System Technical and Economic Features of QED-Engine Driven Space Transportation

By Robert W. Bussard, Energy/Matter Conversion Corporation,
680 Garcia Street, Santa Fe, NM 87505, <emc2qed@comcast.net>

Introduction

Electrostatic-fusion (IEF) devices using polyhedral magnetic fields for potential well confinement of ions offer promise of light-weight sources of very large thermal and electrical power from clean $p+^{11}B$ fusion reactions. These can drive space propulsion systems (called QED engines) with thrust/mass ratios 100-1000x larger than other potential advanced propulsion concepts, over a specific impulse range of $155 < I_{sp} < 1.2E6$ sec. Analyses of transport from Earth to LEO, LEO to LLO (Luna), LEO to LMC (Mars), and LEO to LTO (Titan, moon of Saturn) show single-stage vehicles with payload delivery fractions of 0.11 - 0.20 of gross launch mass, and transit times of 1 day (Luna) to 5-6 weeks (Mars) to 11-13 weeks (Titan). With these, economic models show that a human colony of 4,000 people could be established and maintained n the Moon for about $12B over 10 years, while 1200 people could be put on Mars or 400 people on Titan for roughly $16B over the same period. Smaller numbers of people would yield commensurately lesser costs. Investment of $4-5B/year could put major colonies on the Moon arid Mars and on Titan (or on Jupiter's moons), well within the NASA budget; the QED engines needed to accomplish this could be developed over 15-17 years for about $ 3.0B.

Nomenclature

$C_{E/L}$ = cost per unit payload E/LLO flight

$C_{E/M}$ = cost per unit payload E/LMO flight

$C_{E/S}$ = per unit payload ElF flight

C_{LEO} = cost per unit payload E/LEO flight

$C_{LEO/L}$ = cost per unit payload LEOIL flight

$C_o(\Upsilon)$ = total cost over mission life, Y years

E = Earth's surface

$[F]$ = engine system thrust/mass ratio

f_L = payload fraction of gross mass

g_o = acceleration of gravity at Earth's surface

I_{sp} = engine system specific impulse

L = Lunar surface

LEO = Low Earth (satellite) Orbit

LLO = Low Lunar (satellite) Orbit

LMO = Low Mars (satellite) Orbit

LTO = Low Titan (satellite) Orbit

M = Mars' surface

m_o = vehicle gross mass

m_L = payload mass

N_L = number of LEO/L transfer vehicles

N_{LEO} = number of E/LEO transfer vehicles

v = vehicle speed

δv_c = "characteristic" velocity of flight mission time

Υ = mission time, years

1. Introduction

Practical space flight can be achieved only with propulsion systems of very large specific impulse (I_{sp}) and high engine system thrust-to-mass ratio $[F]$. With high $[F]$, gravity "losses" in g-field-climbing will be minimal; with concurrent high I_{sp} very high characteristic velocities needed for rapid interplanetary transits can be achieved as well. With such engines economically-useful payload can be carried over very large velocity increments by single-stage vehicles with *realistic* structural factors. The limited energy available from chemical reactions limits chemical rocket payload fractions f_L to small values for most interesting space missions, even with multiple staging. And no chemical vehicle system is capable of rapid (e.g. less than one year) interplanetary flight. Large payload fractions and rapid transits *require* high-I_{sp} engines[1] that *also* have "high-thrust" capabilities.

But such engines must be both light in weight and sufficiently energetic to drive the space vehicles.[2] That is they must have super-energetic performance and *no* massive radiation shielding. The high-I_{sp} potentials of nuclear *fission* propulsion concepts are inherently incapable of achieving these conditions.[3] [4] However, special non-radiative nuclear fusion reactions exist that do not require massive shielding, and that yield only energetic charged particles for direct thermal or electrical power production.

The reaction of most interest is that between the fuels p (^{1}H) and ^{11}B. These can be "burned" in special inertial-electrostatic-fusion (IEF) devices.[5] [6] These trap energetic electrons in quasi-spherical polyhedral magnetic fields to produce a negative potential well that confines the fusion ions.[5] Magnetic fields readily, trap electrons, while electric fields easily confine heavier ions. Such IEF power sources yield energetic (MeV) charged fusion products (alpha particles). These can be used to provide quiet-electric-discharge (QED) direct-converted electrical power[7] at high voltage, to heat and expand or to accelerate directly a working fluid to provide rocket thrust at high specific impulse (1500-70,000 sec). These QED engine systems are most useful for Earth-to-LEO, cislunar, and inner solar system flights.

In a simpler concept, IEF fusion products can be trapped in toroidal magnetic fields around the source, and made to heat a propellant-diluent by inter-particle collision. Such diluted-fusion-product (DFP) engines can give very high specific impulse (5E4-1.2E6 sec), best suited for outer planet missions in the solar system. Details of these engine systems, their components and subsystems, layout, mass breakdown and performance are given in earlier studies.[8] [9] [10] Their variation of I_{sp} with [F] is shown in Figure 1, for these and other concepts of "advanced" fusion propulsion.[11]

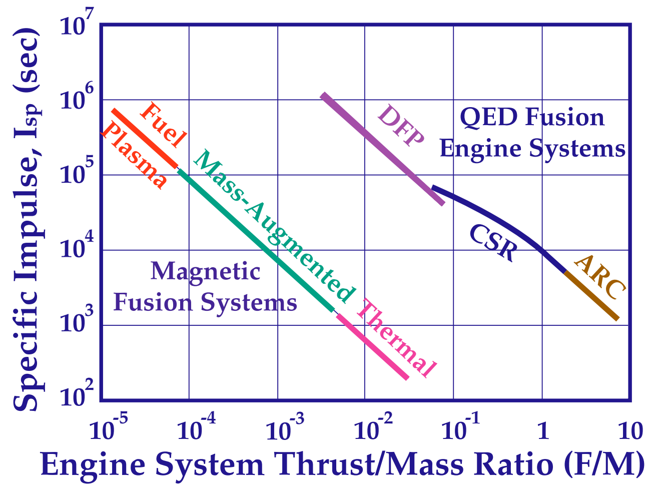


Figure 1— I_{sp} vs. [F] for QED(ARC/CSR), DFP and other "advanced" space propulsion system concepts. [11,13]

This paper presents a summary of cost estimates for several space colonization transport missions, using QED and DFP engines, based on these earlier performance studies. [11] [12] Simple economic models are used to show the costs of transport, establishment and maintenance of human colonies on Luna (the Moon), Mars, and Titan, the large inner moon of Saturn. Vehicle flight performance studies of these three missions have been made previously to assess costs of payload transport from Earth to destination, based on various assumptions of the costs of system development, manufacture and use. These assumptions have been given in detail in the prior works, already cited. Here, the economics analyses use these costs per unit payload delivered to estimate total costs for several models chosen for space colonization.

