

## LOW-COST LAUNCH SYSTEM AND ORBITAL FUEL DEPOT†

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**Abstract**—A new system is proposed for the low-cost launching of large quantities of rocket fuel into Earth orbit and the storage and transfer of the fuel to the NASA Space Station. The system consists of an electromagnetic launcher that fires heat-shielded fuel tanks into high Earth orbit, where they are captured by a long, rotating tether. The tether is in an elliptical orbit that ranges from high Earth orbit down to low Earth orbit, where it drops the fuel tanks into an orbit near the Space Station. The payloads are used for upper stage rockets, for stationkeeping propulsion, and for high-*g*-tolerant supplies for the Space Station. The system has the potential for launching more than 100,000 kg into Earth orbit each year at a launch cost of <\$10/kg.

### 1. INTRODUCTION

Current launch vehicle systems such as the Space Shuttle involve launch costs on the order of \$1000/kg of payload. These costs are so high that commercial activities and routine space operations by private companies are not yet feasible. Before the large-scale development of space resources and energy can succeed, launch costs must be reduced by about two orders of magnitude. Launch costs for liquid-fuel rockets cannot be lower than the cost of the fuel, which is about \$3/kg for liquid hydrogen[1]. This limits the cost of launching to a minimum of \$50/kg of payload, not considering vehicle development and launch operation costs. This cost may be too high for cheap space industrialization. The largest component of space payloads is fuel for upper stages[2].

Various schemes have been proposed for low-cost launch. These include a ground-based laser to provide external power[3], a space elevator with electrical power distributed along its length[4], a rotating tether to pick up large payloads from the ground or upper atmosphere[5, 6] and a launch loop or orbital ring with rapidly moving internal components to support it against gravity[7, 8]. Each of these schemes is attractive theoretically, but either requires great improvements in materials or lasers, enormous masses in orbit, or large, dynamically stabilized masses overhead. A compromise system is needed that puts most of the energy requirements on the ground and that requires minimum mass in orbit[9]. The direct launching of orbital payloads by electromagnetic guns has been proposed[10] but current concepts require an upper stage propulsion

system, leading to large projectiles with small payloads. Tethers also have been proposed for payload launching from the Space Station[11].

This paper proposes a new technique for the low-cost launching and storage of fuel into Earth orbit. The concept presented here is the combination of a relatively small electromagnetic gun on the ground with a relatively low-mass rotating tether in orbit to catch the projectile. The use of the rotating tether eliminates the need for an upper stage on the projectile and provides a method for delivering the payload into different orbits. A preliminary concept definition study is performed, including tradeoffs between gun launch angle and satellite altitude vs required launch velocity, and satellite altitude vs required velocity change for transfer from high Earth orbit to low Earth orbit.

The baseline system is shown conceptually in Fig. 1. The launcher is a railgun that fires its payloads eastward at a fixed muzzle angle, using local electrical power and energy storage. The payloads are slender projectiles that have protective nose cones and are fired several times a day to reach the orbit of the rotating tether, or slingsat. The slingsat consists of a central power and communications module and a long, double-ended, rotating tether with a tip velocity that matches the velocity difference,  $\Delta V$ , between the projectile and the satellite. The tether end attachment captures the payload and holds it securely until the satellite approaches the orbit of the Space Station. The end attachment then drops the payload into orbit near the station, where it can be retrieved as needed by a short tether. Energy balance and drag makeup for the satellite are provided by an electrodynamic tether that interacts with the Earth's magnetic field to change the satellite orbital velocity and rotation rate.

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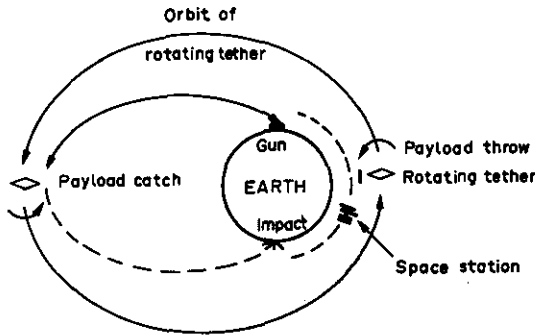


Fig. 1. Ground launcher and rotating tether catcher.

