Papers on the Lunar Settlement

Engineering 8 : Launch Vehicles

0. Introduction This paper accompanies number 03-01 on the logistics of space access. It is intended to present technical analysis of relevant concepts, specifically the expendable multistage rocket for direct ascent, the single-stage reusable terrestrial orbiter, two-stage reusable and partially reusable configurations, the lunar rocket, and the space gun. At this early date, however, nothing more than approximations and general considerations can be furnished. While specific numerical values are indicated in certain places, it must be understood clearly that few of these have any validity ; they are presented in order to indicate trends of expected behaviour.

1. Expendable Rocket For the expendable rocket to be useful, it must be very large. Two factors are at work here : first, *ceteris paribus*, the larger the rocket, the larger the useful mass fraction ; second, expendable rockets normally permit only infrequent launches, so each one must deliver the largest possible payload.

The detailed description of such a rocket requires analysis of technology and usage, but the rocket capabilities are liable to have an effect on the payload just as the payload affects the rocket, so it is not improper to begin here with a broad outline.

A few arbitrary choices and rough estimates are necessary inputs to the design process. Let us propose a launch mass of one thousand tonnes, or one third that of the *Saturn V*. Further, suppose that the propellants are liquid oxygen and liquid hydrogen, that the engines realize an effective exhaust velocity of 4000 m/s, and that the dead mass of each stage is 10% the fuel mass, since the larger lower stages will be more structurally efficient but must bear the load of the upper stages.

If the required velocity increment for direct ascent, without an intermediate orbit, is taken as the sum of the escape velocities of Terra and Luna, 13.6 km/s, the necessary mass ratio with exhaust velocity 4.0 km/s is 30. This sets the maximum delivered mass at 33 t, and indicates that several steps are required. As a little calculation will show, the overall efficiency of a step rocket is improved by using low mass-ratios in the lower steps. We may select a tristage configuration, with step mass ratios of 2, 3, and 5.

The mass of the first step, then, is 550 t, of which 500 t is fuel ; the second, 330 t, with 300 t fuel ; and the third is the lunar-landing stage, 120 t with 96 t fuel. If the first two steps are the same diameter, the third may be built as a nose-cone ; its usable load should be increased by designing the structure for recycling, since it will not be reused.

Of course, 4.0 km/s is the vacuum exhaust velocity, and the rocket must launch from within the terrestrial atmosphere. To account for this, the effective exhaust velocity may be reduced 20% for the first stage only, and the second and third considered as operating in vacuo. The mass ratio of 2 with 4.0 km/s becomes 2.4 for 3.2 km/s and the same final velocity, and the mass of the first stage increases about 40%. The recalculated launch mass is approximately 1250 t.

A better mass-ratio could be achieved with fluorine as oxidizer, less because it is more energetic than oxygen than because of the reduced proportion of hydrogen, as the hydrogen tank is a major contributor to the dry mass. Fluorine, however, is not a well-developed rocket propellant. There are no operational fluorine-hydrogen rocket engines, and the time needed to develop them is probably too long for near-term use. There is a substantial cost for the substance, and the cloud of hydrofluoric acid emitted by the rocket would be toxic, and probably deleterious to the spaceport physical plant.

The difficulty of employing even the hydrogen-oxygen combination, which is far better established now than in 1962, is considerable. The largest available hydrogen-burning engines are in the 2.5 MN class, and the rocket contemplated would require 4 to 8 such in order to lift at all. The *M-1*, hydrogen equivalent to the 7.5 MN class kerosene-fuel *F-1* engine of *Saturn V*, seems called for. Alternative design approaches are possible, of course, such as the use of strap-on boosters, preferably recoverable.

The expendable launcher is very much a known quantity. Once a design approach is identified, the path from design, through subsystem testing and assembly, to use is relatively clear. It is true that modern expendable rockets have design and proving cycles measured in decades, but this is typical of aerospace projects today, and may be said to have more to do with industry structure than with practical requirements. *Saturn V*, the largest rocket ever put into service, was delivered in less than 5 years by a team of experienced rocket men.

2. Single Stage Reusable Rocket It is possible to produce an expendable rocket capable of reaching terrestrial orbit in a single stage, but such a thing would be of little value, having less payload capacity than a comparable step rocket and no more availability. The reduction in complexity is substantially offset by the loss of redundancy from not being able to substitute a malfunctioning stage. The single stage concept is probably only useful for a reusable craft, which can carry numerous small payloads over its lifespan. No such craft yet exists, although a small-scale prototype, the McDonnell-Douglas *DC-X*, has been built.

