Technical Appendix

Rocket specifications

Among the more controversial technical specifications made by Lyndon LaRouche is that of requiring rockets which approach constant accelerations of one gravity throughout their flight to Mars, for health and safety reasons. When combined with efficient payload management—that is, high payload fractions delivered—this means that extremely high power densities are requisite. In fact, this specification means that only fusion-powered rockets could be utilized for full-scale Mars colonization. And for colonization beyond the Asteroid Belt, new, more advanced matter-antimatter technologies would have to be realized.

Efficient payload management and the distance traveled determine the velocity at which propellant is expelled from the rocket engine—the exhaust velocity *W*. The required acceleration, when combined with this mission-determined exhaust velocity, then determines the required rocket power density.

For example, if *W* is the propellant exhaust velocity in meters per second and *A* is the rocket's constant acceleration, the rocket's specific power *P* in watts per kilogram of the rocket's mass is then given by:

$$P=\tfrac{1}{2}WA$$

Table 1 gives approximate values for the required propellant exhaust velocities for various round trip missions assuming the rocket's acceleration *A* is constant and equal to one gravity—10 meters per second squared. The specific rocket power is then determined as shown in the table.

As noted in his June IEEE paper, the Miley plasma focus rocket has a specific rocket power of 50,000 watts per kilogram. The Teller dipole fusion rocket design has a projected maximum specific power of 10,000 watts per kilogram. And existing nuclear fission rocket designs have specific powers of less than 100 watts per kilogram.

Assuming constant accelerations of one-tenth that of gravity—1 meter per second squared—would reduce the specific power requirements for Mars colonization by a factor of 30. (The reduced acceleration also lowers the required mission exhaust velocity.) This would mean that specific powers on the order of 500,000 watts per kilogram would be needed. This is 10 times that projected by Dr. Miley. There exist more advanced plasma focus rocket designs and rocket designs based on laser fusion which meet this specific power requirement or better.

As can be readily shown in elementary terms, nuclear fusion has an upper limit supporting high thrust exhaust velocities on the order of 10 million meters per second. To go beyond the Asteroid Belt will require technologies beyond fusion. This limit for fusion, as well as the limits of other types of chemical and nuclear fission rockets, can be approximated in terms of mass-energy relationships, i.e., $E=MC^2$, where *E* is the energy in joules, *M* the mass in kilograms, and *C* is the speed of light which equals about 300 million meters per second.

For example, nuclear fusion can convert just less than 0.4% of the mass of the reactants into energy. The rocket would be most efficient if all of this reaction energy were converted into a perfectly directed exhaust beam consisting of the fusion reaction products. For nuclear fusion involving the most appropriate reaction, D-He3 (deuterium-helium-3), this would translate into a maximum exhaust velocity *W* of about one-tenth the velocity of light, or about 30 million meters per second. But inefficiencies in terms of fuel burnup, recirculating energy needed to maintain the reactor, and exhaust divergence reduce this *W* to below 10 million meters per second.

Nuclear fission converts four to five times less of the mass of the reactants into energy. This would translate into maximum exhaust velocities and impulses less than half that of nuclear fusion. But nuclear fission technology necessarily involves even greater inefficiencies than fusion technology, such as reactor shielding and lower operating temperatures. Therefore, the ultimate parameters for fission are more on the order of 10 times less than that of fusion, or 1 million meters per second exhaust velocity.

From these elementary considerations it can be easily seen that human flight requiring near 1 g constant accelerations and the economic requirement of high payload fractions would seem to preclude nuclear fission as a workable technology for Mars colonization. Furthermore, nuclear fusion would meet its limits just beyond the Asteroid Belt. To go farther would require technologies with greater mass-energy conversion efficiencies. For example, through pair production we could generate significant quantities of antimatter. And when antimatter is mixed with an equal mass of ordinary matter, all of the mass is converted into energy. If we could find a means or a system for storing this antimatter, the

