ANALYSIS OF INTERSTELLAR SPACECRAFT CYCLING BETWEEN THE SUN AND THE NEAR STARS

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This paper explores what cyclic trajectories are possible within the limits of known laws of physics and practical propulsions systems for a spacecraft or world ship that could travel between the Sun and the near stars periodically. Because of the long durations, the spacecraft is assumed to be massive to house many people for many generations. The spacecraft is initially accelerated up to speed at great expense but from then on only minimal propulsion is assumed to be available for course corrections. The spacecraft, as it nears a star, follows a hyperbolic trajectory. The spacecraft returns by using gravity to “swing” around each of three or more stars, one of which is the Sun. The minimum distance of closest approach found during the flyby was 3 Sun radii, where the heat flux of 7 MW/m^2 was shielded from the spacecraft by a radiation-cooled shield made of a porous woven carbon fiber whose peak temperature was about 2500 K. A thin sheet of graphite by contrast would have a temperature of 2800 K and high evaporation mass loss rate. The coasting speed was found to be 115 km/s or 0.0004c. The minimum period found for this class of trajectory was about 41,000 years for a trip around three stars with a 16 light-year closed path and about 57,000 years for four stars with a 22 light-year path length. If an order of magnitude more heat flux could be handled somehow, then the spacecraft could just skim the surface of the Sun (200 km/s or 0.0007c) giving a minimum cycle time of 24,000 years for three stars and 33,000 years for fours stars.

Keywords: Cycling interstellar spacecraft, cyclic interstellar orbits, minimum travel time interstellar orbits, world ship cyclic trajectory

Notation List

α = interior half-angle of hyperbola, (see fig.1)
a = semi-major axis (see fig. 1)
AU = distance from earth to Sun=1.50 x 10^{11}m=500 light-sec
A = absorbtivity
c = speed of light= 3.00 x 10^8 m/s
Δ = deflection angle (see fig. 1)
ε = eccentricity (see fig. 1)
E = emissivity
$E_{cohesive}$ = energy to heatup and evaporate (MJ/kg)
G = 6.672 x 10^{-11}m^{3}kg^{-1}s^{-2}

$$H = \frac{\lambda_\varphi}{4\pi r^2} = solar\ \ flux\,(W/m^2)$$

h = (non) impact distance (see fig. 1)
J = evaporation rate (kg•m^{-2}•s^{-1})
k = constant of motion = 2 x total energy per unit mass
L = angular momentum per unit mass
1 Light-year = 9.5 x 10^{15}m = 6.3 x 10^4AU
λ_Θ = 3.92 x 10^{26} W = total solar radiation
M_Θ = 2.00 x 10^{30}kg = mass of the Sun
$M_\oplus$ = 6.0 x 10^{24}kg = mass of the Earth
$\mu \equiv GM_\Theta$ = 1.33 x 10^{20}m^3/s^2
σ = Stefan-Boltzmann constant = 5.67 x 10^{-8} Wm^{-2}K^{-4}
p = semi-latus rectum (see fig. 1)
q = total solar flux during time spent in high heat flux

$$\Phi = polar\ angle$$

$$\dot{\Phi} = \frac{d\Phi}{dt}$$

ϕ_∞ = 180°-α (see fig. 1)
R_Θ = 6.96 x 10^8m = 4.64 x 10^{-3}AU = radius of the Sun
r_p = distance of closest approach on a flyby at perigee
r = radius, m
t_H = time spent in high heat flux
v_∞ = speed during coast period between stars
v_p = speed at perigee

1. INTRODUCTION

This paper explores what cyclic trajectories are possible for travel between the Sun and the near stars. The spacecraft is assumed to be very massive in order to house a large number of people for many generations. Owing to the long duration of such a spacecraft flight it will have to be a fully functional and self-contained “world” and therefore has been called a “world ship” [1]. The spacecraft is initially accelerated up to speed at great expense but from then on only minimal propulsion is assumed to be available for course corrections because of its large mass. The spacecraft, as it nears a star, follows a hyperbolic trajectory as shown in fig. 1. The spacecraft returns by using gravity to “swing” around each of three or more stars in