Using oxygen and hydrogen as propellants, and aiming for a low orbit with a velocity of 8.0 km/s, the required mass ratio is 7.4 with the vacuum exhaust velocity, and 12.2 with the 20% reduced exhaust velocity. The truth lies somewhere between these two figures, since the rocket is moving upward, and thus increasing its exhaust velocity, all the time that it is burning.

Two observations may be made on this mathematical basis. The first is that the single-stage orbiter requires a high altitude launch, with reduced atmospheric pressure, in order to perform decently. Quito will make a good home port for such craft.

Secondly, only a very large single-stage vehicle is viable. For the hydrogen-oxygen propellant combination, with exhaust velocity reduced by 10% to 3600 m/s, intermediate between altitude and sea level, the mass ratio is 9.22. Such a rocket, having a launch mass of 100 t, would deliver 10.8 t to orbit. The *S-IVB* rocket stage of the 1960s, which was not sufficiently rugged to survive reentry or be reused, had a dead mass fraction of about 1/10. If superior materials and design were to replicate this figure for the rocket contemplated, its payload to orbit would be perhaps 800 kg. Without practical experience, it is not clear whether this figure can be considered representative, but such a vehicle would be little more than an expensive toy.

A brief investigation may be made of the use of a denser fuel than hydrogen. The fraction of mass required for the fuel tanks would decrease, but the required mass ratio would increase. The following analysis assumes that the vehicle is the frustum, altitude 2*r*, of a right circular cone of altitude 3*r*, a compact shape of the sort typically used in such designs. The non-fuel mass is treated as distributed uniformly over the surface in the form of a titanium skin, nominal density 4510 kg/m^3. The thicker this fictitious layer is, the easier the problem of inclosing the required volume, and the more mass available for other purposes.

At approximately the proportions used in the US Space Shuttle, the bulk specific gravity of the hydrogen-oxygen combination is 0.37. The fuel mass of a vehicle with a gross liftoff mass of 100 t will be 89.2 t, occupying a volume of 241 m^3, and requiring a surface area of 220 m^2. The remaining mass being 10.8 t, the skin areal density is 49.3 kg/m^2, which is equivalent to a thickness of 1.09 cm of titanium. For methane, with a vacuum effective exhaust velocity of 2950 m/s, again reduced 10% to 2660 m/s, the mass ratio is 20.4. At a bulk specific gravity of 0.75, the 100 t vehicle has a volume of 127 m^3 and a surface area of 144 m^2. With a remaining mass of 4.91 t, the areal density is 34.2 kg/m^2, or 0.759 cm equivalent. It does not appear from this that an advantage is to be gained by the use of denser fuel.

The use of fluorine with hydrogen is called for in this connexion if any, but added to the usual drawbacks are the wearing out of the tanks and engines, and danger to the service crew. The only quarter from which relief may obviously be looked for is sheer bulk. A 1000 t vehicle with the same characteristics as the 100 t hydrogen model described above would deliver a payload of 8 t to orbit, and could be economically successful if it had a sufficiently brief turn-around cycle. With increasing size, also, the square-cube law operates to improve the structural and payload mass fractions. Applying the analysis used above to compare fuels to the 1000 t hydrogen vehicle, the areal density is 106 kg/m^2, or 2.36 cm equivalent. It rather appears that the most successful spaceship will be the largest, and in this respect, it is very like the airship.

In any case, it appears that the single-stage rocket presents problems which are not clearly understood. The operational principles of the Luna Project suggest that the appropriate response is to try it. It is unlikely that the first prototype, even if it achieves orbit, will carry a payload capable of supporting early Project activities, and the design cycle to develop from zero a model capable of such support is probably more than five years. Since the sustaining phase will probably occupy at least three years after the landing of the first party, the development of reusable concepts to become operational in this period, taking over from expendable launchers, must be pursued.

3. Reusable Combinations Rather

than design a single vehicle to meet all the challenges involved in going from ground to orbit, it may prove easier to divide the functions among separate vehicles. This can take various forms, including the balloon-rocket combination or "Rockoon", and groups of winged rockets. The discussion below addresses the case in which two stages are used, the lower being purely an aircraft.