TABLE 1
Requirements for 1-gravity flight

Mission	Exhaust velocity W	Specific rocket power
	(meters/sec.)	(watts/kilogram)
Mars	3 million	15 million
Asteroid Belt	7 million	35 million
Saturn	16 million	80 million

TABLE 2
Rocket fuel energy outputs
(joules per kg of fuel)

Chemical	15-26 million
Cryogenic hydrogen	220 million
Metastable helium	460 million
Nuclear fission	80 trillion
Nuclear fusion	350 trillion
Antimatter	90,000 trillion

antimatter mass could then be converted at 100% efficiency into energy. And given the inherent, higher efficiencies for readily converting antimatter gamma ray energy into directed exhaust beams—in fact relativistic directed energy particle beams as the rocket exhaust—antimatter offers the prospects of an improvement over fusion by more than two orders of magnitude (see **Table 2**).

Rocket equations

If we make the approximation that mass associated with gravitational action is equivalent to mass otherwise determined by kinetic action—so-called inertial mass—then the change in the motion of an object can be represented as resulting from a external force acting on that object:

$$F=MA$$

where *F* is the applied force in newtons, the mass *M* is in kilograms and the "change in the motion" of mass *M* is given by the acceleration *A* in meters per second squared. The acceleration *A* is the rate of change of the velocity with respect to time, i.e.:

$$A=dV/dt$$

where dV is the increment of change in the velocity *V*, given in meters per second, and *dt* is increment of time *t*, given in seconds, during which this change in velocity takes place. And therefore:

$$F=M(dV/dt)$$

In a rocket, though, no external force acts on it. Therefore, in order to achieve a change in its velocity, its mass must change. The force due to this change in mass will be proportional to the velocity at which some of the mass of the rocket is ejected and the net amount of mass ejected. This ejection of mass will result in the rocket receiving an impulse in a direction opposite to that at which mass is ejected. The external force is equal to zero, but there is a change in the velocity of the rocket:

$$0=F=W(dM/dt)+(M-dM)dV/dt$$

where *F*, the external force, is zero, *W* is the rocket exhaust velocity, the velocity at which an increment of mass ***dM*** is ejected from the rocket in an increment of time ***dt***. The exhaust velocity ***W*** is taken to be constant. The mass ***M*** of the rocket is no longer taken as constant, but changes to reflect the mass *dM* ejected as propellant. Multiplying through and placing W(dM/dt) on the left side of the equation we have:

$$-W(dM/dt)=M(dV/dt)-(dMdV)/dt$$

The minus sign in front of *W* reflects the geometry in that *dV*, the change in velocity of the rocket is oppositely directed to that of the rocket exhaust. The last term of the above equation can be taken as neglible, i.e., *dMdV* can be taken as being zero, since it is the product of two small increments. Taking this into account we arrive at:

$$-W(dM/dt)=M(dV/dt)$$

Multiplying through by the time increment *dt*, we have:

$$-W(dM)=M(dV)$$

This can be rearranged into:

$$dM/M=-dV/W$$

Integrating both sides we arrive at (1):

$$\ln(M)+K=-V/W$$

where ln is the natural logarithm and *K* is a constant of integration, *M* is the actual mass of the rocket at any given time and *V* is the velocity of the rocket. *W* is the constant exhaust velocity. If we take the initial rocket velocity as zero, then the equation becomes:

$$\ln(M_o)+K=0$$

or

$$K=-\ln(M_o)$$

where M_o is the initial mass of the rocket—the takeoff mass. Substituting this value of the integration constant into equation (1) we have:

$$\ln(M)-\ln(M_o)=\ln(M/M_o)=-V/W$$

and therefore (2):

$$e^{\ln(M/M_o)} = (M/M_o)=e^{-V/W}$$

or

$$(M_o)/M=e^{V/W}$$

where *e* the base for the natural logarithm. The mass delivered by the rocket to its destination can now be determined by (2). If V_f is the final rocket velocity,* then the payload mass M_p can be found by substituting V_f for V and M_p for M in (3):

$$M_o/M_p=e^{V_f}/W$$

From (3) we see that the ratio of the takeoff to the payload mass is an exponential function of the ratio of the final rocket velocity to the propellant exhaust velocity. If we wish to have a large payload relative to the takeoff mass, then the ratio of the final rocket velocity to the exhaust velocity must be small as possible as seen in **Table 3.**

