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ELECTRICAL PROPULSION AND SPACE POWER SYSTEMS

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INTRODUCTION

The overwhelming advantages of Electrical Propulsion compared to Chemical systems make the use of chemical systems for any mission of more than 10 days and anything past Lunar orbit idiotic. Yet many mission planners persist in use of those proven but obsolete technologies, perhaps because they know them and do not understand electrical propulsion. That makes those planners also obsolete. The ISP of ion engines or MHD devices is well over 5000 [↑]sec. with even 10,000 or more possible, compared to 400 - 450 sec. for chemical systems.

[↑]ISP = specific impulse in pounds of force thrust per pound per second of mass expended, usually expressed in the improper but convenient units of “seconds” – the real units are that of velocity and represent the “exhaust” velocity. The exhaust velocity of an electrical system is of the order of 160,000 ft /sec [50,000 m/sec] compared to 14,000-ft/ sec for chemical systems. The optimum exhaust velocity should match the peak mission velocity).

A typical mission with electrical systems allows 25% payload from LEO (Low Earth Orbit), it requires 50% expendable “fuel” and has 25% structure and power supply (ship empty weight). For Mars this compares to chemical 12% or less payloads. The precise acceleration is dependent upon the weight to power ratio of the power system. For Solar Cell solar power or Nuclear SNAP (Small Nuclear Auxiliary Power) systems this is of the order of 0.01 to 0.001 g (1/100 to 1/1000 of the acceleration due to gravity at Earth’s surface). Early designers likened this to one butterfly power, but this tireless butterfly pushes day after day, and for longer missions the cumulative effect is phenomenal, and the total trip time is far less than with the impulsive chemical systems. Even better you are not forced to travel with times set by planetary positions and Hohmann transfer windows. And a change in mission still allows you to get home if you have food and air enough, stretching the fuel to cover the new longer mission. The thrusters are very small, very like a 2 liter soft drink bottle, or a “bread box” pushing a large semi-trailer. Thus multiply redundant systems and spares with manned missions are possible increasing safety.

With chemical systems we are restricted to a window roughly once every 2 years. This happens only when the Hohmann minimum energy transfer ellipse from Earth orbit to Mars orbit touches Earth orbit at a leaving time, where Earth is at the start, and must touch Mars orbit when Mars is at that position to greet us. It is always possible to enter that Hohmann transfer orbit from Earth at any time, but it does no good to arrive at Mars’s orbit if Mars is not there. We can make slight variations in the transfer by using more energy to open a window of about 2 weeks or even 3 weeks from optimum, but this always is at the sacrifice of some payload. The typical Hohmann transfer time is about 277-280 days to coast to Mars.

As example let us take the lower very conservative value of 0.001 g, and calculate a Mars mission. Going up in acceleration a factor of 10 to 0.01 g reduces the times below by a factor of 3.16 (the square root of 10). With Electrical propulsion the window is 365 days of the year. The time varies only from 100 to 140 days and it only takes slightly longer going from worst possible to best possible positions. Let us take the super conservative worst possible position, where Earth and Mars are at opposition, the furthest possible position. We add the 93 million miles (Mm), (150 million km or 150 Giga meters) Earth to Sun distance to Mars' 140 Mm (230 million km or Gm) for a total distance of about 230 Mm (380 Gm). The trip must be calculated in two parts, acceleration to turn over at roughly half way total distance and deceleration to arrive at Mars with velocity low enough to go into orbit, i.e. two equal parts of roughly 115 Mm (190 Gm). Using simplistic distance = _ a t^2 we find 71 days to turn over and 71 more days to Mars for a total of 142 days at 0.001 g. (6.2 x 10^6 sec. to turnover).
At turnover the velocity would be 200,000 ft per sec (132,000 mph)(61,000 m/sec). This is far beyond "orbital" velocities, and thus the ship can not really be said to be in Solar orbit at all. Earth gravity is controlling at the start and Mars gravity well controls the end of the journey, but in the middle the ship is moving so much faster than solar orbital velocity that the solar gravitational term has become a secondary calculation!

Taking the best position of Mars of about 55 Mm (90Gm) the trip takes about 100 days total, 50 days each half, with peak velocity 140,000 ft/sec (93,000 mph, 155km/hr., 42,000 m/sec). The payload remains at 25%, but if there are people then the supplies of air and food consumed must be accounted for as a decreasing part of payload in the longer trips. Any way you look at it the electrical propulsion gives larger payload, shorter time, extra power to use if needed, the ability to stretch available fuel to cover longer missions, etc. It is superior in every detail.

The ISP that goes with a 100-140 day mission is roughly 5000 seconds. That is a fuel use such that the ship uses 100% all of the available electrical energy all the time, and rations out the fuel so that it will just have enough fuel to complete the mission. If it uses fuel any faster, then it will find itself in the embarrassing position of having run out of fuel before it completed the mission. If it uses less fuel, then it could have completed the mission sooner. The ISP depends directly on the mission time. The longer the time, the higher the ISP that must be used (square root proportionality).