As seen in the discussion of the single-stage orbital rocket, the principal problems are the large velocity increment required, and the impairment of the rocket effective exhaust velocity by atmospheric pressure. If the rocket can be carried near the limit of the sensible atmosphere before launching, the second problem is dealt with, and if the carrier can impart velocity as well as altitude, the first can be ameliorated. An additional advantage of high-altitude launch is that it can be more nearly horizontal, in the direction of the intended orbit, than a ground launch.

These gains come at the cost of greatly increased complexity for the overall system, due to the necessity of building, maintaining, and operating the radically different vehicles, separately and as a unit. If the restriction of functions in the individual units reduces the difficulty of the development process more than the need for interoperability increases the difficulty, the development of the two-stage system is justified. Two important cases appear. First, if several different upper stages can be used with the same booster, versatility can be increased over developing an entirely new launch system for each function. Second, if existing hardware can be adapted, the two-stage system may have an advantage in development cycle time over the single-stage system.

4. The Booster Let us examine, as an example of such adaptable hardware, the North American *B-70 Valkyrie* hypersonic bombing aeroplane. This remains, after forty-five years, one of the largest and highest-performing aircraft ever constructed. In terms of speed, altitude, and lift, it appears well adapted to serve as the booster component of an operational prototype two-stage-to-orbit combination.

Performance figures for the *B-70* vary somewhat. Two developmental units were built, both different, and a third vehicle which was to have been the prototype of the production model was scrapped on the ways when the project was cancelled. In round figures, the takeoff mass was 250 t, 110 t being disposable lift (mostly fuel), the engine thrust was 800 kN, and the maximum speed achieved was 920 m/s at 22 000 m altitude.

Some improvement may be possible to the design. Since the second prototype *XB-70* was destroyed in a crash, only the lower-performing first unit now exists, and it is a museum piece. Even if it were to be had, the kind of inspection and renovation necessary to make it ready for flight, not to speak of the modifications required for carrying an upper stage, could well prove more difficult and time-consuming (if not expensive) than building a new example of the third model from the original plans. It is not unreasonable to suppose that some design alterations could be made to better suit the aircraft for its new role.

Structural hard points and other facilities for carrying and launching the upper stage are a necessary addition. Control systems could be replaced with more modern equivalents, and the volume intended for the bomb bays may prove useful as top-up tanks for the rocket's cryogenic fuel, some of which will boil away during the ascent to the launch point. In general, however, the important modifications will be in the area of propulsion, with the intent of increasing thrust, or reducing vehicle mass, or both.

A brief tradeoff analysis may be attempted, to examine the effects of increased speed as against increased load capacity in the booster. Three velocities are taken : 900 m/s, representing realized performance of the *B-70* ; 1000 m/s, representing a reasonable improvement with increased thrust ; and 1200 m/s, representing an extreme probably beyond the achievable performance. It may be supposed that the airframe design is valid only out to about Mach number 3.5. Two conditions of loading are examined : an upper stage of 60 t, representing reasonable performance with a light fuel load, recognizing that the booster need only make a dash to its maximum speed and altitude and then return to base ; and an upper stage of 100 t, requiring some combination of increased thrust and lightening of the aircraft. The target parameter is on-orbit mass.

Mass ratios for a final velocity of 8.0 km/s, assuming a hydrogen engine with vacuum performance at the launch altitude, are : for 900 m/s launch, 5.90 ; for 1000 m/s launch, 5.76 ; for 1200 m/s launch, 5.47. For the 60 t upper stage, the corresponding final masses are 10.2 t, 10.4 t, and 11.0 t. This, of course, is false precision, but it represents clearly the scale of performance variation with respect to booster speed. For the 100 t upper stage, the results are 16.9 t, 17.4 t, and 18.3 t, the least favourable of which is handily superior to the most favourable 60 t case. Considering that the larger rocket can be expected to have the larger payload mass fraction, it appears desirable to concentrate on increasing payload capacity.

With a fixed lift-to-drag ratio, aircraft load capacity can be improved either by reducing mass or increasing thrust. Turbojet thrust-to-mass ratios have improved considerably since the *B-70* was designed, but it is not clear that modern units will maintain their performance at Mach number 3 and above, even with the assistance of variable intake nozzles. More radical modifications, such as employing methane rather than kerosene for fuel, belong properly to the design of a purpose-built booster stage. In the event that an engine were found having the same size and mass as the original, the question of powerplant configuration would depend on high-speed thrust. Four engines having a unit thrust of 200 kN at low speeds, for example, but considerably less at Mach 3, could be installed on an "over and under" basis. High speed power would be provided by ramjets, which are lightweight but generate no thrust at rest, in the remaining two bays.

