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Laser-Powered Heat Exchanger Rocket for Ground-to-Orbit Launch

Jordin T. Kare*
Lawrence Livermore National Laboratory, Livermore, California 94550

A hydrogen-fueled thruster heated by a ground-based laser can produce a specific impulse of 500–800 s, sufficient to reach Earth orbit with plausible single-stage mass ratios, at exhaust temperatures of 1000–2000 K. At these low temperatures, a solid heat exchanger can be both cheap and efficient. A heat-exchanger-based thruster has a fundamental advantage over other laser-heated engines in that it is omnivorous; any laser wavelength or pulse format is acceptable. A low-cost, lightweight planar heat exchanger configuration is presented that retains much of the simplicity of pulsed solid-propellant laser thrusters, and which may permit quick development of a launch system able to launch payloads of >1 kg/MW of laser power into low Earth orbit.

Nomenclature

c = exhaust velocity
D = horizontal acceleration distance
I_{sp} = specific impulse in seconds, lbf s/lbm
MWe = unit of exhaust power P_e
$\dot{m}$ = rate of propellant consumption
m_f = final (dry) mass of vehicle
m_i = initial mass of vehicle plus propellant
m_p = propellant mass
m_t = propellant tank mass
P_e = useful exhaust power
P_l = laser output power
P_{thr} = laser power incident on the thruster collecting surface
p_c = pressure at heat exchanger output, "chamber" pressure
p_t = propellant tank pressure
R = system range, distance from laser to vehicle when thrust stops
T = temperature
v = vehicle velocity
ε = ratio of tank mass to propellant mass
η_{thr} = thruster efficiency
ρ_p = propellant density
ρ_t = tank wall density
σ_t = tank wall tensile strength

I. Introduction

LASER thermal propulsion is a potential alternative to advanced chemical rockets and cannons (railguns, coil guns, light gas guns, etc.) for low cost ground-to-space transport.[1] In a laser launch system a large ground-based laser transmits energy to heat an inert propellant in a small rocket vehicle. Because the capital cost of the laser is large, economical operation requires launching large numbers (thousands) of vehicles, and thus, places a premium on vehicle simplicity and low cost.

A. Existing Laser Propulsion Approaches

Research on laser propulsion for ground-to-orbit launch was stimulated in the late 1980s by the prospect that large ground-based lasers would be built for strategic defense purposes. Much of this work concentrated on a planar solid-propellant thruster driven by a double laser pulse[2]; initial modeling[3] suggested that a specific impulse I_{sp} of 800 s could be obtained at a thruster efficiency approaching 40%. (Thruster efficiency is defined as $\eta_{thr} = P_e/P_{thr}$, where exhaust power P_e is defined as $\frac{1}{2}\dot{m}c^2$.) The pulsed thruster provides extreme simplicity with a completely inert propellant. The planar geometry provides two key advantages: 1) the ability to accept a laser beam over a range of angles (and therefore to accelerate at an angle to the laser beam, and in particular to launch directly to circular orbit without a kick motor); and 2) the ability to steer the vehicle from the ground by controlling the laser beam (and therefore to avoid carrying a guidance system on each vehicle).

However, pulsed thrusters of this type require very high pulse energies (>100 kJ) and impose stiff constraints on the laser pulse shape and wavelength. They also have loss mechanisms that make even 40% efficiency difficult to achieve; the efficiencies demonstrated to date are ≈10%.[4] Optimizing pulsed thrusters is difficult, because the thrust cycle is complex and some aspects (plasma ignition, high-temperature chemical kinetics) are poorly understood. Few high-average-power pulsed lasers exist, and none provide a double pulse; testing thus requires costly new laser facilities. Finally, the pulsed thruster is incompatible with the rf-linac free-electron laser selected in 1989 for testing as a ground-based antimissile weapon,[5] although the subsequent cancellation of most ground-based laser weapon development has made that incompatibility moot.

The major alternative laser-propulsion technology is the hydrogen-fueled plasma-coupled CW (continuous-wave, i.e., unpulsed) thruster.[6,7] These thrusters have high efficiency and high (9–10 km/s) exhaust velocity. However, they have several disadvantages:

1) Plasma coupling requires onboard focusing optics to provide sufficient laser flux to sustain a plasma. These optics add mass and cost and, unless they are moveable, constrain the vehicle to fly at a fixed angle relative to the laser beam.

2) The beam must be directed into an absorption chamber, either through a very high-performance window or by focusing through the nozzle throat. Either approach is an engineering challenge; a focusing nozzle further constrains the vehicle geometry.

3) The absorption chamber and nozzle must be regeneratively cooled. The very hot (>10,000 K) plasma heats the walls by radiation as well as by conduction through the gas.

4) The laser beam must be continuous and uninterrupted, as the plasma is substantially harder to ignite than to sustain.

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*Physicist, "O" Division, M/S L-278, P.O. Box 808.

These problems initially led this author, among others, to conclude that CW engines are inappropriate for ground-to-orbit launching using near-term (<<1 GW) lasers,[8] although they may be well-suited to satellite maneuvering applications.

The use of hydrogen propellant is also a disadvantage because of the need for a cryogenic tank and plumbing, and the associated safety and handling problems. However, the handling of liquid hydrogen is an established technology, and most advanced space launch concepts assume the use of large quantities of hydrogen.

In an effort to find a thruster compatible with available and proposed lasers, we considered several variations of the planar pulsed thruster concept, including solid- and liquid-fueled systems with various types of focusing nozzles. No such system approached the simplicity and flexibility of the solid propellant thruster.

B. Heat-Exchanger Rocket

The concept of the heat-exchanger rocket is sketched in Fig. 1. Radiation (sunlight or laser light) is absorbed by a solid material, heating it to a high temperature. (Solid-core nuclear-thermal rockets, which share many of the same properties and problems, heat a solid reactor core directly via nuclear fission.) The solid in turn heats a propellant, generally hydrogen, which is exhausted through a conventional nozzle. The specific impulse of such a rocket is limited by the maximum temperature the solid heat exchanger can tolerate. The choice of materials is limited by the need for high thermal conductivity and low reactivity with the propellant, while much of the design is controlled by the relatively low heat capacity and thermal conductivity of gaseous propellants.