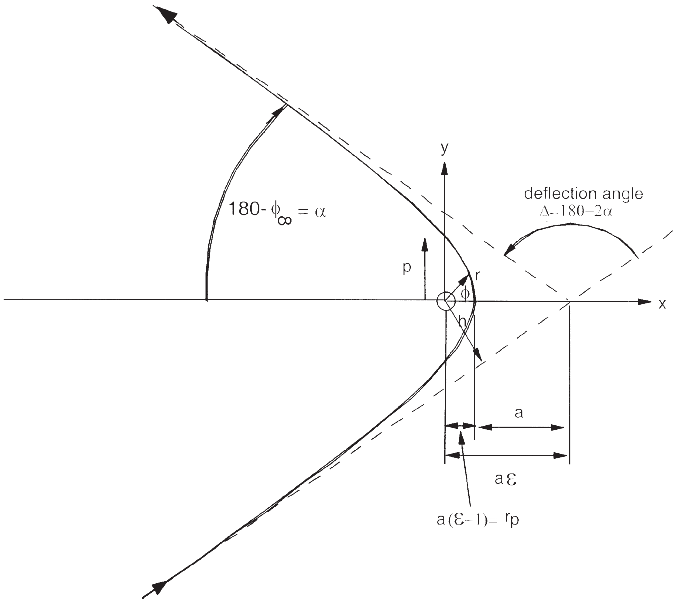

Fig. 1. Hyperbolic trajectory around a star.

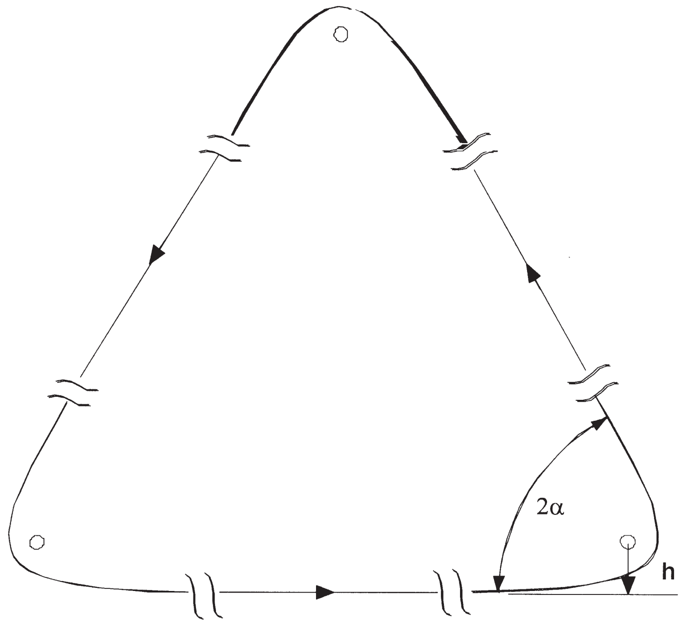

Fig. 2. Hyperbolic trajectories around three stars.

the circuit, one of which is the Sun as shown in fig. 2. If there are three stars in the circuit, a deflection of 120° on average for each passage is required and 90° for a four-star case. To get a deflection this large requires a close encounter flyby. During the flyby of a few hours the heat flux is large and the spacecraft must be shielded to prevent overheating. To get shorter cycle time requires higher speeds that in turn require a closer encounter in the flyby with a higher heat flux to get the same deflection. So there is a trade-off between cycle time and heat shielding, which is the subject of this paper. The concept of cycling space ships between planets similarly has the goal of minimizing propulsion requirements [2].

2. TRAJECTORIES NEAR THE SUN

This section analyzes the trajectory of a spaceship as it passes near the Sun. In particular, it focuses on unbound trajectories that are deflected by the Sun's gravity, but then return to outer space.