Further tradeoffs are possible, since supersonic aircraft are unlike subsonic aircraft in requiring full power at speed rather than only at takeoff, and considering that a spaceplane will not be expected to operate except from specially-prepared facilities. A turbojet complement of less than 800 kN might be installed in tandem with ramjets, and the additional impetus needed to lift off supplied by a catapult.

This process can, however, only be taken so far. The combination of pure ramjet engines with catapult launch would apparently require accelerating the aircraft to above Mach number 1 at ground level, presenting considerable aerodynamic challenges even at a high-altitude launch site. Somewhat better, if top-up tanks were built into the booster, would be firing the orbiter rocket for takeoff thrust, especially if its exhaust were channeled through an ejector to draw air into the ramjets. In either case, however, the booster would be unable to fly back from a remote landing site such as might be reached in an aborted launch, and there is no other way to transport such an enormous thing intact.

The best approach appears to be to begin with a minimal modifications. The aircraft can be flown alone and with a dummy upper stage, to gain experience, before the use of a live upper stage is attempted. Based on experience with this baseline booster, a model with further modifications can afterward be built and proven. Since the base design already exists, it should be possible to begin flying within three years of committing to the configuration, and if it proves viable a second-generation configuration should be possible within three additional years, in time to support the continuing settlement phase.

5. The Orbiter The above has taken the upper stage as given, without examining its design, and assuming a minimal effect on the high-speed aerodynamics of the booster. The broad, flat back of the *B-70* delta wing appears suitable for hosting a flat-bottomed parasite rocket, such as a lifting body of the *FDL-5* design, but the matter deserves further examination. Experience with the *A-12*/*D-21* combination shows that hypersonic separation of “piggyback” aircraft is possible, although demanding considerable care, and that the parasite may be rather large without producing excessive drag or aerodynamic interference in supersonic flight.

The lifting body has an advantage in structural mass fraction over a winged aircraft, and its density empty would be quite low, reducing the problems of atmospheric heating on reentry. A titanium-alloy airframe could probably be employed with minimal heat shielding. If hydrogen fuel were used, a single engine in the 1.5 MN thrust class, such as the *J-2* type, could supply the required thrust for launch masses at least up to 100 t.

Versatility is above alleged to be an advantage of the two-stage spacecraft. It is worth examining some possibilities presented by modifications of the upper stage. The figures given above for the minimum launch speed are 10.2 t final mass for the 60 t orbiter, and 16.9 t for the 100 t model. Assuming that the dry mass is 1/8 of the total launch mass in both cases (somewhat more than for a shuttlecock-shaped vehicle), the possible useful loads are 2.7 t and 4.4 t respectively.

The above estimates are extremely crude, but suggest that even the smaller craft could carry a man, in a pressurized compartment with life support for a reasonable period, and maneuvering fuel to make a controlled reentry. This performance would suffice for demonstration and training purposes. The larger might carry more than one man, and some payload as well, perhaps in the form of additional maneuvering fuel. If the same basic designs were used for unmanned craft, a smaller factor of safety would be allowable, and the useful load could be correspondingly increased. Considering the probably increased difficulty and danger of air-launching an unpiloted vehicle, the 60 t model may not be worth bothering with. The payload of close to 5 t possible with the 100 t type, on the other hand, seems large enough to justify the effort, given the frequent launches possible with a reusable system.

The same type of analysis may be applied to these vehicles as was done above for the single-stage orbiter. Approximating the lifting body as a regular tetrahedron, and using figures for hydrogen and oxygen as before, the required volumes for 49.8 and 83.1 t are 135 and 226 m^3 respectively, and the corresponding surface areas 217 and 306 m^2. The remaining masses of 10.2 and 16.9 t give areal densities of 45.7 and 55.4 kg/m^2, equivalent to 1.0 or 1.2 cm titanium. As with the single-stage orbiter concept, the advantage of smaller structural mass fraction using a denser fuel is swamped by the increase in mass ratio, 11.1 using methane and oxygen with a vacuum effective exhaust velocity of 2950 m/s. For the 100 t rocket stage, the volume is 122 m^3, the area 203 m^2, the remaining mass 9.0 t, the areal density 44.5 kg/m^2, and the equivalent titanium thickness 0.99 cm.