Solar-powered heat-exchanger thrusters have been designed for orbital maneuvering applications. In these designs high I_{sp} is of paramount concern, while cost and thruster mass are secondary. Much ingenuity has gone into circumventing the limits of structural materials, by using absorption chambers with cooled windows and, e.g., suspended particles or grids to absorb and transfer energy. These designs would certainly work as laser-driven rockets, but share many disadvantages with the CW plasma thruster. Solid heat exchangers made of tungsten-rhenium alloys have been designed for operation up to 2778 K.[9] These were not, however, designed for low cost, particularly with respect to the fabrication cost of rhenium-alloy tubing, or light weight.

Heat-exchanger thrusters have been proposed for laser propulsion use,[10] but apparently neglected, presumably because they offer lower I_{sp} than plasma-coupled thrusters. However, the key to this article is the realization that for launches to low orbit, high specific impulse is not critical, and may even be undesirable in a system limited by laser power and range. Since I_{sp} is proportional to $T^{1/2}$, a modest reduction in I_{sp} can yield a large reduction in propellant temperature. Even if high-temperature materials such as tungsten are available, operating at lower-than-maximum I_{sp} and temperatures allow larger temperature drops within the heat exchanger, and thus better heat transfer.

The ideal I_{sp} for undissociated hydrogen as a function of T is shown in Fig. 2. It is clear that if I_{sp} values of 500–600 s are acceptable, a thruster, including both the heat exchanger and the nozzle(s), can be made from cheap, easily worked materials such as nickel.

C. Heat Exchanger Thruster Advantages

A moderate-temperature heat-exchanger engine has several appealing properties:

1) It is omnivorous. Any wavelength of laser light can be efficiently absorbed as long as the flux remains below surface ablation thresholds. The thermal mass of the heat exchanger will average over any pulse rate above ≈100 Hz and any pulse format that does not result in surface breakdown. Small-scale beam nonuniformities and short-duration scintillations will not affect performance.

2) It is efficient. Strongly absorbing coatings are simple to fabricate compared to highly reflective ones. Reradiation from the heat exchanger surface varies as T^4; at 1000 C the loss is only 10 W/cm^2, which is small compared to incident fluxes of order 1000 W/cm^2. The loss would be 400 W/cm^2 at 3000 C. "Frozen flow" losses (a major loss mechanism in the pulsed thruster) disappear, since the propellant molecules are never dissociated. If the propellant flow is reasonably uniformly heated, the major losses arise from less-than-ideal nozzle flow.

3) It is very flexible. The heat exchanger can have various shapes and orientations, and can be designed for a wide range of fluxes. A simple absorber is even less dependent on the beam angle-of-incidence than the pulsed planar thruster, and the thrust axis is also independent of the orientation of the absorbing surface. The thruster can also operate at constant I_{sp} with any laser power below its maximum simply by reducing propellant flow, unlike the pulsed thruster that requires a minimum flux. This allows some thrust to be sustained as long as the vehicle is within sight of a laser, even if much of the beam power is lost to diffraction and scattering.

4) It builds on a very large technical base. The thruster involves no new physics. Hydrogen flow, hydrogen tank design, heat exchanger, and nozzle design, etc., are all exten-

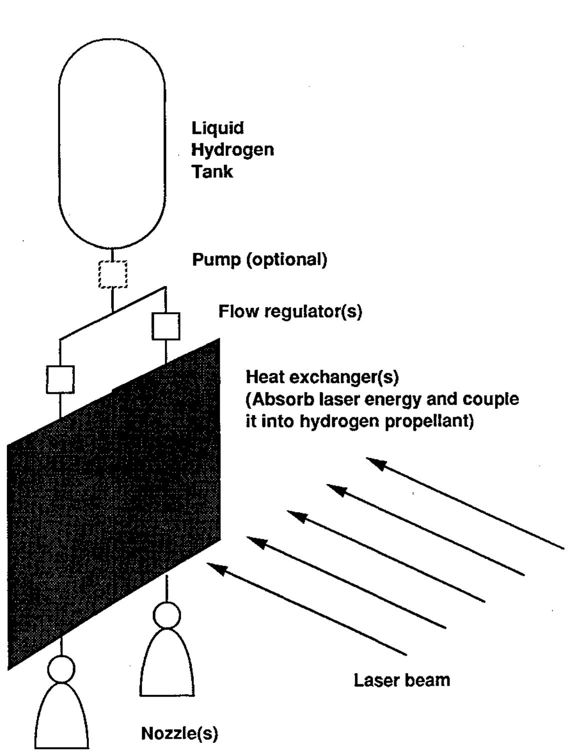

Fig. 1 Heat-exchanger rocket concept.

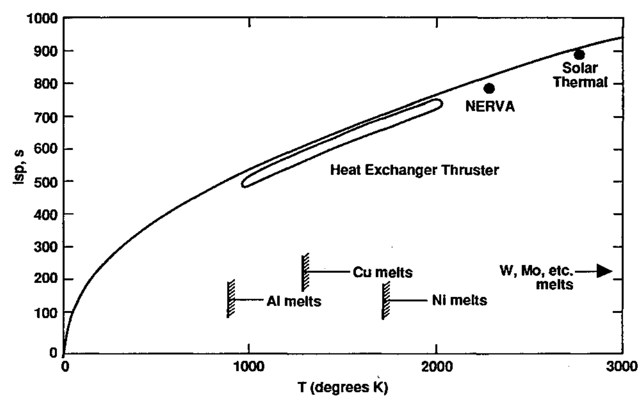

Fig. 2 Ideal I_{sp} vs temperature for undissociated hydrogen.

sively studied disciplines. Continuous flow is much more readily modeled than pulsed flow. Existing work, especially on solar-thermal thrusters and nuclear rockets, may be directly applicable.

5) It is readily testable. CW lasers are available at all power levels up to 1 MW. However, most static testing can be done without a laser. Except for the surface absorption properties (which can be tested on small samples), the thruster behavior is independent of the spectral properties of its heat source, and the required fluxes are low enough ($<10^3$ W/cm^2) to be achieved with incandescent sources, arc lamps,[11] or focused sunlight. Also, many tests can be done with helium or other inert gases instead of hydrogen; in the absence of dissociation, the extrapolations from helium to hydrogen in particular are straightforward.

II. Specific Impulse and Mass-to-Orbit

I_{sp} has been something of a holy grail in propulsion technology, and so designing a thruster with deliberately lower-than-maximum I_{sp} may be surprising. However, laser launch systems operate with constraints substantially different from those of chemical rockets. The vehicle has limited power available, set by the laser, rather than limited thrust. Also, the system has a limited range R, set either by the spreading of the laser beam, or (for launches to low orbit, and in the absence of orbiting relay mirrors) by the curvature of the Earth. Thus, I_{sp} may be profitably traded for increased thrust and higher acceleration.