2. Lunar Colony Mission

The model chosen here for estimation of costs is the establishment of a colony of 4000 people on the Moon over a 10 year period, using a QED/ARC engine-driven HTOL/SSTO vehicle for Earth/LEO transport of Luna-bound payload. This payload includes the equipment, plant, personnel, life support needs, and propellant for transfer from LEO to low lunar orbit (LLO) and subsequently — after proper orbit positioning for landing — to the Lunar (L) surface. Costs are included for rotation of colony personnel to and from Earth once each year, together with return transport of useful payload mass from the Moon.

The outbound people and equipment payload is shuttled up to LEO and transferred to the LEO/L vehicle in-orbit. The LEO/L transfer vehicle is "refueled" on the lunar surface with propellant for the return flight to

LEO by water (propellant) assumed to be available from subsurface lunar sources.

SSTO Vehicle

The E/LEO SSTO vehicle is winged, with a large vertical tail assembly for atmospheric flight stability and a gross liftoff weight (GLOW) taken as 250 T. A rough outline of this is shown in Figure 2, following. The vehicle takes off and lands horizontally, and is driven by two IEF/QED ARC engines using water as propellant, with the capability of variable I_{sp} at constant power. Aerodynamic heating by aerobraking on reentry can be reduced by using reverse thrust from the QED engines, from propellant supplied on-orbit, but this has not been factored into the mass and cost analyses here; aero reentry has been assumed. Also, if desired, popout turbojet engines can be added for short range fly-home capability. Both of these possibilities can best be accommodated by using a slightly larger vehicle than that considered here.

The vehicle flies airborne from takeoff, with wing-lift until M ≈ 10. At this point it is given a high-lift pitch-up to a flight angle of 45° above horizontal, to ensure rapid exit from the sensible atmosphere as the engine system I_{sp} is increased. The flight angle is subsequently decreased with increasing altitude to reduce gravity losses. The net δv_c of the SSTO is 14,3 km/sec from E/LEO. This is split approximately as 7.9 km/sec to satellite orbital velocity, 3.5 km/sec to gravity losses, 2.5 km/sec to drag losses, and 0.4 km/sec to orbit height potential energy addition.

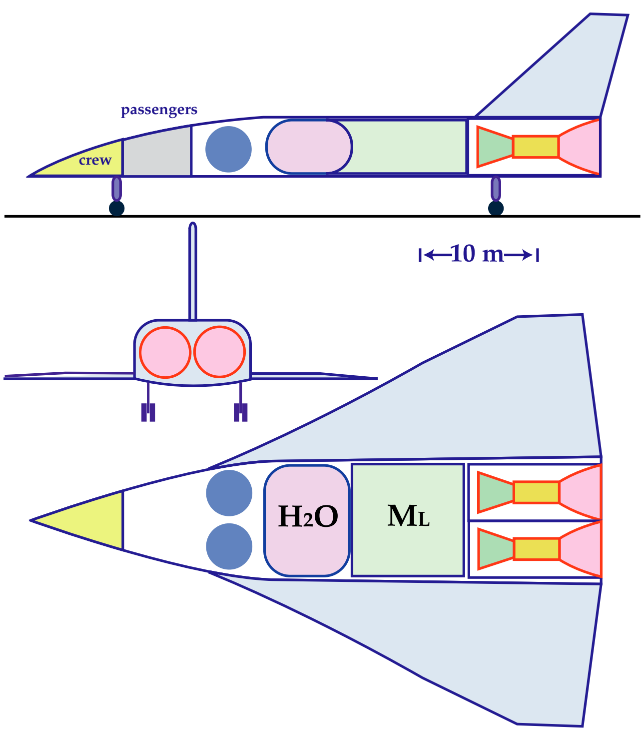


Figure 2 — Outline sketch of E/LEO SSTO vehicle

The two QED engines provide 208.6 T (metric tonnes) initial total thrust at 8000 MWt power each (8400 MWf power, 95% nozzle expansion efficiency), at an initial I_{sp} of 1538 sec. The I_{sp} is varied to yield a system average specific impulse of 2990 sec, well within the capabilities of the ARC engine for use of water as propellant. The I_{sp} is varied as the flight progresses, increasing to 3062 sec at time of aero pitchup. Beyond this point the I_{sp} is increased to its highest value of 3846 sec at the end of the thrust period. This value corresponds to the limiting value for 5% regenerative cooling power. The total thrust from both engines at this time is 83.2 T.

With these parameters, the propellant mass is 95 T, which is expended over a total thrusting time of 2489 sec. In the aero phase approximately 51 T is used over 845 sec, and 44 T is used in the escape-to-orbit phase over the succeeding 1644 sec. The residual dry ("burnt") mass is 155 T, which is distributed as 20 T to fixed flight crew systems/structures, 40 T for the two engines, 30 T for wings, tail and associated structure, 15 T for propellant tanks and structure, and 15 T for landing gear, leaving 35 T as useful payload.

Lunar Transfer Vehicle

The LEO/L(una) transfer vehicle is based on modified subsystems of the E/LEO SSTO vehicle. It is a squat cylindrical structure designed for vertical landing on and takeoff from the lunar surface. A mass saving results from removal of wings and tail assembly, and from use of only a single QED engine. The 50 T mass thus saved is used for additional propellant capacity. However, the payload and other vehicle subsystem masses are kept the same. This eliminates the need for development of completely new vehicle subsystems, even though the vehicle configuration is changed. The engine system is operated in a de-rated mode from that of the SSTO application, in order to ensure system safety and operational reliability.

This vehicle is inherently capable of rapid transit from LEO to LLO and subsequent landing on (and takeoff from) the Lunar surface, A separate LLOL surface lander is *not* needed in this model, thus simplifying the mission; no other surface/orbit shuttle is required. This provides the operational advantage that transfer vehicle propellant resupply (for the return to Earth flight) can be accomplished on the Moon, at minimal cost, rather than through the complexities of on-orbit refueling.