A treatment by Szebehely [3] using standard notation common in celestial mechanics texts is used. Total energy per unit of mass, (1/2)k, is a constant of the motion because the gravitational force is derivable from a potential. The angular momentum per unit mass, L, is also a constant of the motion because the force is directed toward the center and exerts no torque on the spaceship. In polar coordinates, these two conservation equations are:

$$k = \dot{r}^2 + \left(r\dot{\phi}\right)^2 - \frac{2\mu}{r} \quad and \quad L = r^2\dot{\phi} \tag{2.1}$$

Therefore:

$$\dot{\phi} = \frac{L}{r^2} \quad and \quad \dot{r} = \frac{dr}{dt} = \frac{dr}{d\phi}\frac{d\phi}{dt} = \dot{\phi}r' \quad where \quad r' \equiv \frac{dr}{d\phi} \tag{2.2}$$

Eliminating t gives:

$$\frac{L^2}{r^4}r'^2 + \frac{L^2}{r^2} - \frac{2\mu}{r} = k \tag{2.3}$$

Substituting u=1/r gives:

$$u'^2 + u^2 - \frac{2\mu u}{L^2} = \frac{k}{L^2} \quad with\ solution$$

$$u(\phi) = \frac{\mu}{L^2}\left(1 + \sqrt{1 + \frac{kL^2}{\mu^2}}\cos\phi\right) \tag{2.4}$$

Where $\mu \equiv GM_\Theta$ = the gravitational constant times mass of the star (Sun). With r=1/u, this becomes:

$$r = \frac{p}{1 + \varepsilon\cos\phi} \quad where \quad \varepsilon = \sqrt{1 + \frac{kL^2}{\mu^2}}, \quad p = a\left(\varepsilon^2 - 1\right), \quad a = \frac{\mu}{k} \tag{2.5}$$

Here, a = μ/k is the semi-major axis (See Fig. 1) and the distance of closest approach is $r_p = a(\varepsilon - 1)$. These equations define a hyperbola when ε > 1, and of an ellipse when ε < 1. Therefore, positive total energy, k > 0, results in a hyperbolic trajectory with $k = v_\infty^2 = \mu/a$ and $r \rightarrow \infty$ when $\phi \rightarrow \phi_\infty = \cos^{-1}(-1/\varepsilon)$. A related parameter is the (non)impact parameter, h = aε sin ϕ_∞ (See Fig. 1). The parameter, h, is important because it should be measurable and controllable by course correction propulsion from within the spacecraft, and h and v_∞ determine ϕ_∞. The angle swept out by the spaceship as it passes a star is the deflection angle Δ shown in fig. 1. Δ = 180°-2α.

At perigee (the distance of closest approach),

$$\phi = 0, \quad \dot{r} = 0, \quad and \quad r = r_P = \frac{p}{1+\varepsilon} = a\left(\varepsilon - 1\right) \tag{2.6}$$

$$\varepsilon = 1 + r_p / a \tag{2.7}$$

Also at perigee, using the conservation of energy

$$v_P = r\dot{\phi} = \sqrt{k + \frac{2\mu}{r_P}} = \sqrt{v_\infty^2 + \frac{2\mu}{r_P}} \tag{2.8}$$

It is at perigee, the point of closest approach, that the heat load might be a problem. The heat flux at distance r_P from the center of the Sun is $H = \lambda_\Theta/(4 \pi r_P^2)$, and the time spent in this high heat flux is approximately $t_H \approx \pi r_P / v_P$. Therefore, the total heat load per unit of area, q, is $q = H t_H \approx \lambda_\Theta/(4 r_P v_P) = \lambda_\Theta/(4 h v_\infty)$, since $r_P v_P = h v_\infty = L$, the angular momentum about the Sun per unit of mass.

To illustrate the calculations required, an example is given. Choose $v_\infty = 0.0003 \times c = 1.0 \times 10^5$ m/s and $r_P = 3 \times R_\Theta = 2.09 \times 10^9$ m.