This can be illustrated with a very simple trajectory: free-space acceleration from rest to a fixed velocity v in a fixed distance D. This is a reasonable model of a laser launch after the vehicle has "turned over" and begun accelerating downrange; D is roughly equal to the system range R. For a fixed exhaust velocity c and exhaust power P_e, the final mass m_f is

$$m_f = 2(P_e D/c^3)(e^{v/c} - 1 - v/c)^{-1} \tag{1}$$

This is the greatest mass that can be accelerated to velocity v in this range. The corresponding initial mass m_i is given by the rocket equation:

$$m_i = m_f e^{v/c} \tag{2}$$

Figure 3 plots m_f vs I_{sp} ($=c/9.8$ m/s^2) for $v = 7.6$ km/s (an idealized eastward launch to low Earth orbit), and $R = 1000$ km. Note that the mass in Fig. 3 is per MWe, i.e., for 1 MW of exhaust power. (Throughout this article we use P_e and MWe to indicate exhaust power, and P_{thr} and MW for laser beam power reaching the thruster; $P_e = \eta_{thr} P_{thr}$, and so a 1-MW laser beam driving an 80% efficient thruster would produce 0.8 MWe of exhaust power.) P_{thr} is equal to the laser power at the ground less transmission losses, such as atmospheric absorption and diffraction, which either reduce the beam power or cause part of the beam to miss the thruster's collecting surface.

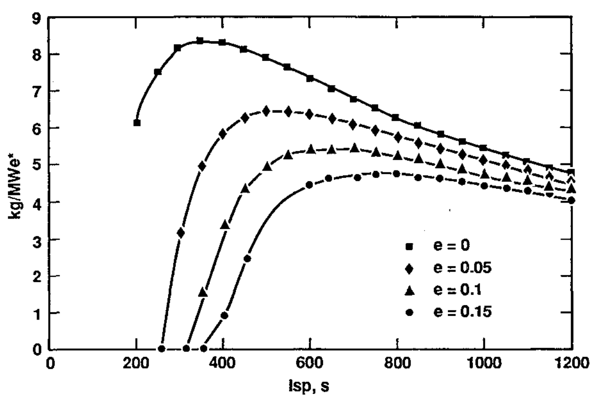

Fig. 3 Mass-to-orbit (maximum mass, excluding propellant tank, which can be accelerated to 7.6 km/s in 1000 km by a rocket with an exhaust kinetic power of 1 MW) as a function of I_{sp} for various rations ε of propellant tank mass to propellant mass.

If m_i is not constrained, m_f is maximum for $c \approx 3500$ m/s. More generally, the optimum for a fixed-I_{sp}, fixed-range system is $c = 0.465v$, provided the vehicle dry mass is independent of the amount of propellant carried. This is true for an ideal pulsed solid-propellant laser thruster, which has no propellant tank or casing.

For the heat-exchanger thruster (and most other rockets), there is a tank mass m_t that depends on the propellant mass: $m_t = \varepsilon m_p = \varepsilon(m_f - m_i)$. Figure 3 also shows the nominal payload mass-to-orbit, $m_f - m_t$, for various tank mass fractions ε. This nominal payload is only an approximate indicator of the actual payload, since it also includes the engine (heat exchanger), structure, avionics, etc., which will scale in different ways with I_{sp}. However, it provides a useful guide to the effect of varying ε and I_{sp}. Obviously $\varepsilon \approx 0.1$ is needed to make a low I_{sp} thruster useful. Given such a tank mass, the variation in payload with I_{sp} above 500 s is relatively slow. The optimum I_{sp} for a fixed-range system increases with the tank mass fraction, but only slightly for moderate tank masses.

The true performance of a laser-propulsion system depends on more than Eq. (1); the vehicle's climb through the atmosphere and then to orbital altitude must be included.

Typical small laser-launched vehicles are short, wide, and light compared to chemical rockets, and will thus be severely affected by aerodynamic drag. The low chamber pressure of pressure-fed thrusters will also limit the performance of laser-driven rockets in the low atmosphere. Thus, we assume vehicles are lifted above most or all of the atmosphere before the laser rocket switches on. This could be done by a separate vehicle such as a cargo airplane or balloon, or by a chemically fueled rocket or ramjet booster. It could also be done by a laser-driven ramjet or hybrid (e.g., chemically powered but laser-steered) thruster.

Once above the atmosphere, a laser-driven rocket must climb quickly to stay above the laser's useful horizon. Typical trajectories can be reasonably well-modeled as a pure vertical acceleration to give the vehicle enough altitude and vertical velocity to coast to orbital altitude, followed by a pure horizontal acceleration to orbital velocity. Unfortunately, even with air drag eliminated, a laser-driven thruster may not be able to lift enough mass from a standing start to achieve the mass-to-orbit given by Eq. (1).

As an example, from Eq. (1), the mass-to-orbit for a pure horizontal acceleration over 1000 km at an I_{sp} of 600 s is 7.3 kg/MWe. The corresponding mass ratio [Eq. (2)] is 3.64, and so the mass at turnover (start of horizontal acceleration) is 26.6 kg/MWe. The coupling coefficient (thrust per unit laser power) at this I_{sp} is 340 N/MWe, and so the acceleration at turnover is 13 m/s^2. If the vehicle starts from rest, it can carry at most an additional 8 kg/MWe of propellant, or it will fall instead of rise. The vehicle must reach $\approx$400-km altitude to stay sufficiently far above the horizon at 1000-km range. ("Sufficiently far" depends on the laser's ability to penetrate the atmosphere far from the zenith, but is at least 20–30 deg above the actual horizon.) Consuming 8 kg/MWe of propellant will produce less than 200 m/s of vertical velocity; enough to lift the vehicle less than 2 km.

One solution is to add a booster stage that provides a modest vertical velocity, such as a solid-fuel chemical rocket, laser ramjet, or conventional ramjet. This provides a compound advantage, since the vehicle can carry additional propellant, i.e., the initial thrust-to-weight ratio using only the main laser-driven thruster can be less than 1. An initial vertical velocity of approximately 1 km/s allows launching the full 1000-km range-limited mass-to-orbit with a thruster I_{sp} of 600 s. However, the booster must be either very cheap or reusable, and it will make the launch system more complex and possibly less safe. (This velocity could be provided efficiently by a "cannon" launcher—a compressed gas gun or electromagnetic accelerator—in a large laser launch system, but not in a sub-100-MW system because the vehicles are too light. Sea-level air drag would decelerate a typical few-hundred-kilogram vehicle traveling 2 km/s at 100 *g*).