The transfer vehicle single ARC engine has 20 T mass and is assumed to operate at 75% of its constant power design value, or at 6000 MWt thrust power (6300 MWf fusion power) here. A low average I_{sp} = 1590 sec also is chosen. on the LEO/LLO inter-orbit transfer (first) leg, and 2760 sec for the LLOL capture segment. With these conditions, the first and second leg thrust values are 75.5

T and 43.5 T, yielding average vehicle accelerations of about 4.3 m/sec^2 and 3.8 m/sec^2, respectively. Higher payload delivery performance would result if the engine were run at 2760 sec over the complete flight (as for the E/LEO SSTO system). However, operation at these reduced values increases flight reliability, and the higher thrust at LEO liftoff is useful for reduction of gravity losses.

The δv_c capability of the transfer vehicle is taken to be 15.8 km/sec. The 2.4 km/sec requirement for soft lunar surface landing is met by additional on-board propellant capacity allowed by removal of the engine and structure masses discussed above. As a result about 13.4 km/sec is available for LEO/LLO transfer.

The total characteristic velocity capability of the vehicle is used in two thrust periods; one of 2633 sec duration at LEO departure and one of 460 sec duration at LLO arrival, including thrust during subsequent L surface descent. The departure δv is split 4.63 km/sec to LEO escape, 4.75 km/sec to gravity losses, and 1.62 km/sec into **excess** transfer velocity. Because of this latter increment, beyond Hohmann orbit needs, the LEO/LLO transfer time is reduced markedly - to about 20.0 hrs. On arrival at LLO the velocity increment split is about 4.35 km/sec to killing the excess arrival speed, LLO capture, de-orbit and descent, with about 0.45 km/sec additional for lunar gravity losses in the descent phase.

The total system mass is 250 T, of which 145 T is in propellant (water), for a dry mass of 105 T, yielding 42% dry mass fraction. The useful payload is taken as 35 T, as before, with 20 T dedicated to fixed crew cab in structure, support, flight control and on-board safety and life support systems. The engine system mass is 20 T, with miscellaneous system structure of 15 T and propellant tankage mass of 15 T (11.1% of propellant mass). The propellant is consumed in two stages; 125 T on the outbound LEO/LLO leg, with the remaining 20 T at LLO/L descent. The dry mass fraction at Luna landing is 0.42 including payload, or 0.28 without payload; this compares favorably to conventional commercial aircraft. Figure 3 shows a very rough sketch of this vehicle.

Mission Mass Transfer and Costing

Because the velocity increments are so closely similar, the cost of payload delivery to LLO (and subsequently to Lunar ground) is about the same as that for E/LEO delivery, itself. For this study these costs were taken (see prior references) to be:

Lift cost; E/LEO	= \$ 27.0/kg	= C_{LEO}
Lift cost; LEO/LLO/L	= \$ 24.2/kg	= $C_{LBO/L}$
Lift cost; E/L	= \$ 51.2/kg	= $C_{E/L}$

The total transfer vehicle costs are just those of the SSTO less the prorated cost of vehicle R&D. This amounts to \$2.8/kg, and is assumed to be part of the original SSTO development cost.

With the δv capabilities given above, the transit time from LEO/LLO is found to be only 20.0 hour, or 0.83 day. Allow 4.0 hour at Luna or LEO for off-load and loading of payload and propellant; this includes about 0.5 hour E/LEO transit time at the Earth end, and 1.0 hour for orbit position adjustment at LLO before landing. The overall total E/LEO/LLO/L transfer time is then 24.0 hour, for a flight recycle time of once per day for each one-way (OW) leg of the complete round-trip (RT) mission; or 0.5 round-trips (RT) per day.

For costing purposes, it is further assumed that an expendable mass (including the mass of the passenger) of 500 kg must be carried for each person transferred on these flights, and that 25,000 kg (25 T) of capital plant equipment must be supplied and installed on Luna for *each* person delivered to the colony. The model is slightly complicated by the further specification that each colonist must be given a RT from Luna to Earth once each year, to provide some continuing contact with "home." Thus, the number of people being E/L transferred will rise steadily as the colony is built up by continuing delivery of new people.

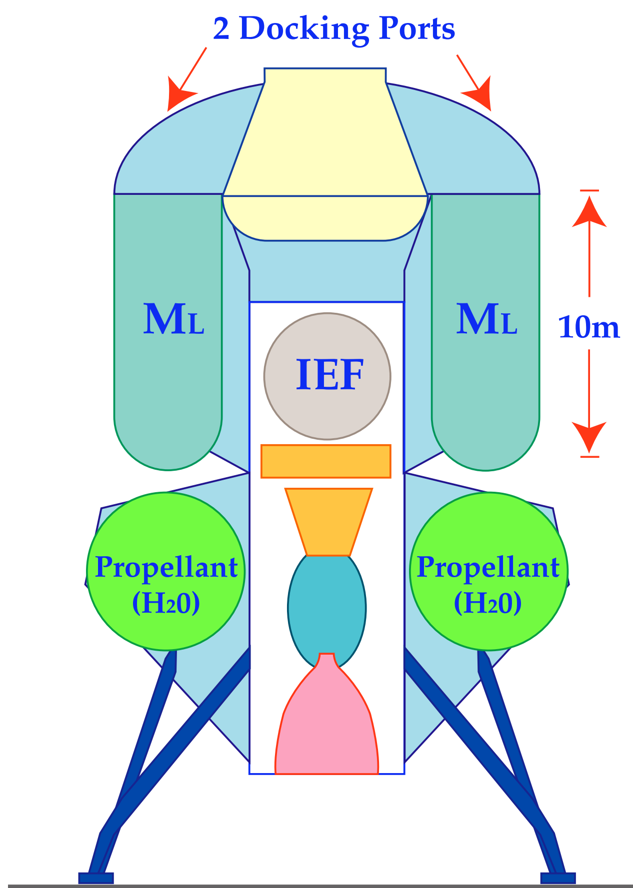


Figure 3 — Sketch of LEO/LLO/L transfer vehicle

However, the transfer rate of equipment mass remains fixed at the rate of new personnel delivery to the colony, which is 400/year. This yields an equipment mass delivery rate of 1.0E7 kg/year at a direct cost of 0.512E9 $/year. A final complexity is introduced by the assumption that operating and maintenance (O&M) costs are 8% per year (0.08/year) of the total capital plant cost installed, calculated as the plant equipment delivery cost to Luna. Since both this and the number of people being transferred (Earth recycle) vary with time, so does the annual cost of this mission.