2. SYSTEM ANALYSIS

From Bate *et al.*[12] the general expression for the orbit of a projectile launched with a velocity  $V_0$  and angle  $\beta$  above the horizontal is given by eqns (1) and (2):

$$1/a = 2/r_e - V_0^2/\mu, \tag{1}$$

$$e^2 = (r_e V_0^2/\mu - 1)^2 \cos^2 \beta + \sin^2 \beta, \tag{2}$$

where  $a$  and  $e$  are the orbit semi-major axis and eccentricity,  $\mu$  is the Earth's gravitational parameter, and  $r_e$  is the radius of the Earth (location of the launcher). The apogee of the resulting orbit,  $r_{max}$ , is equal to  $a(1 + e)$ .

We seek a system with the minimum requirements on the launching gun and on the catching satellite. The minimum drag results from a vertical launch, and this also results in the minimum launch velocity to reach a given altitude, as derived from eqns (1) and (2) and shown by the lowest curve on Fig. 2. However, this minimum launch velocity results in the payload just reaching the target altitude, and full orbital velocity would have to be imparted to the payload by the rotating tether. The payload can be given nearly orbital velocity by a horizontal launch, but this requires much higher launch velocities, as shown by the upper curve on Fig. 2. The minimum velocity required for putting a payload into orbit from the Earth's surface is about 8 km/s for a horizontal launch. The velocity requirement rises only slightly with orbital altitude to slightly over 10 km/s

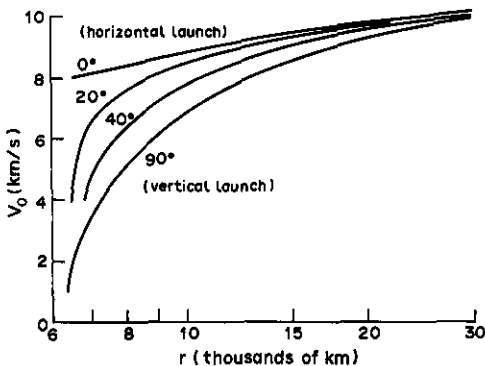


Fig. 2. Launch velocity required to reach various distances.

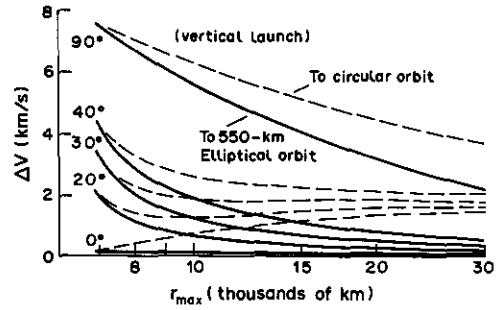


Fig. 3.  $\Delta V$  for slingsat rendezvous in circular and elliptical orbits.

into geostationary orbit. Unless the gun is on a high mountain, however, the atmospheric drag penalty is too severe for a horizontal launch. The horizontal and the vertical launching represent the extremes of having the ground launcher provide most of the velocity change or having the orbital catcher provide the most. Other launch angles provide intermediate values. The velocity of the satellite in circular orbit at  $r_{max}$  is given by  $v_c^2 = \mu/r_{max}$  and the velocity of the projectile at  $r_{max}$  is given by  $v_i^2 = (2/r_{max} - 1/a)$ . From these values, we can obtain the velocity difference, or  $\Delta V_c$ , between the velocity of the projectile at apogee and the corresponding circular orbit velocity:

$$\Delta V_c = v_c [1 - \sqrt{2/(1 + R_p)}] \tag{3}$$

where  $R_p$  is defined as the ratio of the apogee to the perigee for the projectile orbit. The dashed lines in Fig. 3 show the required  $\Delta V$  at the satellite to catch the incoming payload as a function of the satellite circular orbit radius and for different launch angles. Horizontal launch requires the lowest  $\Delta V$ , whereas vertical launch requires the highest. The curves for launch angles of 20, 30 and 40° show some interesting results. For the lower angles, there is a satellite altitude that requires a minimum  $\Delta V$ . For a 20° launch angle, it is about 9000 km radius and requires a  $\Delta V$  of about 1.25 km/s at the satellite. This is within the capabilities of existing high-strength materials such as Kevlar. A fuel depot satellite in this orbit with a Kevlar tether could be used to catch fuel tanks and store them, perhaps in a Space Shuttle External Tank. It could refuel orbital transfer vehicles on their way between LEO and GEO, halving the  $\Delta V$  required by these vehicles.