Alternatively, one can accept a lower final mass and proportionately shorter range; this also lowers the altitude required to stay above the laser's horizon. This exacts a larger penalty in useful payload for the heat-exchanger thruster than for, e.g., pulsed solid propellant thrusters, since the heat exchanger mass is fixed.

Kantrowitz[12] has pointed out that, with fixed power, temporarily lowering the thruster I_{sp} allows one to lift a larger initial mass at the expense of higher propellant consumption. The I_{sp} of the heat exchanger thruster (HX) can be varied in flight simply by changing the propellant flow rate. A more radical change involves switching to a higher molecular weight propellant, such as water, methane, or ammonia. These are much denser liquids than hydrogen, and so a sizable extra propellant mass could be carried at a small penalty in tank mass. This dual-propellant option should be investigated, but optimizing the heat exchanger for two different working fluids might be difficult. (Denser propellants are tempting for the entire flight. However, their threefold penalty in I_{sp} relative to hydrogen is too large; even with zero tank mass they yield substantially less mass-to-Earth orbit than can be achieved with hydrogen.)

Finally, one can (reluctantly) give up some of the benefits of low temperature, and use higher thruster I_{sp}. For $R = 1000$ km, an I_{sp} of ≈ 800 s places the most mass-to-orbit from a standing start. Lower I_{sp} systems are limited by the takeoff mass; higher I_{sp} systems by the system range. If longer ranges are possible, the optimum I_{sp} would be greater. Obviously, higher I_{sp} would also be desirable for higher-velocity launches to high orbit or Earth escape.

III. Thruster and Vehicle Design

A laser launch system inherently launches large numbers of relatively small vehicles. In a small system, the vehicles must be largely or completely expendable to avoid recovery costs, and thus must be as cheap and simple as possible. To explore what is possible, we consider a baseline vehicle concept with the following features:

1) Low-pressure, pressure-fed operation. Low pressure minimizes the propellant tank mass. A pump would allow use of a very lightweight tank, but would add substantial cost and complexity. This trade should be considered in more detail, but for initial analysis a pressure-fed system is preferred.

2) Planar heat exchanger with no windows or optics. A flat heat exchanger (as opposed to, e.g., a conical or cylindrical surface) can be uniformly heated by a beam from any angle, and can be easily subdivided into sections feeding separate nozzles to allow thrust vectoring by changing the laser beam profile.

3) Nominal I_{sp} of 600 s, corresponding to an exhaust temperature of $\approx 1000°C$. This allows the highest-temperature portions of the thruster to be fabricated from nickel at low material cost. Higher-temperature materials, notably tungsten, would permit I_{sp} of 800 s, but are comparatively expensive and difficult to fabricate.

4) Nominal heat exchanger diameter of 2 m and power level of 20 MWe. This corresponds to the minimum-size vehicle considered for a pulsed solid-propellant launch system,[13] and is an acceptable collector diameter for a nominal optical system (10-m-diam beam projector and 10.6-μm laser wavelength) and a system range of 1000 km.

A. Heat Exchanger Design and Fabrication

For any I_{sp}, the mass-to-orbit is at most a few kg/MWe for a 1000-km range. Thus, the heat exchanger must handle ≈ 1 MW/kg of heat exchanger. If the system is pressure-fed, the pressure drop in the heat exchanger must be as low as possible, at most a few tens of psi. The nominal power level and area of the heat exchanger yield a mean flux on the heat exchanger of 700 W/cm^2 (assuming $\eta_{thr} = 90\%$). It was initially unclear whether this performance could be achieved, particularly with a design that could be fabricated cheaply. However, the DUMBO nuclear rocket heat exchanger design[14] transferred 25 MW/kg of heat exchanger (at a relatively high hydrogen pressure). With this as encouragement, a heat exchanger was designed that appears to meet all the requirements for a launch vehicle.

The heat exchanger is based on the laminar-flow microchannel concept. This concept was developed (and extensively tested) for liquid cooling of semiconductors by Tuckerman.[15] The DUMBO nuclear rocket heat exchanger is similar in principle. Laminar flow provides high heat transfer rates at lower pressure drop than comparable turbulent-flow heat exchangers. The heat exchanger consists of very small high-aspect-ratio channels, as illustrated in Fig. 4. The channel width is constrained by the thermal conductivity of the propellant gas, and is nominally 200 μm for hydrogen. The channel height is constrained by the thermal conductivity of the fin material and the thermal loading (W/cm^2) of the heat exchanger face. For a nickel heat exchanger at 1000 W/cm^2, the temperature drop along the fins approaches 1000 K/cm, and the maximum usable channel height is a few millimeters. This also meets the constraint implied by the combination of specific power and flux, which is that the areal density of the heat exchanger not exceed, on average, approximately 1.4 g/cm^2.

The microchannels are formed in panels, with a channel length of typically 2 cm, and are joined by relatively large cross section headers, as shown in Fig. 5. This header-and-panel structure minimizes the flow length, and thus, the pressure drop, in the microchannels. It also maximizes the flow cross section of the heat exchanger, and thus reduces the gas flow velocity. (A single set of microchannels would have an unacceptably small cross section; e.g., the entire rim of a 2-m-diam $\times$ 2 mm-thick disk is only 125.6 cm^2. With 50% fin area, the flow cross section would be only 63 cm^2, requiring near-sonic or even supersonic flow velocities at our nominal pressure and mass flow rate.)

A full-scale heat exchanger would consist of several stages joined by manifolds, which could be arranged in any desired geometry; Fig. 6 shows a linear arrangement. This multistage design allows different sections to be optimized for different temperatures. Nickel is used for the high-temperature parts

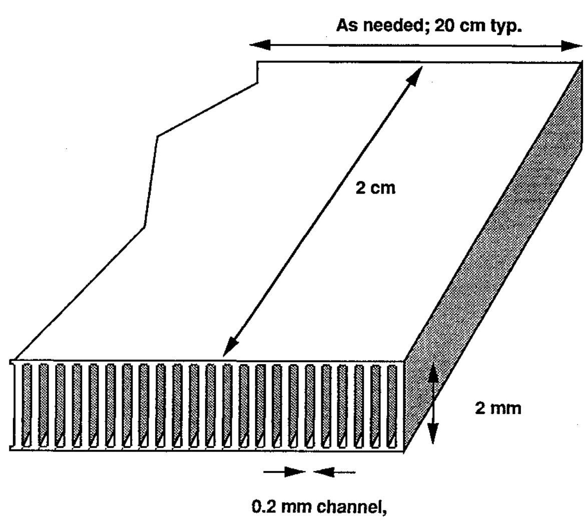

Fig. 4 Laminar-flow microchannel heat exchanger, channel structure and dimensions.