Finally, it is important to note that 145 T of propellant (water) must be delivered to LEO, or loaded at L (for E-return flight) for every 35 T of useful payload. This mass must be lifted up from Earth via the SSTO shuttles, thus adding more E/LEO flights and cost than are required for the basic payload mass delivery rate, itself. This gives an additional mass factor 4.14 above that for the useful payload alone. Accounting for this the OW mass delivery rate is then:

$$(dm/dY) = 5.14\ (1E7 + 2E5\ Y)\ \ \text{kg/year}$$

The cost of useful payload transfer is the total flight cost, above. However, the propellant mass is costed only for delivery to LEO or for return vehicle loading on Luna. Water on-board at L is assumed to cost the same as given above for delivery to LEO, $27.0/kg. With this, and assuming that O&M costs as given above will apply on both out- and inbound flights, each OW flight leg will cost the same, for the same mass carried back from Luna to Earth as that delivered on the outbound leg. Taking all of these features into account, the annual OW outbound flight leg cost is found to be given by:

$$(dC_o/dY) = 5.12E8 + 0.736E8\ Y\ \ \$/\text{year}$$

which, when integrated, gives the total OW mission cost for 4000 people plus supporting equipment and facilities delivered over its 10 year period to be:

$$C_o(10) = \$\ 12.48E9 = \$12.48\ B$$

Upon completion, the total installed plant equipment is then 1E8 kg or 100,000 T (metric tons). With 4000 people as colonists, the Earth recycle/visit rate will be 4000 people per year. This recycle rate costs 1.053E9 $/year, or about $263,000/year per colonist. If a colony of 10,000 people were desired over the same period the total 10 year cost for OW delivery would rise in proportion, to $31.2 B. As noted previously, the costs cited above are limited to those of the OW outbound delivery leg (except for cited recycle rate).

The actual total cost is about twice this value, since every outbound leg flight is succeeded by a return inbound flight. Costing of this return flight is not practical until some value can be assigned to payload delivered from the Moon to the Earth. If Lunar-derived payload mass has a market value on Earth greater than about $125/kg, the entire mission will pay for itself from lunar-derived product sales. Assessment of this requires a much more complex (and imaginative) model than that employed here.

The E/LEO SSTO vehicle previously studied has a payload capacity of 35,000 kg with a GLOW of 250,000 kg. The Luna transfer vehicle is taken to have the same transfer payload capacity of 35 T. Thus the two vehicles load match in LEO, which is the only in-space point for payload/vehicle transfer. Appropriate modular packaging and cargo bay design should make this transfer very easy. The mass delivery rate is time-dependent, as shown above. With the 35,000 kg payload capacity per vehicle, the number of LEO/L RT transfer vehicle flights required is

$$N_L = 285.7 + 5.71\ Y\ \ \text{LEO/L flights/year}$$

Initially this gives 285.7 RT flights/year while at the end of the mission the requirement is 342.8 RT flights/year. These correspond to 5.49 RT flights/week and 6.59 RT flights/week. Since one LEO/L transfer vehicle can make 3.5 RT per week (each OW flight requires 24 hr = 1 day, a minimum of two vehicles is required. Use of four vehicles would allow one vehicle to be in-orbit on LEO, with one on standby on Luna (surface) at all times, while the required transfer shuttles would be in transit. This provides extreme backup system safety against vehicle failure.

To support the mass lifting schedule required for delivery of *both* propellant and payload to LEO needs an additional 4.14 vehicles for each transfer vehicle arrival and departure (adds a factor of two), or:

$$N_{LEO} = 2365.6 + 47.31\ Y\ \text{SSTO flights/year}$$

This gives 45.4 flights/week and 54.6 flights/week as the E/LEO shuttle requirement. If the SSTO can be shuttled up and back in 12 hours, for example, with 3 hour on-orbit, 1 hour for flight time, and 8 hour for ground turn-around time, then 2.6 SSTO vehicles used continuously would support the system. Allowing 4 SSTO vehicles gives a 65% utilization factor, with considerable margin for down time. Thus only 4 SSTO and 4 LEO/L transfer vehicles used as described here, are needed to support the requirements of the entire Luna colony mission.

These vehicle systems can be built only after the basic QED/ARC engine is developed. Other current studies[12] suggest that this could be accomplished over 15-17 years at a cost of about $3.0B. Allowing another $7-10 B for vehicle development over an additional 8-10 years places the start date for E/L operations of the sort described here at about 2022-2024. Thus, a colony of 4000 people could be well established on the Moon by ca. 2035, at a cost less than 10% of the current NASA budget over 10 years. Nothing else even comes close to this possibility.

3. Mars Colony Mission

Beyond the Moon lies Mars. The characteristic vehicle velocity δv_c required for fast, practical missions to Mars is very much greater than that needed for rapid Lunar transport. But both missions need the E/LEO services of the SSTO, to provide vehicle structure, passengers, propellant and payload to LEO for LEO/LMO transfer vehicle transport to low Martian orbit (LMO), and thence to Mars' surface. Here, unlike the Lunar case, it is most convenient to utilize a *special* LEO/LMO transfer vehicle designed for use in rapid transit flight to the Martian orbit.

Previous studies showed two examples of this in a 500 T single-stage system.[11] One of these was powered by a single 6000 MWt ARC/QED engine system operating at fixed specific impulse of 5500 sec: the other used a controlled space radiator (CSR-A) engine system, with 50% use of two 1000 MWt systems (only one operating), able to operate at propellant specific impulse up to 7800 sec. The vehicle payload capacities were 72 T for the ARC vehicle and 103 T for the CSR-A vehicle, while the flight times were 33.2 and 37.9 days respectively, for transfer at Mars' closest approach.

Thermal REB heating of propellant is avoided in another engine type (CSR-B) that uses electrical particle acceleration for propellant exhaust, with a space radiator system to achieve higher I_{sp}. A schematic outline of these engine system types is given in Figure 4; their design features have been detailed previously,[13] and their performance is given in summary form in Figure 1, above.