All of these figures are comparable to the single-stage figures, indicating that the two concepts are equally viable. The criteria of choice appear to be time and capability. It is expected that the two-stage solution will be developed more rapidly than the single-stage, but will remain limited by the capacity of the booster aircraft.

The development programme will be eased by the interchangeability of upper stages, and the fact that both stages can fly to a landing. If several are built, even with a single booster, modifications can be made without withdrawing the whole system. Flight tests and pilot training might be conducted by dropping an empty airframe, or a flyable mockup, from a large subsonic transport, and by launching from the ground (perhaps with a partial fuel load) if sufficiently strong landing gear is provided. Since the major danger is a collision following a botched air launch, supersonic experience before mating the components is a necessity. As with the single-stage solution, reusability allows development to proceed by incremental steps in the course of operations with working ships. Since the Project cannot remain dependent upon expendable launch vehicles, even if they are required for the initial phase, the development of both approaches is indicated, in order to insure that one will work.

6. Partial Reusability The booster aircraft of the two-stage reusable combination might be used in another way. If a reusable rocket benefits from an improved mass ratio when launched from high altitude and high speed, an expendable rocket will also benefit.

The usefulness of combining an expendable upper stage with a reusable lower stage is a matter of availability. The same payload can be lifted to orbit using a significantly smaller rocket, or conversely the same rocket can lift a larger payload, if it is launched from the air. The question is whether this fact will permit substantially more payload to be orbited per unit time than ground launch, such as by permitting the use of readily-available expendable rockets beyond their rated performance, or allowing the use of simpler (perhaps solid-fuel) units having less efficiency but more convenience. The present U.S. Space Shuttle is partially-reusable, but (in effect) discards the bottom stage rather than the top stage, and so does not present a model for the concept mooted. The topic deserves consideration, as a concept which could begin delivering payloads to Luna as soon as the booster is proven, but nothing definite can presently be said.

7. Lunar Rocket The purpose of the rocket fabricated from lunar materials is twofold. First, it serves as a mode of lunar global transportation. Second, when launched by the space gun, it is used for maneuvering in lunar and terrestrial orbits. Its characteristics are defined largely by the first purpose and by the requirement that, since it will be fabricated in large quantities, it should not use any scarce materials.

A transportation mechanism can be characterized broadly by load and range. If the lunar rocket is to be useful, it should at least be capable of carrying a man, with supplies for several days and various tools, or a significant cargo. As a round figure, a suitable payload might be 1 t. Since Luna is an airless body, rocket braking must be provided, and the total velocity increment is equal to twice the velocity required for the trajectory. To go, brake, return, and brake again would require a velocity of four times the initial.

The question of range becomes important. A rocket intended to travel one quarter of a planet's circumference requires about nine-tenths of orbital velocity ; half that velocity gives a range less than one tenth the circumference. Accordingly, the rocket will be designed to reach orbit and make a soft landing. This will enable one-way payload delivery anywhere on Luna, or two-way travel within a reasonably large zone. As the zero-altitude circular velocity is 1680 m/s, the total velocity of the rocket is 3360 m/s. This is less than the exhaust velocity attainable with oxygen-hydrogen fuel, and such a rocket would require a mass ratio of only 2.32. The fuel actually to be used, however, will have a poorer performance.

8. Performance Estimate Among conventional rocket fuels and oxidizers, only aluminum and oxygen are plentiful in the lunar environment. These two substances do not form a conventional propellant combination. Aluminum is normally encountered in solid rockets, as a dispersed phase in an organic binder, accompanied by perchlorate oxidizer, while oxygen is typically used as a liquid and paired with a liquid fuel.

It does not appear practicable either to feed a liquid-fuel motor with molten aluminum, or to encapsulate oxygen in solid aluminum. Accordingly, the rocket will be of the hybrid type, using a solid fuel grain with a separate oxidizer. This type of engine has the advantage over the pure solid that it can be throttled and restarted, although not controlled so finely as the liquid-fuel motor.

The principal difficulty presented by aluminum as a rocket fuel is that the product of combustion is a refractory solid, having a tendency to adhere to the parent surface. The resulting problems help to give shape to the final design.