Table 1 Properties of nominal microchannel heat exchanger

Material	Copper (low temperature); nickel (high temperature)
Fabrication	Electroplating
Channel height (heat exchanger thickness)	0.2 cm
Channel width (nickel section)	200 μm
Fin width (nickel section)	200 μm
Channel length	2 cm
Stages	4 (liquid—100 K; 100–300 K, 300–800 K, 800–1273 K)
Gas exit temperature	1273 K (1000°C)
Max. surface temperature	Approximately 1450 K (1177°C)
Design flux	7 MW/m^2 average
Areal density	6 kg/m^2 average
Specific power	1.1 MW/kg (1.0 MWe/kg for η_{thr} = 90%)
Entrance pressure	500 kPa (5 atm)
Exit pressure	200 kPa (2 atm)

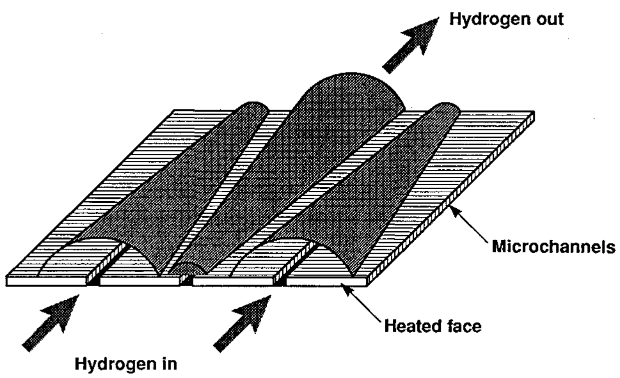

Fig. 5 Assembly of microchannel heat exchanger panels and headers.

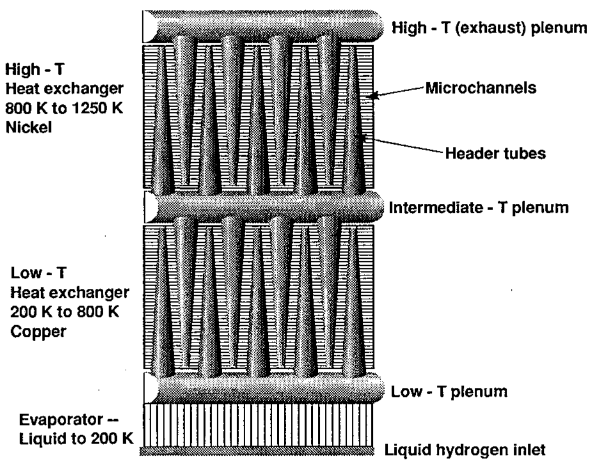

Fig. 6 Complete multistage heat exchanger, linear (as opposed to radial) geometry.

of the heat exchanger because it is resistant to attack by hydrogen and melts at over 1700 K. Lower temperature parts could be made of copper, whose improved thermal conductivity allows thinner fins, and thus, lower mass.

The manifolds also allow turbulent mixing of the gas flow to even out gas temperature variations, and can be designed to accommodate the thermal expansion of the individual sections. Finally, there is a flow instability in laminar-flow systems, identified by the designers of DUMBO, which limits the maximum temperature rise in a single set of uniform channels to $T_{\text{out}}/T_{\text{in}} < 4$. Thus, heating hydrogen from near 27 K to over 1250 K requires at least three stages of heat exchanger. (DUMBO avoided this problem by using nonuniform channels, which is possible here, but would add to the manufacturing complexity of the heat exchanger.)

The heat exchanger structure can be fabricated by an electroplating process developed by Steffini[16]; individual panels are formed by stacking alternating strips of nickel ribbon (fins) and conductive spacer material. The panels are then electroplated to form front and back surfaces, and the spacers etched out or otherwise removed. Integral headers can be made by plating over and then removing appropriate forms, allowing mass production of heat exchangers with a minimum of machining and welding. Sample heat exchanger sections have been fabricated but not tested under load. It appears possible to construct a heat exchanger out of, e.g., tungsten and rhenium, using a similar process; tungsten cannot be plated but can be deposited by other techniques, whereas rhenium is platable but it is very expensive.

Table 1 gives the nominal properties of a baseline heat exchanger.

The heat capacity of the high temperature part of the exchanger is about 0.3 J/cm^2 K. At 600 W/cm^2, the heat exchanger can heat or cool at $\approx$2000 K/s as the laser beam is shifted to steer the vehicle, and significant changes in temperature (and therefore thrust) can occur in $\approx$0.1 s.

The working life of the heat exchanger is only a few minutes, and so it can operate close to its material limits. It can also operate with defects that would be unacceptable in most applications; e.g., pinhole leaks. An elegant feature of the planar heat exchanger is that the local thermal load can be controlled by adding a reflective coating (or removing an absorbing one; bare copper is a good reflector). Thus, the poorly cooled area under headers, nozzles, joints between heat exchanger sections, etc., can be protected from overheating, as can defective regions such as blocked channels. The laser energy striking these areas is lost, but should be a small fraction ($\ll$10%) of the whole beam.

B. Propellant Tank

The second major vehicle component is the propellant tank. The tank size depends on the details of the launch system (e.g., initial vertical velocity, system range), but can be estimated roughly by assuming an initial vehicle thrust-to-weight ratio of 1. For the baseline case (600-s I_{sp}), the initial vehicle mass is then 34.6 kg/MWe, of which somewhat less than 28 kg/MWe is propellant. For a 20-MWe system, the total hydrogen mass is 560 kg, and at a liquid density of 70 kg/m^3, the tank volume is 8 m^3, and its diameter roughly 2.5 m. At this size scale, the tank wall is almost purely pressure-loaded; hydrogen is so light that the loading due to its weight is negligible even at 10 g acceleration. The tank mass for a spherical tank is

$$m_t = \tfrac{3}{2} m_p (\rho_t p_t / \rho_p \sigma_t) \tag{3}$$

For an aluminum tank ($\sigma_t \approx$ 50,000 psi) holding hydrogen, the tank-to-propellant mass ratio ε is $\approx$0.0012 (p_t/1 psi). To keep ε under 0.1, the tank pressure must be limited. A tank pressure of 500 kPa (70 psi) is compatible with the baseline heat exchanger design, neglecting small pressure drops between the tank and the heat exchanger entrance. For a 70-psi tank, $\varepsilon = 0.085$. The low pressure, comparatively small size, and short operating life of the tank allow designs with modest safety margins. A mass allowance for insulation is not included, since the tank needs a hold time of only a few minutes if it is topped off immediately before launch.