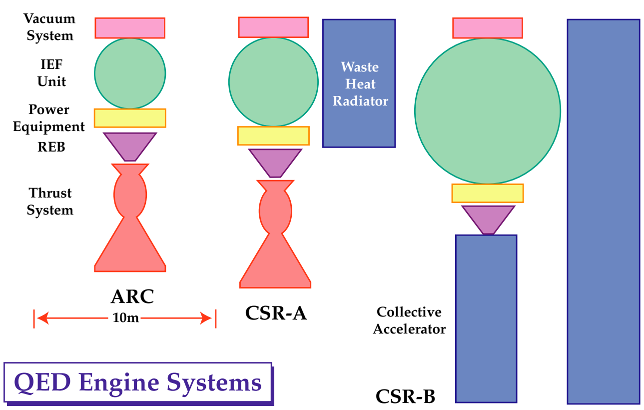


Figure 4 — Schematic outline of sequence of QED engines

A fundamentally different IEF-powered system uses direct heating of diluted fusion products (DFP), without the medium of temperature-limited thermal-to-electrical conversion machinery, to achieve ultra-high temperatures and propellant specific impulse. The I_{sp} vs [F] performance range of this DFP engine system, shown in Figure 1, is best suited for single-stage vehicle flight missions to the outer planets and beyond, where $I_{sp} > 50{,}000$ sec is required, while the ARC and CSR-A,B engines best fit transfers to the inner planets at the level considered here for the Mars Colony Mission. The DFP engine system is discussed further in the Titan Colony Mission section, following.

In this study it was desired to avoid the use of LH_2 (as in the prior examples [9]) and to use water — indigenous to Mars — as propellant, yet to retain the approximate payload capability of the ARC example system. But the ARC engine can not produce $I_{sp} = 5500$ sec with water within the limit of 5% regenerative cooling fraction used above. The desired I_{sp} requires <2%; this can be attained with a CSR-A engine and its waste heat radiator. To this end, a modified CSR-A-driven vehicle of 500 T was taken, with 329 T of water propellant and a dry mass of 171 T. The system studied was assumed to operate at 3000 MWt thrust power from dual 3150 MWt IEF source units, but each running at half power or, alternatively, with one on and one off (to provide a fail-safe margin), producing a thrust of 10.9 T at fixed $I_{sp} = 5500$ sec throughout the flight. This vehicle has an engine mass of 43 T, including 18 T of radiator system, 50 T to structure, tankage, crew and life support (as for the LEO/LLO.L-vehicle), leaving 78 T for net payload.

The reduced propellant mass yields a smaller characteristic velocity of $\delta v_c = 59.0$ km/sec (vs. 71.0 km/sec for the reference CSR-A system) which results in a longer minimum E/M transfer time of 42.3 days; average transfer time is about 65 days over the range of E/M constellational configurations. With the defined Mars transfer vehicle, the system flight time of 42.3 days is found to split into 1.2 days of thrust acceleration at LEO departure, 40.4 days coasting, and 0.7 days of deceleration thrusting at LMO arrival. These figures would he reversed for the return LMO/LEO flight segment. The gravity losses during thrusting are 4.0 km/sec out-bound from LEO and 1.2 km/sec at LMO capture, for a total of 5.2 km/sec; with these the initial coast speed is 25.5 km/sec. The corresponding travel distances are 2.0 Mkm, 88.6 Mkm and 0.6 Mkm for the transit.

Once on LMO the transfer vehicle "refuels" with water from LMO orbit tanker stations; three of these are in an equilateral constellation array, to allow case of Mars' orbit capture maneuvering by the transfer vehicle. These stations are reloaded by a continuous M/LMO shuttle system that carries water, return payload, and recycle personnel up from the Martian surface to match the return rates required by the transfer vehicle system. This M/LMO shuttle is similar to the E/LEO shuttle, but modified to operate with a single engine of 8000 MWt power. It must produce a vehicle characteristic velocity of $\delta v_c = 7.0$ km/sec (4.1 km/sec for orbital speed, 1.9 km/sec to drag, 0.6 km/sec to gravity losses) for M/LMO flight. This requires only 60 T of water propellant at an average $I_{sp} = 2550$ sec, with average thrust of 106.7 T, the

I_{sp} can be varied to optimize flight performance as desired. With these conditions, the vehicle mass is distributed as 20 T to engine, 10 T to tankage, 30 T to wings, tail, etc., 20 T for crew and associated systems, and 10 T for landing gear. This gives a payload capacity of 100 T, or 40% of gross mass.

Each on-orbit tanker station is 5 m in radius, with a 20 m cylindrical straight section between spherical end caps. The volume of each is 6.25x larger than the propellant volume required for a *single* LMO/LEO return flight, thus providing a large safety margin for overall system operation. Each tanker station is vertically-oriented in LMO and is gravity-gradient stabilized. Propellant transfer is accomplished by a controllable boom transfer line from the tanker station to the transfer vehicle propellant tankage.

To match the transfer rates required by the Mars Colony Mission, M/LMO shuttles must make 3.35 propellant delivery flights for each LMO/LEO return transfer flight. The model here allows 5 shuttles per return flight. This gives a capacity of 1.65 shuttles for down-link de-orbit propellant for shuttle return to the surface. One flight up and back per day is needed. With five vehicles, one can remain on LMO and one on the ground at all times, giving down-link redundancy and providing backup safety to the system.

De-orbit LMO/M flight requires significantly less propellant than up-link delivery, because drag can be used to reduce vehicle speed. An initial thrust period would be followed by Mars aero-braking, with a final thrust period utilized for landing on Mars' surface; analogous to the flight profile used for the LEO/E surface return system. The propellant allocated, above, would allow 0.49 of the de-orbit mission delta-v under thrust, with only 0.51 δv taken by atmospheric drag. With this system, the cost of M/LMO payload delivery (or LMO/M transfer) was taken as $24.2/kg, as before for LEO/LLO/L delivery. The total specific cost of E/M surface-to-surface transfer is then found to be

$$27.0\ (\mathrm{E/LEO}) + 181.4(\mathrm{LEO/LMO}) + 24.2\ (\mathrm{LMO/M}) = \$\ 232.6/\mathrm{kg} = C_{E/M}$$

The specific capital plant mass required for Martian colonists was assumed to be twice that for the Luna Colony Mission, at 50,000 kg (50 T) per person. In addition an expendable mass of 1000 kg per person (again, twice Luna) was allowed per transfer/trip, each way. The colony was based on a personnel complement of 1200 people, to be delivered over a 10 year period. As before, one RT per year is allowed per person for return to Earth, once settled on Mars.