It is possible to lower the required  $\Delta V$  even more. When the payload reaches the satellite, it has less than circular velocity and is moving slower than the satellite. If the satellite were in an elliptical orbit with the apogee at the rendezvous point, it would have less than circular velocity, requiring less  $\Delta V$ . The minimum satellite velocity at apogee results when the perigee is just outside the atmosphere. The  $\Delta V_c$  between the projectile and the satellite at apogee in its elliptical orbit is given by:

$$\Delta V_c = v_c [\sqrt{2/(1 + R_c)} - \sqrt{2/(1 + R_p)}] \tag{4}$$

where  $R_e$  is the ratio of the perigee to the Earth's radius. Figure 3 shows the  $\Delta V$  required to rendezvous with a satellite in an elliptical orbit with the perigee at an altitude of 550 km. The  $\Delta V$  is much lower than for the circular orbit, for any launch angle, and there is no minimum in the  $\Delta V$  curve. For example, at a  $20^\circ$  launch angle, the  $\Delta V$  at the satellite is only 500 m/s for a satellite orbit apogee of 12,000 km.

The use of an elliptical orbit has another advantage over the circular orbit, in that it comes close to the Space Station altitude. If the orbital parameters are chosen properly, the orbit of the rotating tether can be specified so that the payload is released into the Space Station orbit for storage and later use. The slingsat would not be required to process or store the fuel, and its design could be very simple.

In order to minimize the gun launch velocity, Fig. 2 shows that the gun should be pointed vertically and the satellite should be in the lowest possible orbit. On the other hand, in order to minimize the  $\Delta V$  at the satellite, Fig. 3 shows that the gun should be pointed horizontally and the satellite should be in the highest possible orbit. The latter requirement is the more stringent, and it calls for a low launch angle, perhaps  $20^\circ$ . The interplay of these conflicting requirements on the total  $\Delta V$  is shown in Fig. 4 for a gun at sea level with a muzzle angle of  $21^\circ$ . As the satellite altitude at apogee is increased, the gun launch velocity also increases. The  $\Delta V$  required to drop the payload at the Space Station in LEO also increases, but the  $\Delta V$  required to catch the payload in HEO decreases. The near optimum solution is to design for the  $\Delta V$  at HEO and LEO to be the same. This occurs at an apogee of about 10,000 km. Defining the total  $\Delta V$  as the gun launch velocity plus the larger of the HEO or LEO  $\Delta V$  and plotting it on the figure gives a slight "elbow" in the total  $\Delta V$  curve at the 10,000 km point.

3. LAUNCHER ANALYSIS

The requirements foreseen for upper stage propulsion[2] are about 115 metric tons of water and 10 tons of storable liquid fuel per year. This is a total

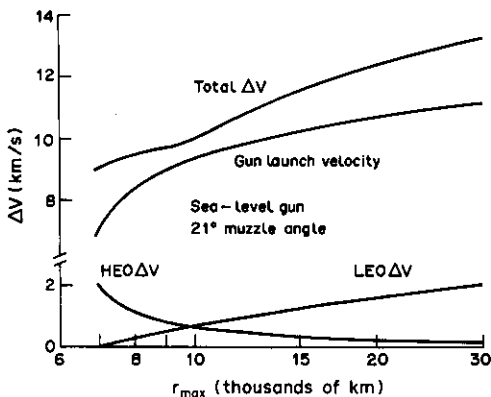


Fig. 4. Total  $\Delta V$  vs slingsat apogee.

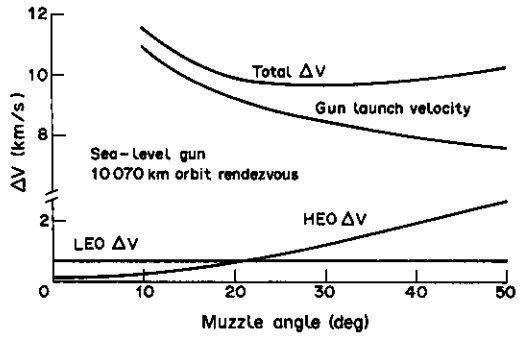


Fig. 5. Total  $\Delta V$  vs launch angle.