High-performance fiber composite tanks have been constructed with strength-to-density ratios of 307,000 N-m/kg,[17] roughly 2.5 times better than aluminum, which would yield $\varepsilon = 0.035$ for 70-psi hydrogen. These were small tanks for much higher pressures, however, and presumably quite expensive. High-performance plastic tanks are also possible; Rusek and Shelley[18] have suggested that liquid-crystal-polymer (LCP) plastics, relatives of familiar high-strength plastics such as Kevlar®, might allow mass production of molded tanks with substantially higher strength-to-weight than aluminum tanks. This would allow either lighter tanks or higher pressure; higher pressure would relax constraints on the heat exchanger and nozzle.

Ideally, one could choose the thermal properties of the tank and supports to keep the hydrogen boil-off rate just above that needed to maintain the tank pressure. A simple vent would then maintain constant pressure with reasonable propellant loss. However, a pressure control system, such as a thermal link that closes when the pressure drops, will probably be necessary as well. Depending on the heat exchanger design, it may be possible to feed vented hydrogen into the heat exchanger at an intermediate-temperature point, and thus get some thrust from it. If not, it can be used to cool the vehicle side walls, which are exposed to the fringes of the laser beam for part of the flight. (The side walls can be highly reflective, so little or no cooling will be needed. Unlike the walls of a pulsed-thruster vehicle, they are exposed only to neutral hydrogen, and are not subject to deposition of propellant residue.)

C. Nozzles

The nozzles of the heat-exchanger thruster are uncooled, and can be formed either from the same materials as the heat exchanger or from lighter-weight high-temperature materials such as ceramic composites. The nozzles should be closely coupled to the heat exchanger to minimize pressure drops in connecting ducts, but do not need to be directly attached to the heat exchanger.

The nozzle throat area can be calculated from the laser power and thruster exhaust temperature and pressure. A rough calculation assumes the energy content of the exhaust is 20 kJ/g (6 km/s exhaust velocity and 90% efficient nozzle), so that the propellant flow rate is 50 g/s per MWe. The density of hydrogen at a nominal "chamber" pressure of 2 atm and 1200 K is roughly 50 g/m^3, so that the flow volume is $\approx$1 m^3/s per MWe (the liquid flow rate from the propellant tank is 0.7 l/s per MWe). Taking a sound speed of 2400 m/s and a critical pressure ratio (throat pressure/chamber pressure) of 0.53 gives a throat area of just under 8 cm^2/MWe. Thus, the baseline 20-MWe thruster has a total throat area of $\approx$160 cm^2. For moderate expansion ratios, the nozzle exit will be small compared to the heat exchanger (e.g., for 50:1 expansion, the total exit area would be 0.8 m^2, or about one-quarter of the heat exchanger area). The stresses on the nozzle are modest, and so the nozzle mass should be negligible compared to the heat exchanger mass.

Assuming the vehicle is to be steered by moving the laser beam on a heat exchanger divided into sections, there must be three or more nozzles. These can be in any configuration, including sections of a single nozzle, or one main nozzle and three or four small attitude control nozzles.

D. Vehicle Structure

There are several ways to arrange the tank, heat exchanger, and nozzles making up the heat-exchanger rocket. These can be divided into "side-fire" and "tail-fire" geometries. Figure 7 shows some possible vehicle geometries schematically. In side-fire geometries (Figs. 7a and 7b), the flat heat exchanger faces to the side of the vehicle. The vehicle is not axisymmetric, and must be oriented via, e.g., roll control jets so that the absorbing surface faces the laser. The plane of the heat exchanger may be parallel to the vehicle axis for structural strength or minimum drag, or tilted to present the greatest possible area to the laser beam. The configuration of Fig. 7b is compact, and does not need a separate stray-light shield for the tank, but is asymmetric, and may therefore be harder to control.

The tail-fire configuration (Fig. 7c) has the heat exchanger surface normal to the vehicle axis, facing aft. It is similar in overall shape to vehicles using a pulsed planar thruster and would fly similar trajectories. This geometry is compact and structurally efficient, and does not need to maintain a fixed roll orientation. Figure 8 shows an artist's conception of a complete tail-fire vehicle, to provide an idea of the actual scale and appearance of the components.

The nozzle in the tail-fire geometry shown is an aerospike nozzle, a plug nozzle with an annular throat and a short, truncated plug.[19] The exhaust gas trapped against the heat exchanger surface is not ionized, and will be essentially perfectly transparent to the laser. This configuration is particularly elegant, since the heat exchanger doubles as the nozzle expansion surface, and both low- and high-temperature plumbing are minimized. However, conventional nozzles around the heat exchanger rim would not be significantly heavier.

The side-fire geometry requires separate conventional nozzles and a roll-control system. However, as shown in Fig. 9,

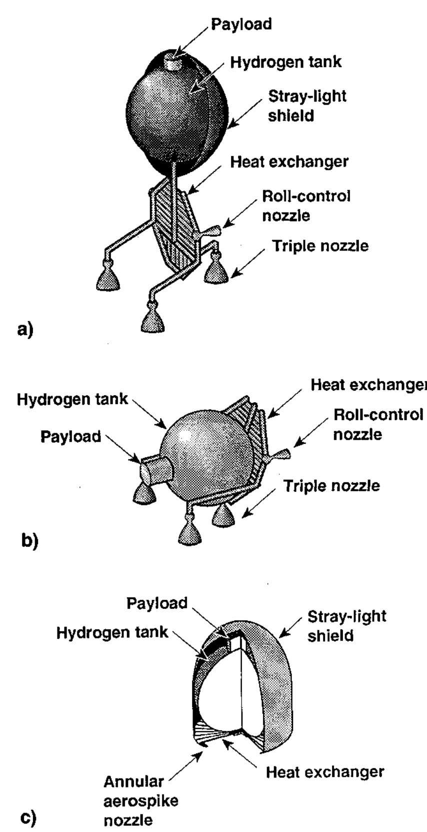

Fig. 7 Possible geometries for a heat-exchanger-based vehicle. a) symmetric side-fire vehicle, b) compact asymmetric side-fire vehicle, and c) tail-fire vehicle with annular aerospike nozzle.