Calculate $k = v_\infty^2 = 1.0 \times 10^{10} m^2/s^2$, $a = \mu/k = 1.33 \times 10^{10}$m, $v_P^2 = v_\infty^2 + 2m/r_P = 1.37 \times 10^{11} = (3.73 \times 10^5 \text{ m/s})^2$, and $\varepsilon = 1 + r_P/a = 1.16$. This value of ε gives $f_\infty = \cos^{-1}(-1/\varepsilon) = \cos^{-1}(-0.866) = 150°$. That is, the full interior angle of the hyperbola is 2 x 30° = 60°. The deflection angle is 120°, requiring three star encounters to complete a roundtrip.

The heat flux at distance r_P from the center of the Sun is $H = \lambda_\Theta/(4 \pi r_P^2) = 7.4$ MW/m² for r_P=3 Sun radii. The time spent in this high heat flux is approximately $t_H \approx \pi r_P/v_P = 4.8$ hr, which results in about 1.3 x 10^{11} J/m² incident on the ship. The results are given in Table 1, case No. 1, where the units are: velocity (km/s), distance (10^{10} m), angle (°), Heat flux (MW/m²), time (h), and Total Heat (10^{10} J/m²).

These heat fluxes, H, can be compared to some practical cases. First, the maximum on earth is 1.39 x 10^3 W/m². Second, water-cooled copper heat dumps for high power ion beams can tolerate 5 x 10^6 W/m² in steady state. And, third, radiatively cooled tungsten wires immersed in an ion beam will rise to a temperature of about 2000°C when the beam deposits about 1 x 10^6 W/m² on their projected surfaces. Cases 2 and 3 in Table 1 result in heat fluxes that are too high to be tolerated even for some of the rather short times, t_H. The ability to shield against heating while in near passage with a star is discussed in Section 5, where Case 4 is shown to be practical. The Case 4 trajectory is shown to scale in figs. 1 and 2.

The time in years for the spaceship to travel 1 light-year is $t = c/v_\infty$, where c = 300,000 is the speed of light in these units or a few thousand times our speed of 115 km/s. It would take twelve thousand years for the spacecraft to travel to even the nearest stars, the Alpha Centauri triplet, which are 4.3 light-years away.

In section 2.1, we treat examples of cycling around four stars, one being the Sun; in section 2.2, three stars and in section 2.3, the general case of the deflection angle for stars of various mass and luminosity type.

2.1 An Orbit Around Four Stars

A spaceship with speed of a few 10^{-4} times c must pass close to a Sun-like star in order to be deflected by even as much as 120° (a=30°). Such a close encounter with a star would cause severe heat problems, and any triangular orbit around any of the near stars would require at least one deflection of about 120°. For that reason we looked for 3 nearby stars that, along with the Sun, form a small near-regular quadrilateral as shown in Fig. 3.

All of the 12 stars that lie within about 10 light-years from the Sun lie within ±7.7° of a plane (RA=17.7°) that includes the Sun. This surprising fact allows us to ignore the small displacements out of the plane, and to plot them on that plane and look for the smallest quadrilateral that contains no internal angle as small as 60° (α = 30°) as shown in Table 2 and fig. 3. The smallest angle is the Sun's at 34°. The 4 stars - our Sun, α Centauri A, Sirius A, and Wolf 359 – define the smallest good orbit. The total path length is 21.7 light-years, which means that the period of the orbit would be 57,000 years for a spaceship traveling at 115 km/s (0.00038 times the speed of light) and passing the Sun at about 3 Sun radii at closest approach (case 4, Table 1). In section 5, a practical heat shield is discussed for case 4, Table 1 at 3 Sun radii during the 5 hours of passage close to the Sun. If the spaceship just skimmed the Sun's surface [case 2, Table 1] (clearly a limiting case of extreme heat shielding), the speed would be 198 km/s (0.00066c) and the roundtrip time would be 33,000 years.