of 125,000 kg/year, or 342 kg/day. Allowing for the mass of the structure and heatshield, this can be achieved by the launch of two 300-kg projectiles per day. To be fired in a railgun launcher, the projectile should be in the form of a long, slender tube with a high-temperature nose cone for thermal protection and fins for stability, after Rice *et al.*[13]. It could be fitted with a pair of square sabots to cover the fins and to adapt it to a square-bore railgun. The projectile would have a diameter of about 0.2 m and a length of 6 m. With this size projectile and a well-designed nose cone, the hypersonic drag coefficient could be about 0.1. Selecting 10,000 km as the optimum satellite apogee, we may then look at the effect of gun launch angle on the total  $\Delta V$  required. Assuming the projectile is launched due east at a latitude of  $28.5^\circ$ , it gains 0.409 km/s velocity from the Earth's rotation. The required atmospheric exit velocity  $V_f$  and its angle  $\alpha$  from the vertical are found from the orbit injection velocity  $V_0$  to be:

$$V_f^2 = (V_0 \cos \beta - V_e)^2 + (V_0 \sin \beta)^2, \tag{5}$$

$$\cos \alpha = V_0 \sin \beta / V_f, \tag{6}$$

where  $V_e$  is the Earth's rotational velocity. The muzzle velocity  $V_m$  required to produce a given atmospheric exit velocity  $V_f$  is given in simplified form by:

$$V_m = V_f \exp(\rho C_d A / 2m\gamma \cos \alpha), \tag{7}$$

where  $\rho$  is the atmospheric density,  $1.225 \text{ kg/m}^3$  at sea level,  $C_d$ ,  $A$  and  $m$  are the projectile drag coefficient, frontal area and mass,  $\gamma$  is the atmospheric density variation with altitude,  $1.096 \times 10^{-4} \text{ m}^{-1}$ , and  $\alpha$  is the muzzle angle from the vertical. With a fairly dense projectile of narrow cross-section, the atmospheric losses are quite modest until angles from the vertical of  $70^\circ$  are reached. Firing at this muzzle angle from 2000 m elevation, the total velocity loss in traversing the atmosphere is <12% of the muzzle velocity; firing from sea level, the loss is <14%.

Using these equations, the gun launch velocity required to reach a satellite apogee of 10,700 km as a function of the launch angle from the horizontal is shown in Fig. 5. Plotted at the bottom of the figure is the  $\Delta V$  required at HEO and at LEO, and plotted

at the top is the total  $\Delta V$ . This figure shows that a launch angle of  $20^\circ$  results in the HEO  $\Delta V$  and LEO  $\Delta V$  becoming equal. From Figs 4 and 5 we may select optimum values for the satellite orbit and the launch angle.

An efficiency of 20% can be expected from conversion of electrical energy into projectile kinetic energy[10], resulting in the need for 67 GJ of energy for each projectile. At 2000  $g$  acceleration, the powder required is 144 GW. This power level will require the use of an energy storage device such as a capacitor bank or homopolar generator. Because the peak power is needed for only 0.468 s for each launch, the average daily power required for the launcher is a more modest 1.56 MW for two projectiles per day. With energy storage devices, the off-peak power of the local electric utility would suffice to launch large tonnages into orbit. The peak power required could be reduced greatly by using several multi-armed satellites spaced throughout the same orbit and firing smaller projectiles several times per day, while still providing a total of 342 kg/day of payload into orbit.

Because of rail erosion at low payload velocities, railguns need a pre-accelerator to inject the projectiles. A compressed gas system could be used to accelerate the projectile to about 1 km/s, and then the railgun would accelerate it to its final velocity of 9.18 km/s. At 1000  $g$ , the pre-accelerator would be just 50 m long; at 2000  $g$ , the railgun would be 2000 m long. This arrangement could be assembled on a mountainside, preferably with an elevation of about 2000 m at the muzzle. If range safety required it, however, the launcher could be sea-based and fire over the open ocean, at the cost of a higher muzzle velocity.

#### 4. SLINGSAT ANALYSIS

The catching/throwing satellite, or slingsat, is designed to rotate its long tether at an angular velocity such that the end moves with the same velocity as the incoming payload. The end attachment can then catch the payload and hold it against the radial acceleration. For release of the payload near the Space Station, the rotational velocity is adjusted to match the station orbital velocity. The slingsat can operate at a constant angular velocity if the  $V$  required to catch the payload is equal to the  $\Delta V$  required to throw the payload.

Designing the rotating arms for constant stress results in an exponential taper from the root to the tip and minimizes the mass of the arms[14]. Figure 6 shows the ratio of the arm mass to the payload mass and the ratio of the arm root cross-sectional area to the arm tip area as functions of the relative velocity of the incoming payload. These values are based on the properties of Kevlar 49 as the basic arm material[15]. To catch a 300-kg payload at the design velocity of 670 m/s is well within the capabilities of Kevlar 49, and the required mass ratio is only 2.29.