Table 2 Mass budget and payload for five launch vehicles

Case	1	2	3	4	5[a]
I_{sp}, s	600	600[b]	600	800	800[b]
Initial velocity, m/s	0	0	1000	0	0
Range, km[c]	678	725	1111	907	987
Altitude, km[c,d]	259	290	460	363	394
Initial mass, kg/MWe	27.5	30	37.5	22.5	25
Final mass, kg/MWe	4.35	4.66	7.31	5.06	5.56
Propellant, kg/MWe	23.2	25.3	30.2	17.4	19.4
Tank at 0.085 m_p	1.97	2.15	2.57	1.48	n/a
Heat exchanger, kg/MWe	1.0	1.0	1.0	1.0	n/a
Nozzle, kg/MWe	0.1	0.1	0.1	0.1	n/a
Structure, kg/MWe	0.5	0.5	0.5	0.5	0.5
Total empty mass, kg/MWe	3.57	3.75	4.17	3.08	0.5
Useful payload, kg/MWe	0.78	0.91	3.14	1.98	5.06
Nominal efficiency	0.8	0.8	0.8	0.8	0.2
Payload/P_{thr}, kg/MW	0.62	0.73	2.51	1.58	1.01

[a]Pulsed solid-propellant thruster.
[b]Initial specific impulse reduced to give minimum 15 m/s (1.5 g) acceleration.
[c]At orbital insertion.
[d]Determined by the maximum laser zenith angle of 70 deg.

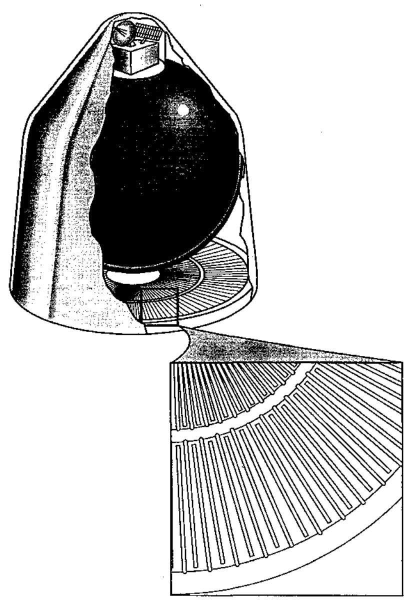

Fig. 8 Artist's conception of a laser-driven launch vehicle using a circular radial-geometry heat exchanger and annual aerospike nozzle.

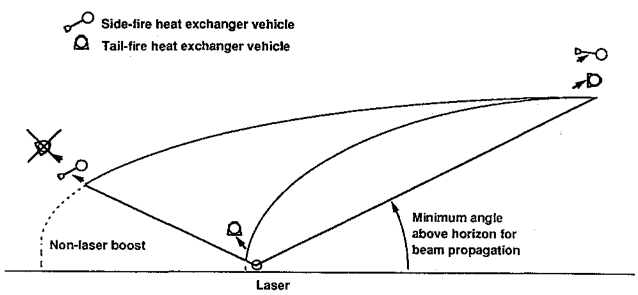

Fig. 9 Trajectories of tail-fire and side-fire vehicles; the side-fire geometry allows a longer overall trajectory if the vehicle is launched far uprange of the laser.

a side-fire vehicle can fly longer trajectories with a given laser, provided it starts some distance uprange of the laser. This may substantially increase the launch system payload size.

IV. Vehicle Performance and Mass Budget

A modified version of the LAUNCH computer code[20] was used to calculate the expected performance of several configurations of the HX thruster, as well as one example of a solid propellant thruster for comparison. Results are given in Table 2. Except as noted, all masses are in kg/MW of exhaust power, and are thus independent of laser size and thruster efficiency. The exhaust power was taken to be constant through the launch (i.e., diffraction and atmospheric absorption losses were turned off). The nominal limits on the launch trajectory were a laser zenith angle of 70 deg and an orbit perigee of 200 km; Earth rotation is not included in the code. The vehicle mass budget assumes a tank mass fraction ε of 0.085, and a heat exchanger mass of 1 kg/MWe. The corresponding numbers for the solid-propellant vehicle are, of course, both zero. The structural mass is fixed at 0.5 kg/MWe for both types of thrusters, although since we have done no detailed vehicle design, this is simply an educated guess.

These cases were not extensively optimized, but a range of initial masses were tried in each case, with those shown giving the highest payload. Case 2 is included to indicate the possible gain from launching at reduced specific impulse to increase the initial acceleration; the gain is small but not negligible. Case 5 corresponds to the roughly 1 kg/MW of laser power previously estimated (e.g., by Kare[13]) for the pulsed solid-propellant thruster. The efficiency taken for the pulsed thruster (20%) is conservative, and pulsed-thruster payloads of up to 2 kg/MW of laser power may be feasible.

The mass estimates in Table 2 are necessarily rough. The useful payload might be reduced by the mass of, e.g., residual propellant and telemetry that must be carried to orbit. Conversely, the payload could be substantially increased by a lighter propellant tank. A detailed design model will be needed to accurately calculate payload mass-to-orbit. However, there is clearly considerable margin to work with, as the payload mass is half the vehicle empty (dry) mass, and 4–8% of the takeoff mass.

A. Vehicle Scaling

As an example of a possible launch vehicle, we consider case 3 in Table 2. At 20 MWe of exhaust power, it would have a payload of 63 kg. This is a reasonable value for, e.g., a 50-kg small satellite plus mounting and deployment hardware. With 80% efficiency, the laser power reaching the vehicle would need to be ≈25 MW. An actual laser power of 50 MW would provide a substantial margin for optics and transmission losses.

This launch vehicle uses 604 kg of liquid hydrogen. This occupies 8.6 m^3, or a spherical tank with a diameter of 2.5 m. The nominal heat exchanger would be 2 m in diam, and so a vehicle diameter of 2.5 m is quite reasonable. The tank wall, if made of aluminum, would be roughly 1 mm thick, with an areal density of 3.0 kg/m^2.

The masses of both the heat exchanger and the hydrogen tank scale linearly with the laser power, and thus the overall configuration should change only slowly with system size. The hydrogen tank size will vary only as $P_e^{1/3}$. The heat exchanger diameter varies as $P_e^{1/2}$ for constant areal power density. Neither variation is fast enough to make the system difficult to scale, although a very large vehicle might be somewhat unwieldy, e.g., a 200-MW vehicle would be over 6 m in diameter. Increasing the heat exchanger flux limit would reduce the diameter, and would become more desirable as the vehicle becomes larger.