First, a solid product of combustion does not expand and escape from the nozzle. Thus, left to itself, such a burning process will not produce any momentum transfer. Second, if the product remains at the surface, no new fuel area is exposed, and the reaction is rapidly extinguished. Therefore, it seems reasonable that the rocket should operate with a substantial excess of oxygen. This will absorb some of the reaction heat and serve as a working fluid ; it should also help to erode away the oxide from the metal surface, in the form of minute particles (hopefully!) which will join the exhaust stream, participating in momentum transfer while not choking the combustion process.

It appears necessary to reduce the thermal conductivity of the fuel grain. Unlike common solid propellants, massive pure aluminum is highly conductive, and if the loss of heat through the combustion surface were not sufficient to extinguish the reaction, there would be a danger of bodily melting and sloughing, with consequent failure of the rocket.

Two approaches may be identified. The first is to reduce the thermal conductivity of the bulk metal, by alloying it with some other flammable metal. It is a property of alloys that a small admixture even of one highly-conductive metal into another drastically reduces the thermal conductivity. Also, alloys tend to have lower melting points than the parent metals. Calcium and magnesium are fairly common in lunar rock and might be used. Secondly, if the grain is made not from solid metal, but from a consolidation of small particles, such as sintered powder with moderate compaction, heat transfer between individual particles is impeded, and the whole mass has a much higher insulating power than before. As long as the mass is not sufficiently porous to allow the rocket gasses to escape through the walls, inclosing the grain with a case may be avoided, improving the structural mass fraction.

The result of these measures should be that the oxidation of the metal surface will cause melting and even local boiling immediately below the surface, shedding the oxide into the oxygen stream, and exposing new surface area to combustion. The bulk of the rocket, meanwhile, will retain its integrity. Firing the rocket will probably require squibs of thermite, and to allow restarting these might be mounted in a mechanism like a revolver barrel, at the end of the oxygen fuel line. A thermite squib would be rotated into place and fired, and the oxygen then turned on to blow the burning material into the thrust chamber, spattering it on the walls and igniting the propellant charge.

This is no more than a conceptual design, but with sufficient assumptions a simple thermodynamic analysis can be performed to evaluate feasibility. The fuel is chosen to be an alloy of ten weight percent calcium in aluminum, burned with a fifty percent stochiometric excess of oxygen. The fuel is taken as in the solid phase, the oxidizer as gas, and the products of combustion as solid. As the grain is a metal, having inherent structural strength, and the exhaust is vacuum, some liberty is available in the selection of chamber pressure and expansion ratio. For convenience, the grain geometry is supposed to be a constant-area type, although constant thrust could be maintained by throttling the oxygen supply.

The chamber pressure is selected to be

2.0 MPa, and the expansion ratio 20. The solid products of combustion are assumed to occupy zero volume, and to be intimately mixed with the excess oxygen gas, being expelled through the nozzle. The momentum transfer is accordingly the same as if the excess oxygen had been heated indirectly, but the exhaust velocity is less by a factor of 5.4.

A thermodynamic analysis proceeds as follows. The oxygen is introduced into the chamber by pumps, in the liquid state and at chamber pressure. Heat from the reaction vaporizes the oxygen, and the stochiometric quantity immediately combines with the fuel metal. Heat is then transferred from the products of combustion to the excess oxygen until an equilibrium temperature is reached. The mixture of oxygen and combustion products then expands isentropically through the nozzle.

The overall exhaust velocity calculated by this method is approximately 800 m/s. This is a very poor figure, requiring a mass ratio of 67 for the velocity desired. The reliability of this figure, however, is also extraordinarily poor due to the very approximate methods used to derive it, to the fact that the calculated chamber temperature is over 8000 K, a condition under which chemical equilibrium must be considered, and to the fact that transfer of heat from the solid phase to the gas phase during expansion was not considered.

The physics of the problem suggest that a much larger excess of oxygen is desirable, both because of the decrease in average molecular weight of the exhaust, and because at the high chamber temperature quoted only 12% of the combustion energy can be transferred out of the solid. At a lower equilibrium temperature, more heat would be present in the gas phase, and thus available for propulsion. Other propellant compositions must also be investigated. The burning of a series of test motors is clearly called for, in order to observe the actual behaviour of the device.

9. Space Gun Under lunar conditions, with low escape velocity and no sensible atmosphere, the possibility arises of treating space-launch payloads as projectiles, accelerating them by external means near ground level. This is the purpose of the so-called space gun.

Obviously, a gun in the strict terrestrial sense is not what is required. Something capable of launching relatively large objects at relatively gentle accelerations is called for, if the device is to be used for manufactured goods or *a fortiori* men. Electromagnetic machines of various types have been proposed.