In reality, the relative motion of the stars needs to be accounted for, including the speed increment to our spacecraft on each passage. However, in order to arrive at an approximation to the roundtrip time in all cases, the relative motion between stars is ignored for the present work. Course corrections are made to adjust the distance of closest approach, r_p by measuring h, and the angle α, appropriate to each pair of stars as shown in fig. 2.

2.2 An Orbit Around Three Stars

Using 3 stars as shown in fig. 2, one being the Sun, the other two being white dwarfs we get a shorter path length. For Sun-Proxima-Barnard's Star we have a path length of 15.6 light years and a speed of 113 km/s giving r_p=3 Sun radii, the cycle time is 41,000 years. The path including the Sun, Proxima and Alpha Centauri is precluded as it requires close to 180° deflection angle on the solar flyby.

If we consider three white dwarfs not including passage by the Sun, for example, Barnard's Star-Proxima-Sirius B, we have a path length of 15.7 light years. The small angle at Proxima of alpha = 35.5° gives v_∞ = 162 km/s with r_p = 0.035 R_Θ and a cycle time of 29,000 years. The radii of these white dwarfs are small, and the energy given off is small enough that the radiation shield discussed later is less demanding than to pass 3 radii from the Sun.

2.3 Deflection Angle for Various Stars

The equations have been solved from the point of view of a navigator who wants to plot a course. He can pass close to a low luminosity star either just skimming the surface if the heat flux

TABLE 1: *Hyperbolic Trajectories.*

No.	v_∞	h/R_Θ	r_P/R_Θ	v_P	Δ	ε	a	H	t_H	q
1	100	11	3	373	120	1.16	1.3	7.4	4.8	13
2	198	3.3	1	649	112	1.21	0.34	64	0.94	22
3	140	7.6	2	459	112	1.21	0.68	16	2.6	15
4	115	9.8	3	375	112	1.21	1.01	7	4.9	13

TABLE 2: *Location of the Closest Stars.*

Star	Dist.(ly)	RA(°)	Dec.(°)	Class	AbMag	x (ly)	y (ly)
Sun	0	0	0	G	4.8	0	0
Proxima	4.2	21.75	-62.7	M	15.5	1.93	-3.73
α Centauri A	4.4	22.0	-60.8	G	4.3	2.15	-3.84
α Centauri B	"	"	"	K	5.7	"	"
Barnard's Star	5.9	27.0	+4.7	M	13.2	5.88	+0.48
Wolf 359	7.8	16.5	+7.0	M	16.6	7.74	+0.95
BD+36°2147	8.3	16.5	+36.0	M	10.5	6.71	+4.88
Sirius A	8.6	10.1	-16.7	A	1.5	8.24	-2.47
Sirius B	"	"	"	DA	11.3	"	"
Luyten 726-8 A	8.7	2.5	-18.0	M	15.4	8.27	-2.69
Luyten 726-8 B	"	"	"	M	15.8	"	"
Ross 154	9.7	28.3	-23.8	M	13.0	8.88	-3.91
Ross 248	10.3	35.6	+44.2	M	14.8	7.38	+7.18