Using the vehicle performance model cited previously, the total OW mass delivery rate is found to be

$$(dm_o/dY) = 3.24\ (1E7 + 2E5\ Y)\ \ \mathrm{kg/year}$$

including propellant to orbit, initial colonist transport to Mars, personnel recycling, and Mars capital plant and equipment. Over 10 years the total mass transported OW (including propellant) is 3.56E8 kg or 356,000 T (metric tons). Using the formula and the costing figures above, the annual OW outbound E/M cost is:

$$(dC_o/dY) = 1.424E9 + 2.79E7\ Y\ \$/\mathrm{year}$$

Integrating this gives the OW cost for 1200 people plus supporting equipment and facilities, delivered over the 10 year Mars Colony Mission period as:

$$C_o(10) = 15.64E9 = \$15.64B$$

For this mission, the total mass of installed plant and equipment is 0.6E8 kg or 60,000 T. As for the Luna Colony case, the personnel recycle specification leads to a large return flight cost requirement; so that the total operating cost for colony establishment will be about double that just estimated. But, as before, if Mars-Earth return payloads can find Earth (or Luna) markets at sufficient prices (e.g. at over $200/kg), every return flight segment could be made to pay for itself. Assessment of this possibility can not be made without more knowledge about such future markets for off-planet exotic materials, goods, or systems.

The number of E/M (LEO/LMO) transfer flights initially is 78.46 RT flights/year; at 10 years this rises only to 80.00 flights/year. Taking an average of 3 RT flights/year per vehicle shows that 26.7 vehicles fill this requirement. Assume 30 vehicles are assigned to the LEO/LMO mission; then 27 will be in service at any given time, leaving 3 for other uses. These three could logically be distributed so that two vehicles are on-orbit at LMO at all times, while the other is on-orbit at LEO, to provide backup redundancy and system safety for the transfer segment of the mission.

This transfer rate gives a colonization rate of exactly 10 people each month or 30 people moved to Mars every quarter. Since about 80 RT flights/year are flown, the personnel transfer could be accomplished by sending 15 people out on every tenth flight, at intervals of 1.5 months. The first colonists could set up initial capital plant and equipment (housing, power, life support, transportation, M/LMO shuttles, water extraction, communications, transfer traffic control and management, orbital tankers, etc.), with subsequent colonists expanding from the initial very small base. Shipment compositions and equipment and personnel schedules would necessarily change over successive flights, to optimize the colony setup, deployment, and operation. These can not be considered here; the rates cited above are only averages to illustrate the general nature of the colonization system and process that could be achieved with QED engines.

4. Titan Colony Mission

The final mission considered here is that of establishing a colony on Titan, the largest moon of Saturn. This body is believed to have ice (and thus water) on its surface; it is larger than Earth's moon, and has a larger surface gravity, being about 1/5 that of Earth.

It is sufficiently far from Saturn, itself, that its orbit is not prohibitively deep into Saturn's gravitational energy well. Thus it is accessible to ultra-fast deep-space vehicles with high delta-vee capabilities, without significant δv propulsion and payload penalties.

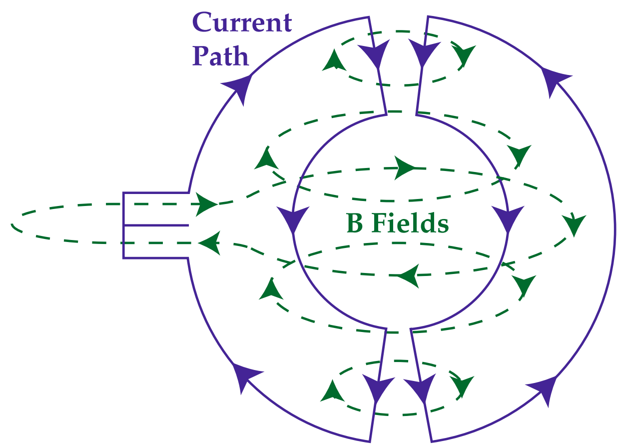
However, in order to accomplish such a colonization mission over a reasonable period of time — e.g. ten years, as used in the previous cases — the vehicle propulsion system used must be able to provide transit times measured in tens of days, rather than in tens of years. The orbit of Saturn varies from 9-10 AU from the Sun, thus its distance from the Earth may vary from 8-11 AU, depending on whether it is in opposition or conjunction with the Earth. Average flight speeds needed are then in the range of 10 AU per 100 days, or 15 Mkm/day or 200 km/sec, for example. Such average speeds imply four times this for vehicle characteristic velocity for an acceleration-coast-deceleration flight profile, thus this must be in the range of δv_c = 800 km/sec, a truly stunning value by current standards.

Simple vehicles are single-stage with reasonable structure factors and payload fractions. These are attainable only with propellant mass fractions that are less than [1 - (1/e)] of the launching gross mass. Such "(1/e)" vehicles can be attained only if the propulsion system propellant exhaust velocity is approximately equal to the vehicle characteristic velocity, so that $I_{sp} \approx \delta V_c/g_o$. These requirements have been employed in design scaling of engine systems and vehicles for the Luna and Mars colony missions just discussed, which set the engine system I_{sp} and thus the choice of engine type (e.g. ARC vs. CSR-A, etc). However, to do so here requires that the engine system specific impulse be in the range of $I_{sp} \approx$ 80,000 sec.

The QED, ARC and CSR engines are all, in the final analysis, limited by thermal considerations of propellant pre-heating from unavoidable regenerative cooling, or by the mass of radiators needed to avoid such limits.

These limits or overlapping boundaries between engine types are suggested on the I_{sp} vs. [F] curve shown in Figure 1. To exceed these limits it is necessary to abandon intermediate thermal-to-electrical power conversion machinery for propellant heating, and go to *direct* collisional heating of propellant/diluent by the fusion product ions themselves. This requires a fundamentally different type of engine than those cited for the inner planet missions. The engine needed for these outer planet missions must be able to give propellant specific impulse values in excess of 50,000 sec, to yield useful, simple vehicle systems. Such an engine can be based on use of the IEF fusion core as a source of fast fusion product ions for direct heating of propellant/diluent.