With 0.01 g the times drop to 32 to 44 days! That is no worse than the colonial sailing ships of four centuries ago. The times for Jupiter and the planets farther out also are within reason, Jupiter is about 200 days at 0.001 g, 63 days at 0.01 g with very little variation since Earth position almost does not matter any more. Saturn is 275 days, or 85 days and even Neptune / Pluto is only 488 days at 0.001 g, or 150 days at 0.01 g. <u>The Planets are open to us with this technology! THIS IS HERE NOW.</u>

There are three basic types of electrical propulsion that have been developed:

1) Thermal Arc Jets, TAJ
2) Ion Propulsion
3) MHD, Magneto Hydro Dynamic, or J X B accelerators.

THERMAL ARC JETS

The TAJ's are simple rocket engines but with the difference that the gasses are heated electrically to make them much hotter than is possible with chemical rocket engines. In chemical engines the energy is limited to that of the chemical reaction. With TAJ as much energy as is desired up to the temperature limit of the material in the thruster itself can be added. With Carbon-Carbon throat this is about 3000 K. As a matter of practice ISP of 950-1100 sec. are all that are practical, which is ideal for 10 day lunar missions, but not useful for Trans-lunar missions such as Mars. The weight ratios for this system are the typical electrical 50% propellant, 25% payload, 25% structure and power supply. Assuming that the basic system is reusable for a lunar shuttle, then the pay off comes with only one reuse over chemical systems, and is overwhelming for three or more uses. Thus TAJ are probably the L-4 taxis of the future.

HYBRID TAJ CHEMICAL SYSYTEMS

I propose a second possible variation here, a hybrid chemical / electrical system. In this part of the energy comes from a normal chemical reaction in a more or less normal rocket combustion chamber and then the residual allowable energy up to the throat materials temperature limit is pumped in from electrical energy. This hybrid system has the advantage of better thrust to weight ratios, but the disadvantage of requiring two propellant feed systems with all the control complexity of normal rockets. It also is markedly safer, being more "fail safe" in that an electrical system failure is not necessarily fatal, and often an aborted mission, dumping cargo to save human crew, or part of the cargo, is possible with the chemical system alone. This system is the preferred Earth-Lunar and L-4 / L-5 orbital transfer system. The mass transfer ratios here change, and assuming H2- LOX fuel, with the energy for 450 seconds from the fuel and 550 from electrical power, the weight ratios change to about 28% structure and power supply, 30% fuel, and 42% payload. Using economic optimization, even higher payload fractions are possible for less fuel, by taking longer times to complete the mission. Thus the overall efficiency factor is roughly 40% higher. The weight factor optimization is highly dependent on structure weight and power supply weight to power ratio, but the bottom line 40% increase in payload is not sensitive to that weight factor design optimization. The efficiency is changed upward only a little (a 2.7% payload increase) by going to H2 - F2. Using standard 1000 psi chamber to 1 ATM. Values, 398 ISP goes to 409; and that ratio is maintained for higher chamber pressure and higher vacuum expansion ratios.

ION ENGINES

The Ion engine uses electrostatic forces to accelerate charged particles. In the past these have been charged ions of Cesium, Mercury, or inert gasses. In practice any mass that can be charged will do, and a fine powder of "moon dust" with electrostatic charge will work as well as anything else. With gas expansion rockets the velocity of the exhaust depends inversely on the square root of the mass of the gasses, and thus hydrogen is the ideal working fluid. With H2-

LOX for example the highest ISP exhaust is not the ideal stochiometric combustion ratio of two hydrogen atoms for each oxygen burned (weight 16:1 or 6.25% fuel), but rather a fuel (H2) rich mixture of about 10-11% hydrogen that lowers the exhaust molecular weight to get higher velocity. In air this also leads to after-burning of the exhaust plume as air mixes into the exhaust. With ion engines the mass of the particle accelerated does not particularly matter, only the efficiency of coupling the energy flow to the particles.

The energy in any one particle is mv^2 and the sum total of energy for all n particles is a constant, nmv^2, limited to that energy flow which can be supplied by the power supply. The thrust or each particle is the momentum mV, and the total thrust is nmv. As the mass is raised the number must drop in proportion because of the power limitation, and thrust is not changed.

Ionization can be done by simple electron bombardment, knocking off electrons and creating positive ions. These ions which are ejected must be neutralized. For positive ions this can be done by emission of electrons equal in charge to the positive ions ejected. A simple electron gun such as in any TV or Computer monitor will do that. Alternatively use of both + and – ion systems also can maintain electrical neutrality. But no matter what, the ejected beam must be neutralized and neutral or the space craft will become charged and thus attract and collect the very ions it ejected, loosing that thrust as the ions return.
Ion engines are probably going to be used for Mars, and while metal working fluids like Na, Hg, or K now look attractive, but I personally think it a poor environmental practice since the ions will damage satellite electronics and solar cells, thus suggesting Ar as the probable best choice.

