further elaborate the engineering and economic issues associated with this proposed system for low-cost propellant launch from a tethered balloon. We believe that this system could deliver propellant for under \$400/kg to equatorial orbit, or ~\$1000/kg to either the Earth-moon L1 point (where it could be stored indefinitely) or to a 28.5º inclination LEO orbit where it could propel dry hardware arriving from Florida out of the Earth's gravity well. The key benefit of this approach is that the major risks could be retired for relatively modest cost (comparable to a single present-day expendable launch) and the entire system put in place for the cost of a single proposed heavy-lift launch (of the Saturn-V class or larger).

Modest initial steps toward demonstrating the feasibility of this approach would include prototyping the small vernier thrusters needed for the payload module, tethering a balloon for perhaps a week at the nominal altitude near Baker Island, prototyping key components of the propellant-depot robot, demonstrating the feasibility of building the "cable car" launcher that would go up and down the tether, and continued system engineering and analysis. An important following step would be to conduct a longer-duration proof of concept experiment at Baker Island using a subscale balloon and tether system to demonstrate stable, long duration tethered balloon flight at an altitude of ~22 km. This would include a buoy with tether-management spool, multiple solar/battery cable cars to clear ice and provide aircraft warning, and the balloon. Preliminary analysis indicates that a sub-scale balloon in the range of 20 to 30 m in diameter would be required for such an experiment, with a likely choice of a high strength fabric-based balloon material and a spherical superpressure balloon architecture. Such balloons could be made robust enough to survive the expected wind speeds both during ascent and float, while the superpressure would help preserve their shape at float and thereby minimize aeroelastic deformations and attendant material fatigue. Important secondary outcomes of the sub-scale experiment would be to obtain long-term direct measurements of the wind, icing, and lightning conditions at Baker Island and to gain operational experience with the cable-car and tether management subsystems.

With this approach, a crewed Mars mission could be undertaken with as few as four current expendable launches, along with a few thousand tons of propellant modules (e.g., ~4 years of operation of the proposed launch system). This would deliver a 50-ton surface exploration and Mars ascent infrastructure (including mobile habitat and ascent vehicle) to the surface without aerocapture or aerobraking, as well as another 50-ton Earth return habitat with sufficient propulsion for all-propulsive capture both at Mars and upon return to Earth. Sufficient propellant would be carried in both directions to provide abundant radiation shielding for the crew. The current Design Reference Architecture (5.0) for proposed human missions to Mars calls for 7-12 cargo launches to LEO at 130 tons each (fewer launches using unproven nuclear thermal propulsion, more using conventional LH_2/LO_2 propulsion). At $10,000/kg, the cost of these launches is 9-17 billion dollars. The corresponding cost of 4 years of operation of the system summarized in Figure 5, plus four 25-ton ELV launches for the dry hardware, is estimated to be 2.7 billion dollars; a savings of 70-85%. The dry hardware can be expected to be lower-cost for our approach, for the reasons given in Section 5, and no risky new developments would be required in the areas of nuclear propulsion or surface power, Mars aeroentry, radiation protection, closed-loop life support, or in-situ resource utilization.

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BIOGRAPHIES

Brian H. Wilcox is a principal member of the technical staff in the Autonomous Systems Program Development Office at JPL. He was the supervisor of the JPL Robotic Vehicles Group for over 20 years, during which the group was responsible for planetary rover development leading up to the Sojourner and Mars Exploration Rovers. The group was responsible for development of the electronics, control, flight software, ground software, and mission operations of the Sojourner rover that explored part of Mars in 1997. Brian was personally responsible for the imaging and hazard detection sensors and the hazard avoidance algorithms on Sojourner. More recently, he is now the Principal Investigator for ATHLETE – the All-Terrain, Hex-Limbed, Extra-Terrestrial Explorer (see http://www-robotics.jpl.nasa.gov/tasks/index.cfm). He has a B.S. in Physics and a B.A. in Mathematics from the University of California at Santa Barbara, and an M.S. in Electrical Engineering from the University of Southern California.

Evan G. Schneider is a senior at MIT where he is majoring in Mechanical Engineering and furthering his lifelong interest in the sciences. He is a graduate of The College Preparatory School, located in Oakland, California, where he won the Science Department prize at the end of his senior year. He designs, builds, and flies model rockets, airplanes, and helicopters, machines and welds, and races remote controlled cars in his free time. He is an avid tinkerer and has dismantled, reassembled, and occasionally improved objects ranging in complexity from clocks to hydrogen fuel cells. He worked as a summer intern at JPL under Brian Wilcox and David Vaughan in 2006 and 2008.

David A. Vaughan is the supervisor of the Propulsion Flight Systems Group at JPL. Prior to joining JPL, he was manager of the Analytical Engineering Department at Lockheed-Martin Michoud Space Systems in Huntsville, AL. He has over 30 years experience in cryogenic and storable propulsion systems. He has been involved with various technology developments including LH_2/LOX propellant densification, the X-33 LH_2/LOX propulsion system, no-vent fill, advanced LH_2/LOX turbo-pumps for the Space Shuttle main engine, and electric ion propulsion. He has a BSc. in Mechanical Engineering from Bath University, England, and a MSc. in High Speed Aeronautics from the von Karman Institute for Fluid Dynamics, Belgium.

Jeffrey L. Hall is a group leader and senior engineer at JPL who leads its technology development and mission application efforts in aerocapture and aerobots. His work includes design of aerocapture vehicles and missions, development of ballutes, Earth flight testing of balloons and airships, and design of advanced planetary aerobots. He has a Bachelors degree in Engineering Science from the University of Toronto, and Masters and Ph.D. degrees in Aeronautics from the California Institute of Technology.

Chi Yau (Tony) Yu received his B.S. in Aerospace Engineering at Georgia Institute of Technology in 2009. He is currently pursuing his Master's Degree in Aeronautics and Astronautics Engineering at Stanford University. Over the summers, he has worked both as a bicycle mechanic in his home town of St. Simons Island, GA, and as a student propulsion engineer for the past four summers at the Jet Propulsion Laboratory. At JPL, Tony has supported a number of projects on various missions such as Mars Science Laboratory, Cassini, and MoonRise. His experience at JPL include, but are not limited to, mechanical design, prototyping, and testing of a throttle valve calibration device, pyrotechnic shock damping device design and testing, hydrostatic test confinement enclosure design, propulsion subsystem modeling, testing, data analysis, validation and integration, battery thermal regulation enclosure fluid dynamics and cryogenic tank system modeling, in-space propulsion module depot concept study, and propulsion system telemetry analysis, validation and budget projection. Tony enjoys working and learning with his hands and meeting challenges with an open mind and new perspectives. His favorite hobbies include mountain biking, hiking, driving and go-kart racing, marksmanship, and rock climbing.