The IEF source is contained in a spherical toroidal external magnetic field, shown schematically in Figure 5, that is able to confine diluent ions for relatively long periods. Here fast alpha particles from fusion reactions of p + ^{11}B in the source will be trapped in the external toroidal field, until they have made sufficient collisions with the diluent ions therein to reach near local thermodynamic equilibrium at the desired mixed-mean energy level. The heated diluent/propellant mixture then can escape through a magnetic diverter nozzle on the torus equator, as shown in Figure 6, to provide thrust.



Current Flow and Toroidal B Field Configuration

Figure 5 — B field configuration in DFP engine

Since diluent heating is by direct ion/ion collisions, almost any material can be used as propellant to reach any desired I_{sp}; water is again a convenient choice for this mission. The performance of this diluted fusion product (DFP) engine is shown in Figure 1; I_{sp} ranges from 5E4 sec to 1.2E6 sec. A more detailed discussion is presented in previous work,[12] giving design conditions required for performance levels of interest for fast missions to the outer planets, and beyond.

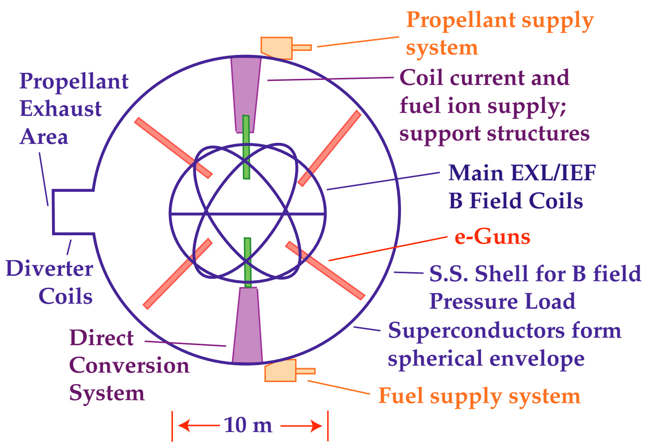


Figure 6 — Schematic outline of DFP engine system

The engine system employs a 5 m radius IEF source inside a 10 m radius spherical torus shell. This outer shell provides a *B* field of 4.1 Kg. This is large enough to trap both fusion products and diluent ions (dissociated and ionized NH_3, water, CH_4, etc.) for sufficient collisional heating to reach the energy per particle needed for the I_{sp} desired. The I_{sp} required for no-coast flight ranges from 72,500 sec to 80,700 sec for transits over distances ranging from 8.0 AU (closest approach) to 11.0 AU (most distant path). A specific impulse value of I_{sp} = 70,000 sec was used for the analyses here; thus all mission flight profiles utilized mid-course coast flight segments.

The Earth to Titan transfer vehicle using this DFP engine would leave from LEO and travel to a low orbit about Titan (LTO). Unlike the LEO/L vehicle, its thrust is too small to allow direct surface landing; some form of LTO/T shuttle is required to offload passengers and payload to the Titan surface. However, the low surface gravity of Titan allows a T/LTO shuttle of only modest δv capability, as compared to that required for the M/LMO shuttle vehicle.

The E/LTO transfer vehicle loaded gross mass in LEO is taken to be 400 T (see prior work [12]), with an engine system mass of 52 T (including waste heat radiators for 2% power direct conversion), 252 T of propellant, 25 T for fixed crew and habitat quarters and associated life support systems, and 26 T for structure (propellant tankage, etc). The remainder is useful payload of 45 T. The vehicle outline is shown in the sketch given in Figure 7, following.

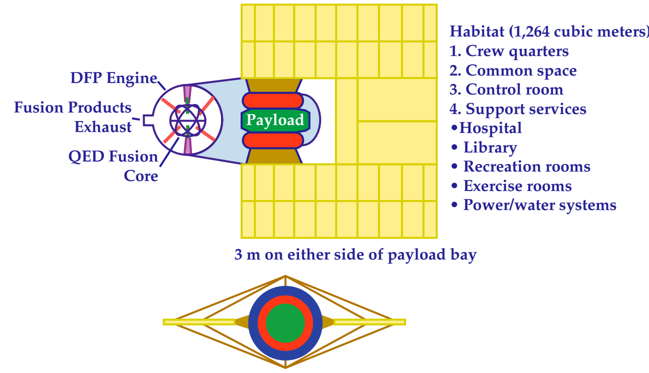


Figure 7 — Schematic outline of LEO/LTO transfer vehicle

With these masses, the vehicle speed at the end of the initial thrusting phase is found to be 343.2 km/sec, or 0.2 AU/d. Additional vehicle energy required for capture into Titan's orbit around Saturn and into low orbit around Titan must be added at this point in the flight. Since escape from Saturn at Titan's orbital distance requires 8077 m/sec and escape from Titan's surface needs 2990 m/sec, the total velocity increment needed at Titan is 11067 km/see, or about 11.1 km/sec.

Because of the very high speed of 343.2 km/sec at mid-course, the energy addition can be made here with a velocity increment of only $[(343.2)^2 + (11.1)^2]^{0.5} = 173$ m/sec; which requires about 125 kg of additional propellant. This is trivial and ignorable as compared to the rest of the flight requirement.

The LTO/T shuttle is based on the subsystems and configuration of the LEO/LLO/L transfer vehicle: it also uses vertical takeoff and landing for ground/orbit transfer. To match LEO/LTO on-orbit delivered payload masses requires a shuttle mass of only 120 T, with a 6000 MWt ARC engine operating at an average I_{sp} = 3000 sec.

This vehicle has 18 T for crew et al, an additional 7 T for landing gear and structure, and 5 T for tankage with only 20 T of propellant needed for the ground/orbit mission. The characteristic δv_c, required for this is only 5.73 km/see, of which 2.73 km/sec is in gravity losses. The small propellant mass per unit payload results in additional cost from LTO to ground of about 45% of that for LMO/M delivery cost, but this latter cost is used here (see below).