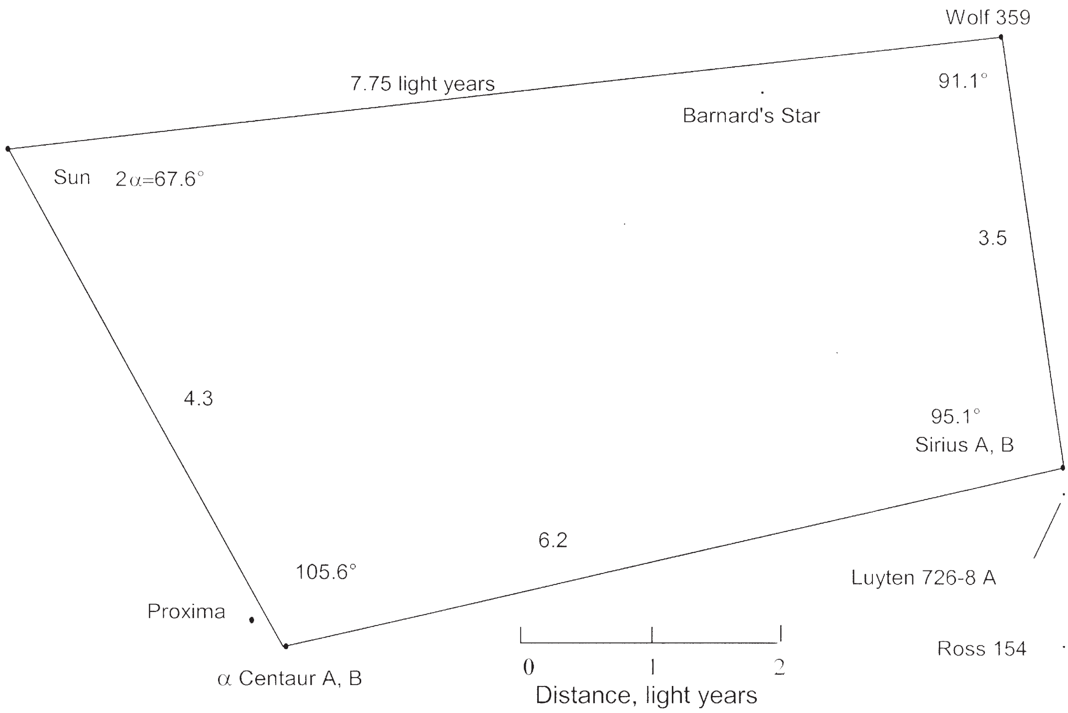

Fig. 3 Trajectory around four stars with a round-trip period of 57,000 years.

is less than 7 MW/m^2 or out to a radius such that the heating is about 7 MW/m^2 that can be handled by the heat shield, as will be discussed in Section 5. Using the mass, luminosity and radius of the near stars one solves for the deflection angle versus v_∞ [4]. Remember the average deflection angle needed for a roundtrip for three stars is 120° and for four stars is 90°.

The deflection angle is derived from Eq. 2.5 and 2.6:

$$\Delta = -180^o + 2 \cdot \cos^{-1}\left(\frac{-1}{\frac{r_p \cdot v_\infty^2}{m \cdot G} + 1}\right) \tag{2.9}$$

Notice the trajectories are self-similar as long as the ratio r_p/m is the same, where m is the mass of the star. The results are plotted in fig. 4 for the closest approach, set by skimming the surface or maximum heat load of 7 MW/m^2, whichever is larger. The ratio of R_p/m is given in units of Sun radius and Sun mass.

These examples are low luminosity white dwarfs except for the Sun. Sirius-B being simultaneously low luminosity and small radius results in the largest deflection. If low radiation (<7MW/m^2 at perigee) neutron stars or black holes were available, large deflections could be obtained.

3. HOW MUCH DOES IT COST TO ACCELERATE MASS TO A SPEED V AND HOW LONG DOES IT TAKE?

How much does it cost to accelerate mass to a speed of 115 km/s or 3.8x10^{-4} times the speed of light, which is the assumed speed in calculations used elsewhere? A spaceship where people live for thousands of years has to be a fully functional "world", a world ship. The mass must be much more than a ton per person, the mass of an ordinary car. Let's suppose the mass is 10 tons per person. To get the spaceship up to speed, use of a mass beam [5] is assumed. Other propulsion methods such as nuclear-pulsed propulsion are possible. Small masses are accelerated toward the spacecraft from space-based beam projectors with their independent power station. These then seek the center of a laser beamed both from the