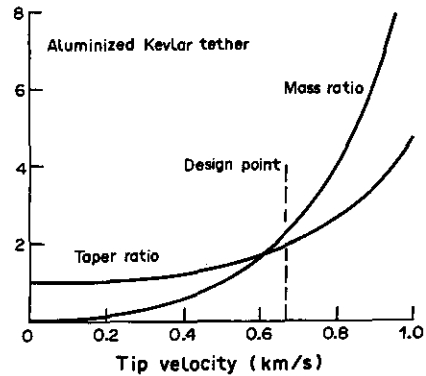


Fig. 6. Slingsat taper ratio and mass ratio vs tip velocity.

The required taper in cross-sectional area is 2.02 from the root to the tip.

The overall satellite design would consist of a central satellite for power, communications and orbit determination, one or more pairs of 23-km-long tapered arms supporting Kevlar nets 10–20 m across to catch the payload, and end attachments to hold the payloads firmly for later release. Such long arms are necessary to limit the radial acceleration of the payload at the tip to  $2g$  and to simplify the rendezvous and payload catching. Because Kevlar deteriorates under u.v. radiation, it is necessary to coat the arms with a layer of protective material such as aluminum. This layer also serves as a conductor to carry an electric current. Such a current in an extended spacecraft interacts with the Earth's magnetic field and creates forces on the spacecraft[16]. Using the arms independently for reversible currents allows the satellite to change its rotational velocity and its orbital velocity. By proper phasing of the current, the satellite can also control its inclination, line of apsides, and eccentricity[15].

The rotating satellite will change its orbital velocity as the result of catching and throwing payloads. For a satellite with a mass of 300 kg at the center, 1600 kg in the rotating arms, and 50 kg at each end attachment, the velocity is decreased by 87 m/s at each 300-kg payload capture and increased by 101 m/s at each release. The net result is a slight gain in energy over a complete catch/throw sequence, because the payload on its gun-launched trajectory has more orbital energy than its final orbit near the Space Station. The electrodynamic tether can be used to bleed off this excess energy to produce power for the satellite. This will bring the satellite back to the proper initial conditions for the next catch or throw and keep it in proper orbital phase with the gun and the Space Station.

The mechanics of catching and throwing payloads will induce dynamic loads into the satellite arms, resulting in lateral and longitudinal vibrations[17]. These dynamic disturbances have not been investigated in detail, but they appear to be well within the margin of safety for the Kevlar arms. The possibility of unstable resonances of the satellite structure,

however, may require that some care be applied in the choice of arm length, payload mass, and tether taper ratio. The long arms of the slingsat create a large cross-section for meteoroid collision. This can be minimized by making the arms in the form of hollow tubes with a diameter of at least 8 mm at the tip. This will provide an estimated time between failures due to meteoroid collision of more than 6 years[15].

## 5. OPERATIONS

Let us review briefly the concept of operation of the gun-tether launch system. Example design parameters for the system are shown in Table 1. The gun is located at a latitude near  $28.5^\circ$  north, preferably on a mountainside with the muzzle at high altitude. A mountain peak in the Sierra del Nido range in northern Mexico has the proper latitude and height for a 2000-m muzzle location. The gun fires directly east, making maximum use of the Earth's rotation. The muzzle is inclined upward at  $21^\circ$ , giving a flight path angle of  $20^\circ$  after atmospheric exit. The 300-kg projectile leaves the muzzle at 9.175 km/s and leaves the atmosphere at 8.431 km/s. This velocity is just sufficient to take it to an altitude of 10,070 km, where it will rendezvous with the lower arm of the rotating tether. Alternatively, the muzzle can be located at sea level at the Kennedy Space Center and fire at a somewhat higher velocity of 9.375 km/s. From either of these locations, a projectile missed by the tether would impact in the Indian Ocean southeast of Madagascar, away from any inhabited areas.

The slingsat is placed into a  $28.5^\circ$  inclined elliptical orbit with perigee of 550 km and apogee of 10,093 km, in order to approach the Space Station orbit in the same plane. The unperturbed orbit has a nominal period of 1/11 of a day. With adjustment for nodal regression, this gives one opportunity per day for receiving payloads. Each satellite receives one payload per arm per day. A two-armed slingsat would catch two payloads spaced about 4 min apart, and a four-armed slingsat would catch 4 consecutive

payloads at about 2-min intervals. Spacing three satellites around the orbit would allow daily launches in spite of the rotation of the line of apsides caused by the oblateness of the Earth. At all times, then, one of the three satellites would be near apogee as it came into position for projectile launch.