For this mission, it was assumed that an on-ground colony of 400 people would be established on Titan over a period of 10 years, with recycle rotation back to Earth allowed for each colonist, once every year. With the chosen specific impulse, the transit times for E/T flight are in the range of 75 to 90 days, or 2.5-3 months for each OW trip. Personnel recycle then requires 5-6 months, and LEO/T vehicle flight usage can not practically exceed 1.5 RT per year. Allowing 6 months on Earth for rotational leave then gives only a 50% duty factor for Titan-based personnel.

With this assumption 400 people must be in transit at any one time *after* establishment of the 400 person on-ground colony. While the colony is being established, the personnel interplanetary transfer rate is about twice the on-ground population count, but the transport rate of capital plant and equipment to set up the colony is just that required for the 400 colony staff personnel on the ground.

From these assumed ground rules the economics can be estimated in the manner used previously. However, here 60 T of capital plant and equipment, and an expendable mass of 2000 kg/person is assumed for each transfer flight.

E/LEO transport costs are taken at $27.0/kg, as before, and at $24.2/kg for low Titan satellite orbit to ground (LTO/T), as for the LMO/M Mars mission costs. The LEO/LTO delivery costs have been shown to be $280/kg, excluding any financial costs, but with O&M costs taken at 2% per flight over 100 flights per vehicle. These economic and cost assumptions and conditions have been discussed in more detail previously.[8, 9, 13]

Under these conditions the total cost rate for this mission becomes:

$$(dC_o/dY) = 1.235\text{E}9 + 0.772\text{E}8\ \text{Y}\ \ \$/\text{year}$$

and the integrated cost over the 10 year period is:

$$C_o(10) = \$\ 16.21\text{E}9 = \$16.21\ \text{B}$$

for OW delivery of a 400 person colony to Titan, with 24,000 T of capital plant together with supporting facilities, housing, life support, medical, communications et al, and transfer vehicle maintenance and refueling equipment.

As discussed above, RT costing requires knowledge of market value on Earth of material transported back from Titan, or from other points in the outer solar system. Certainly such vehicles could be used to mine the asteroids, to move small asteroids, establish colonies on the moons of Jupiter, etc, whose potential future value is yet difficult to estimate.

5. Summary

The three Colony Missions analyzed here cost a total of about $45 B for their completion, allowing for market value of returned payloads to equal return operating costs. Thus colonies of 4000 people, 1200 people and 400 people could be placed on the Moon, Mars and Titan, for a total cost less than 1/3 of the current NASA annual budget over a ten year period. Once established these could become self-supporting and engage in expansion of their own, again using QED/DFP engine systems.

The value of these colonies is almost impossible to quantify at present. The only model available for such guesstimates is that facing Columbus when he first set sail for what was to prove to be the North American continent. History has shown that his estimates and those of his contemporaries were woefully inadequate to project the future value of the civilizations and industries that have grown in this new environment.

Would space colonization be expected to be any less important or valuable? The IEF QED/DFP engine systems offer almost the only means presently seen to carry out such ventures. Their development and use will give us the tools to find the answers to these interplanetary questions.

Revision History

Presented as Paper AIAA 97-3071 at the 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, July 6-9, 1997, Seattle, WA.

Reformatted and color illustrations added May 2007 by Mark Duncan. References updated in April 2009.

References

[1] Maxwell Hunter; "Solar System Spaceships," *Thrust Into Space*, Holt, Rhinehart & Winston, NY, 1966, Chapter 5

[2] Robert W. Bussard and Richard D. DeLauer; *Nuclear Rocket Propulsion*, McGraw-Hill Book Co., NY, 1958

[3] Robert W. Bussard and Richard D. DeLauer; *Fundamentals of Nuclear Flight*, McGraw-Hill Book Co., NY, 1965

[4] Robert W. Bussard; "ASPEN-II: Two-Staging and Radiation Shielding Effects on ASPEN Vehicle Performance," LA-26-80, 6/25/62, declassified w/deletions 9/6/67: and in "ASPEN: Nuclear Propulsion for Earth-to-Orbit Aerospace Plane Vehicles," Robert W. Bussard, Proceedings International Conference on Spaceflight, Rome. June 1971

[5] Robert W. Bussard; "Method and Apparatus For Controlling Charged Particles," U.S. Pat 4,826,626, 5/2/89, and "Physics Considerations of Magnetic-Inertial-Electrostatic Confinement - A New Concept for Spherical Converging-Flow Fusion," Fusion Technology, Volume 19, March 1991, 273

[6] Nicholas A. Krall; "The Polywell: A Spherically-Convergent Ion Focus Concept," Fusion Technology Volume 22, August 1992. p. 42

[7] Ralph W. Moir and William L. Barr; "'Venetian Blind' Direct Energy Convertor for Fusion Reactors," Nuclear Fusion, Volume 13, 1973, 35, and "Test Results on Plasma Direct Convertors." Nuclear Technology and Fusion, Volume 3, 1983, 98

[8] Robert W. Bussard; "The QED Engine System: Direct-Electric Fusion-Powered Rocket Propulsion Systems." Paper #263, Proceedings 10th Symposium Space/Nuclear Power and Propulsion, Albuquerque, NM, 10-14 January 1993

[9] Robert W. Bussard and LW. Jameson; "The QED Engine Spectrum: Fusion-Electric Propulsion for Air-Breathing to Interstellar Flight," AIAA ppr 93-2006, 29th Joint Propulsion Conference, Monterey, CA 6/28-30/1993; in JPP, Volume 11, Number 2, pp 365/372

[10] Robert W. Bussard, Lorin W. Jameson and H.D. Froning, Jr.; "The QED Engine: Fusion-Electric Propulsion for Cis-Oort/Quasi-Interstellar Flight," Proceedings 44th Congress International Astronautical Federation, 10/22/1993, Graz, Austria

[11] John F. Santarius; "Magnetic Fusion Energy and Space Development," Proceedings 24th Intersociety Energy Conversion Conference, IEEE, Volume 5, 1989, pp 2525

[12] Robert W. Bussard and H.D. Froning, Jr.; "Milestones, Costs and Schedules for QED Development," private communication; April 15, 1997

[13] Robert W. Bussard and Lorin W. Jameson; "From SSTO to Saturn's Moons: Super-performance Fusion Propulsion for Practical Spaceflight," AIAA paper #94-3269, 30th Joint Propulsion Conference, Indianapolis, IN, June 27-29, 1994