Increasing the flux limit might or might not improve the specific power limit of the heat exchanger. Intuitively, a more compact heat exchanger is likely to be lighter, but the differences may be small. Conversely, however, a lower-flux heat exchanger might be only slightly heavier. Thus, a test vehicle might be designed with, e.g., a 2-m-diam heat exchanger handling only 5 or 10 MW. This would allow launching a token payload to orbit with a comparatively modest laser and beam director, substantially smaller than the minimum system needed with a pulsed thruster. The test-vehicle scaling could be further improved by using a side-fire geometry to get the maximum possible laser range. As little as a 5-MW laser and a 4-m beam director (at 10.6 μm) could put a few-kg payload in orbit.

An even smaller vehicle, launched as a sounding rocket, would demonstrate all the basic properties of the system. A $\frac{1}{2}$-m-diam heat exchanger would be well-matched to a 1-MW class laser, and could be powered to 100–200-km ranges with reasonable optics. It would have a very small, light hydrogen tank (since it would only be powered for a minute or so) and could accelerate at several *g*.

B. Vehicle Cost and Economics

It is difficult to estimate the cost of building a heat-exchanger vehicle, particularly in production quantities. To achieve "interesting" costs of <<$1000/kg of payload launched, the production cost per vehicle must be at most a few thousand dollars. However, by the nature of laser launch, an economical launch system will use at least several thousand vehicles per year, a rate more familiar in the automotive industry than in aerospace. Given ground-based guidance and control, each vehicle will carry very little avionics and have at most a handful of moving parts.

The heat exchangers themselves, while complex structures, can be mass-produced starting with sheet or ribbon stock of fin and spacer materials and cast or molded forms for headers. Electroplating on the size scale required is a routine industrial process. Joining of heat exchanger panels will probably require welding, but over limited areas and with modest quality and accuracy requirements. This ease of fabrication strongly drives the choice of materials for the heat exchanger; higher performance materials can be used only if comparably cheap fabrication techniques are available.

Overall, the complexity of the heat-exchanger vehicle is vastly less than that of a modern automobile. A more appropriate comparison, in both mass and complexity, is a large household appliance such as a refrigerator; indeed, modern refrigerators, with microprocessor controls and high-efficiency compressors, may be more complex. Assuming comparable production runs, we would predict a similar unit cost of about $500. Granting the much greater experience base associated with refrigerators, it is still reasonable to predict per-vehicle costs of $1000–$10,000. At $5000, the vehicle cost contributes $100/kg to the launch cost of our nominal 50-kg satellite.

No lasers of the scale needed for a launch system have been built, but the cost of a 20-MW laser and associated facilities has been estimated[13] at $450 million. Electricity costs will depend on the price of electricity, laser efficiency, and vehicle payload and trajectory, but are also of order $100/kg or less (e.g., running a 10% efficient laser for 360 s takes 1 kWh/kW, or 50,000 kWh for a 50 MW laser; at 4 cents/kWh, each 50-kg payload launch would cost roughly $2000 in electricity). Thus, at all but the highest launch rates (>10,000 per year) the maintenance and amortization of the laser system will dominate the cost per launch.

V. Conclusions

A relatively low-temperature heat-exchange thruster, using liquid hydrogen as a propellant, is a promising choice for laser propulsion to low Earth orbit. A low-pressure thruster and low-pressure-drop laminar-flow heat exchanger allow the propellant to be pressure-fed, thus keeping the vehicle simple. The heat exchanger can probably be fabricated cheaply by an electroplating process, and the propellant tank can be simple aluminum, although a higher-performance tank is desirable.

The overall performance (payload to orbit per MW of laser) is comparable to that predicted for a 20–40% efficient, 800-s I_{sp} pulsed thruster. The higher expected efficiency compensates for the weight of the heat exchanger and tank. With a modest (<1 km/s) vertical boost, the payload can exceed 2 kg/MW of laser power; even with no initial boost, the payload should approach 1 kg/MW.

The overall system performance could be improved by using a higher-temperature heat exchanger to get a higher I_{sp}, but the gain would be moderate for launches to low Earth orbit. The gain would be larger in launching to higher orbit or to Earth escape. There is no fundamental reason that the heat-exchanger thruster could not be fabricated from, e.g., tungsten or suitable-coated graphite, and operated at 800-s I_{sp}, but low-cost fabrication processes will be needed.

Laser propulsion, like many other innovative launch technologies, has been handicapped by the high cost of demonstrating performance on a realistic scale. The planar heat-exchanger thruster may well be the easiest of all advanced propulsion technologies to develop and test. If the mass and performance estimates of this article are borne out by detailed design and testing, the heat-exchanger thruster may also be one of the lowest cost and highest performance technologies for launching payloads to low Earth orbit and beyond.

Acknowledgments

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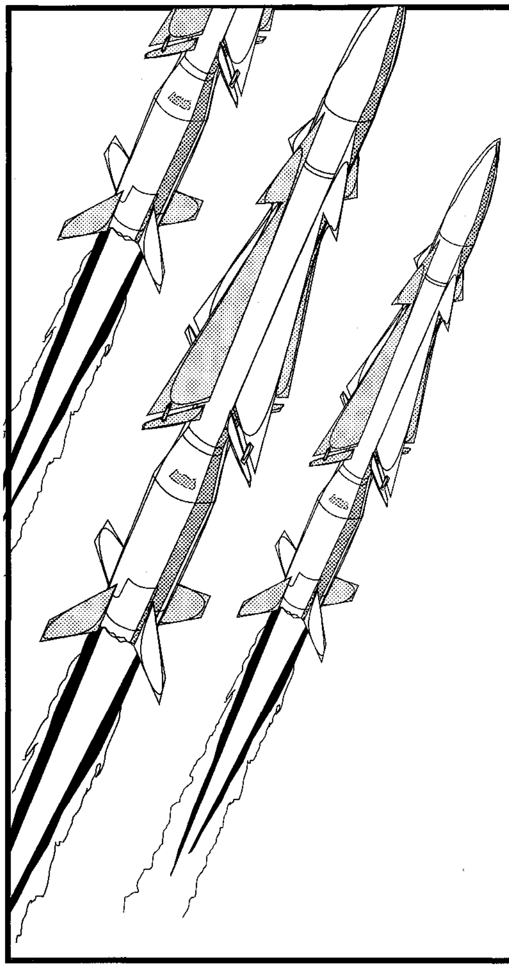

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