Quite clearly, rocket performance in terms of payload delivered is strongly determined by the ratio of the final rocket velocity to the exhaust velocity V_f/W. The smaller this ratio, the greater the payload delivered. The final rocket velocity is directly determined by the particular space mission. And in order to deliver a significant fraction of the "takeoff" mass as a payload, the rocket exhaust velocity W should be almost equal to this final rocket velocity V_f. Ordinary chemical rockets operate with high V_f/W ratios and therefore deliver relatively small payloads. (In fact, it is this inefficiency of chemical propulsion which leads to the requirement of multistage rockets for space missions).

Actually, V_f is not really the final rocket velocity. This actual, final velocity is determined by how we maneuver the rocket and the particular orbits we proceed through in space. Our V_f is really the rocket's operational final velocity—the velocity we would have if we kept the rocket accelerating in the same direction throughout the mission. It is actually referred to usually by rocket designers as the mission ΔV.

If we turn to the general requirements of human space flight and colonization, it is rather amazing how precisely these rather elementary considerations determine a distinct series of rocket technologies with respect to regions of the Solar System. Human space flight requires vehicles that undergo a relatively constant acceleration approaching that of one gravity on Earth—1 g with

$$dV/dt=g=10 \text{ meters per second squared}$$

This requirement of near 1 g for human space flight is dictated by both the need to minimize the time that passengers are exposed to cosmic radiation and the maintenance of the biological health of average human beings.

True colonization means that the space colony is economically integrated with the economy of Earth. This requires that the rockets have large payload-to-takeoff ratios. The distance traveled between Earth and a particular region then determines the required V_f's—ΔV's. And this determines the required rocket exhaust velocity W, which must generally be about the same magnitude as the mission V_f.

As shown above, the exhaust velocity W is directly determined by the basic scientific properties of a particular technology. Colonization of Mars and operations within the area defined by the Asteroid Belt require nuclear fusion technologies. Colonization beyond this region, such as Saturn's moon Titan, require more advanced matter-antimatter technologies.

Another important rocket parameter, the rocket's specific impluse I, can be simply derived by dividing the exhaust velocity by g. That is: $I=W/g$. W is given in meters per second and $g=10$ meters per second squared, therefore I is given in units of seconds.

TABLE 3
Rocket performance

V_f/W	M_o/M_p	M_p/M_o
10	22,026	.0000454
5	148	.00674
1	2.72	.368
0.5	1.65	.606
0.1	1.10	.905

*To arrive at the V_f mission requirement the distance D in meters to the destination is determined. Given that the following hold for 1 g constant acceleration:

$$D=\tfrac{1}{2}gt^2$$

$$V=gt$$

then the total V_f of the mission for a round trip to the destination and back can be found in the following manner. Assume that half of the trip to the destination is spent accelerating toward the destination at 1 g. Then the second portion of the trip to the destination is spent decelerating to arrive at an orbital velocity appropriate for the destination. But for determining the rocket performance, it does not matter whether the rocket is accelerating or decelerating. Therefore, the time t required for the trip is simply determined by calculating the time needed for the accelerating portion of the trip to the destination and multiplying this time fourfold for the entire round trip time. Then V_f of the mission is found by multiplying g times this round trip time:

$$t=4[2(D/2)/g]^{1/2}$$

$$V_f=gt$$

or

$$V_f=4(Dg)^{1/2}$$

Mars is on the order of 56 billion meters from the Earth when the two planets are in nearby portions of their orbits and therefore:

$$V_f=4[(5.6\times10^{10})(10)]^{1/2}=3\times10^6 \text{ meters/sec.}$$

And taking $W=V_f$, we would find that the specific rocket power P in watts per kilogram is found by:

$$P=2(D^{1/2})(A^{3/2}).$$

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