MAGNETO HYDRODYNAMIC DUCTS – MHD or J X B

The MHD or J X B engines use a magnetic B field and a J current perpendicular to the magnetic field to accelerate a conductive (plasma) fluid (Right hand rule). The head end of a MHD usually is a thermal arc jet which has some gas seeded with an alkali metal such as Cs, K, or Na to lower the temperature needed to create conductive plasma. Because of the proposed hybrid energy enhancement, Li, which has not been nearly as extensively tested, is suggested as possibly a better seed element. The lower velocity limitation of the exhaust gas to meet temperature limitations of the TAJ is bypassed by further MHD/JXB acceleration to reach the higher velocities desired for Trans-lunar applications. The energy/velocity in the TAJ corresponds to about 1000 seconds, while the MHD raises the velocity to about 5000 or more seconds so the power ratio is about 80% MHD to 20% TAJ, and the TAJ in enhanced systems is 8% chemical 12% electrical.

The key problem with this system is the efficiency of coupling the electrical power into velocity may be lower than that of Ion engines. It also is a more complex system, involving all the problems of TAJ, plus adding a third seed component, and then add to that the MHD follow up system. It is the most complex of all, and thus the lowest expected reliability. Spares, multiple redundancy, and high QC in design and operation can offset this. One question is where to add the seed element – in the initial pre-expansion chamber with the original gas mix, or later? It is usually a small percentage of the total flow, of the order of 1-2%. But adding it later this slows the already high velocity expanded gas, and that is undesirable. Adding it early can do two favorable things, it provides conductivity to add the energy in the TAJ, and there is an almost unknown effect in chemical rockets fuels called tri-propellants that are superior to the well-known bipropellant systems such as H2-LOX or H2-F2. This works by providing heat from the third component while the working fluid is made of lower molecular weight hydrogen.

The comparison of optimal tri-propellants is shown below for standard conditions (1000 psi chamber to 14.7 psi exhaust with shifting equilibrium in the expansion).

H2-O2 = 398
H2-F2 = 409
H2-O2-Be = 450 * (because of toxicity, Be is NOT suggested!)
H2-F2-Li = 435
H2-O2-Li = 415
H2-O2-Na = 405

Thus the "fuel" can carry part of the energy of the final ISP, and the total ISP is slightly enhanced from the seed material by more power than would be present in an ion engine. This is of the close order of 8-10% in the TAJ part. The total power enhancement from more energy must be offset by consideration of possible lower coupling efficiency, but the overall prospects are sufficiently good that this must be closely examined.

One must remember that the thrusters are VERY small compared to the size of the vehicle, and in a manned mission perhaps change and repair of thrusters could be a welcome diversion in a otherwise long and tedious period. Or more likely any repair could be very unwelcome, tedium or not. I feel certain that MHD/JXB systems also will be used for Mars and trans-Mars missions, competing with Ion engines.

SPACE POWER

Inside the Asteroid belt roughly, solar power, from Solar cells works very well. Some where beyond Mars but before Jupiter the strength of sunlight falls below a practical level and we are forced to Nuclear power. The solar power to weight ratio is limited to the efficiency of the solar cells. The 18-20% efficiency now available is near a 24% theoretical limitation so while it is well known, cheaper to build, and safer, it also is limited. Unmanned "freighters" may well remain solar powered for centuries. Nuclear power shows more promise of high power to weight ratios, for higher accelerations, thus in the long run Nuclear power for manned transport is a clear top choice.

There; however, is a trick for solar power here (Boeing Patent expired) well worth knowing. Everyone now makes flat banks of solar cells. But reflectors to concentrate the sun light about 2.6:1, and increase the intensity on the light falling on the solar cells is actually better in all ways, cheaper since there are fewer costly cells, and lighter weight. The Solar cells come in rows alternating with inverted "V" ridges of reflective concentrator strips. The corrugations improve the strength/rigidity of the basic structure. As we go further from the sun, huge very light weight reflective "wings" to catch and concentrate more and more light become necessary.

Going toward the sun a reverse problem of overheating happens at roughly Venus. Solar cells work properly at Venus, but not at Mercury without cooling. The added “wing” concentrators should not be used going inside Earth orbit. Other solar power devices such as Rankine cycle generators or thermoelectric junctions should be considered inside Earth / Mars Orbit as well.

Repeating Nuclear power from fission reactors are highly desirable for even the earliest Mars missions (or better yet if they are ever developed fusion reactors – but the progress, or really lack of progress, there makes these questionable in the extreme). Fusion power was “about 20 years off” “due in 1995” when I visited Oak Ridge in 1965, and it still, 30 years later, is the same “20 or more years off”- or perhaps never. Since when plotted the expected date keeps moving and the due date moves faster than time passes, this is a divergent series not a convergent series. Projections for success depend on individual genius innovation that is not predictable.