The "rail gun" requires the projectile to be suspended in a magnetic field and conduct an electric current in the normal direction, and is limited by several factors, including destruction of the electrical contacts. A type of "coil gun," referred to as a mass driver, is incorporated in the O'Neill colony proposals for the purpose of delivering lunar payloads to cislunar space. This machine consists of a long line of toroidal electromagnets, each of which is triggered to conduct a pulse of current by switches keyed to the passage of a "bucket" in which the payload is carried. This bucket is fitted with strong permanent magnets, and must be braked to a stop at the end of the gun.

A second type of coil gun relies on

polyphase alternating current. The coils are wound in such a manner as to set up a traveling magnetic wave within the tube which they form. The result is that a conductive body placed at the muzzle is, by self-induction of currents in the skin, held firmly at the centre of the bore and moved toward the other end. The velocity of the body asymptotically approaches the phase velocity of the traveling wave, with an acceleration dependent principally on its mass and surface electrical properties. The energy efficiency of this type of gun can be improved by switching circuitry which feeds energy only to coils near the projectile, but its operation does not require such. If the projectiles are sufficiently uniform, the coil spacing can be increased toward the muzzle end in accordance with the velocity profile, improving the acceleration without requiring higher power frequencies.

All of these so-called guns are special cases of the electric motor, and with any kind of competent design should be able to achieve a conversion efficiency of electrical to kinetic energy of 50%. On this assumption, a gun expelling its projectile at lunar escape velocity, 2380 m/s, will require an energy input of 5.7 MJ/kg. For a projectile of 5 t mass, accelerated at 200 m/s^2, which even the most delicate manufactured goods should be able to withstand, this translates to a power of 2.4 GW for 12 s. With the same mass, but a lower acceleration of 50 m/s^2, suitable for men, 560 MW is required for 48 s.

Even the higher-acceleration gun requires a bore length of 14 km, and although the excavation of very deep shafts has been mooted in connexion with the construction of Luna City, it seems reasonable that the gun should be laid out horizontally. A vertical gun located anywhere near 50° N would have its muzzle pointed well out of the ecliptic plane. An oblique shaft directed at Terra would not be radial, and would send its projectiles somewhere else. The proper obliquity for putting packages into terrestrial orbit, or trajectories around Terra for other destinations, could be determined, but appears unuseful.

Most uses would require some rocket power on the projectile, and this once granted, a horizontal gun has advantages. Firstly, by varying the excess velocity, a variety of hyperbolic trajectories is available. The combination of a slight initial velocity deficiency and properly timed rocket thrust will allow access to escape trajectories not directly available. Again, the horizontal gun allows placing the projectile into lunar orbit with a small apoapsis rocket. The resulting orbit would be of high inclination, but this does not appear to be a disadvantage. It would allow access to a large fraction of the lunar surface without plane changes, and increase the achievable variety of transfer orbits.

Lunar orbit is essentially at infinity as far as Terra is concerned, and the energy required for a plane change at infinity is zero. Accordingly, a terrestrial orbit of any desired inclination can be reached from an equatorial lunar orbit. The advantage of a high-inclination lunar orbit is that, by performing the lunar escape maneuver at a carefully selected moment, any inclination of terrestrial orbit can be reached by firing the rocket along the line of flight. To reach non-equatorial terrestrial orbit from equatorial lunar orbit would require either firing at an angle to the line of flight, or a second burn after the escape maneuver to perform the plane change. The same applies, *mutatis mutandis*, to high-inclination escape trajectories.

Accordingly, the horizontal gun, laid out along a parallel of latitude, appears perfectly suitable. In practice, a slight elevation of the muzzle may be required in order to avoid striking the terrain, but this will result in only minor operational changes. As lunar development increases, it may prove economical to construct additional guns, at other azimuths, for point-to-point transportation, although the controls problem of catching a projectile in a gun muzzle appears sufficiently difficult that rocket braking will still be required.

A. Conclusion The choice of space-launch techniques and vehicles must necessarily have a major effect on the execution of the Luna Project. Options have been described, and briefly discussed, but none of the approaches discussed represents an available, off-the-shelf product. Without sufficient data to make definite choices, the only possible recommendation is that study of all alternatives be pursued at least to the point that some selection can be made. It is recognized that some combination of choices may be superior to the exclusive use of one.

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