The firing time is adjusted so that the projectile intercepts the end of the satellite arm and is caught in the Kevlar net. The projectile will require some guidance and small velocity changes to ensure hitting the target. The projectile is then firmly attached to the end of the tether. The satellite uses its electrodynamic tether to produce accelerating forces on the arms of the satellite, adjusting the orbit to make up for the velocity changes caused by catching and throwing payloads. The satellite then waits for the proper orbit phasing to drop the payload at the Space Station. When the satellite drops a payload at the Space Station, it gains orbital energy; this energy can be used to produce power through the tether until the orbital velocity is lowered sufficiently to bring it back into the proper orbit for the next catch.

The Space Station crew finds the fuel payload orbiting slightly above them and moving more slowly. As the Space Station passes a payload, a short tether can be extended upward to catch the fuel and bring it down to the station. In doing so, the payload exerts some force on the station, raising its orbit slightly. This can be used for drag makeup. For economy, the tungsten nose cone and fins, or the entire empty projectile, can be returned via cargo shuttle to the gun for re-use. This will limit the number of heat-resistant nose cones required. The payload can include all the station's needs for water, all upper-stage fuel, stationkeeping propellant, and other high-g supplies such as radiation shielding. Some lower-g-tolerant materials also might be shipped if they were immersed in a water tank for protection from the gun acceleration.

The cost for launching Earth-orbit payloads by the gun-tether combination could be two orders of magnitude below current cost. The energy cost is estimated to be \$2/kg of payload, the cost of the projectiles (including re-usable nose cones) is estimated to be another \$2/kg, and the operations costs could be reduced to the range of \$6/kg of payload by using a very small crew of a dozen people. Not counting capital development costs, the total launch costs could then approach \$10/kg of payload.

## 6. CONCLUSIONS

The combination of electromagnetic gun and rotating tether, or slingsat, has the potential for carrying all the fuel for upper stage rockets now envisioned for the NASA program. In addition, the system could transport high-g payloads into orbit for the Space Station and for construction projects in Earth orbit. By putting the fuel tanks in holding orbit somewhat above the space station, the very act of retrieving

Table 1. Example launcher and slingsat design parameters

Launcher		Slingsat	
Muzzle velocity	9.175 km/s	Satellite	300 kg
Muzzle angle	$21^\circ$	Two arms	1600 kg
Acceleration	2000 g	Two catchers	100 kg
Shot time	0.468 s	Total mass	2000 kg
Peak power	144 GW	Payload mass	300 kg
Average power	1.56 MW	Length	45.8 km
Latitude	$28.5^\circ$ N	Tip area	$13.8 \text{ mm}^2$
Altitude	2000 m	Root area	$27.9 \text{ mm}^2$
		Arm radius	22.9 km
		Taper ratio	2.017
		Mass ratio	2.287
<i>Projectile</i>		Tip velocity	670 m/s
Length	6 m	Tip accel.	2 g
Diameter	0.2 m	Orbit inclin.	$28.5^\circ$
Payload	190 kg	Perigee	6928 km
Structure	40 kg	Apogee	10,093 km
Nose cone, fins	60 kg	Period	1/11 day
Guidance	10 kg	Arm material	Kevlar
Sabots	20 kg	Meteor life	6.2 yr
Total mass	320 kg		
Drag coeff.	0.1		

them boosts the station orbit and helps in drag makeup. Small payloads could be launched frequently for not much more than the electrical energy costs of \$2/kg. The power required at the launch site is not large compared with existing power consumers. The entire system promises launch costs of \$10/kg into low Earth orbit (LEO) or high Earth orbit (HEO). The gun/slingsat combination puts minimum requirements on both the gun and slingsat capabilities, and is capable of payload growth to hundreds of tons per year by using several multi-armed satellites spaced about the nominal orbit. The system requires high launch accelerations, and thus is limited to unmanned payloads. It could be used in conjunction with an advanced manned launch vehicle for a complete, low-cost space transportation system. In contrast to conventional or laser-powered rockets, the gun/slingsat launching system also would be essentially pollution-free, except for the noise of the sonic booms during projectile launches.

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