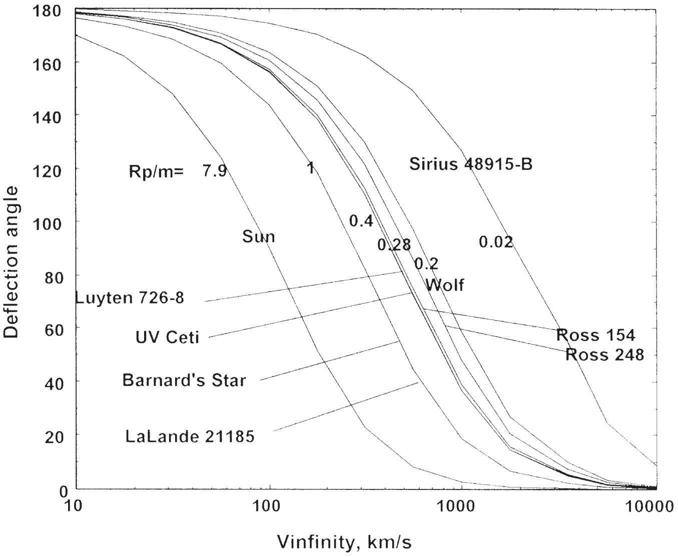

Fig. 4 Deflection angle versus speed, v_∞.

target spacecraft and from the mass beam projector rather than spread out [6]. The masses are vaporized and/or ionized as they approach the spaceship and made to "bounce" off a reflector (a physical or magnetic "sail"). The beam velocity is gauged so that the reflected mass is left with negligible kinetic energy in the solar frame of reference–the kinetic energy of the beam having been efficiently transmitted to the spacecraft. In other words the jet efficiency is assumed ideal at 100%. Suppose power-limited electric propulsion is used and the expellant mass is externally supplied. The mass beam projector, its power supply and source of mass would be space based.

$$F = M\frac{dV}{dt} = 2(v-V)\frac{dm}{dt} \qquad (3.1)$$

where M is the fixed mass of the ship and V is the spaceship speed and v is the variable exhaust speed. We will assume the ideal case of elastic rebound where v=2V. A lesser value will increase the required beam power and require heat dissipation on the spacecraft.

$$v = 2V$$

$$\dot{m} = \frac{dm}{dt} = \text{mass flow rate} \qquad (3.2)$$

$$P = \frac{1}{2}\dot{m}(2V)^2 = \text{power in the flowing mass} \qquad (3.3)$$

$$F = M\frac{dV}{dt} = 2(v-V)\dot{m} = 2V\frac{P}{2V^2},\ or\ MVdV = Pdt \qquad (3.4)$$

Integrating, assuming P = constant.

$$E = \frac{1}{2}MV^2 = Pt \qquad (3.5)$$

Using Table 1 case 4 example of v_∞=115 km/s

$$E/M = \frac{1}{2}MV^2/M = Pt/M = 0.5\cdot(1.15\cdot10^5)^2 = 6.6\cdot10^9\ J/kg \qquad (3.6)$$

The cost, assuming a cost of electricity typical of bus bar cost of 0.05 \$/kWh and an efficiency of electricity to acceleration is 1/3, is given next:

$$Cost/M = \frac{1}{1/3}\cdot\frac{6.6\cdot10^9\ J/kg\cdot0.05\$/kWh}{1000\frac{W}{kW}\cdot3600\,s/h} = 270\ \$/kg \qquad (3.7)$$

Owing to the ideal assumptions on elastic rebound and other assumptions, the actual cost will be considerably higher. For comparison, the cost per unit mass into low earth orbit today in the shuttle is about \$20,000/kg or almost 100 times more. If each person needs 10 tons, then the cost would be $2.7'10^6$ \$/person and \$27 billion for the 10,000 population. This is about the cost of the Apollo moon project for propulsion alone. For comparison, World War II cost the US about \$4.3 trillion. Clearly, there will be incentives to economize on mass per person and efficient acceleration tricks such as using the gravity "slingshot" effect commonly used by NASA even today where an increment of velocity up to that of the object of the flyby is picked up.

The time to accelerate is:

$$t = \frac{\frac{1}{2}MV^2}{P} \qquad (3.8)$$

The power supply is assumed to be space based but some course correction power on board is needed as well as for running the society (lighting, heating, agriculture, communications, etc). Suppose the power source on board supplies power at 1 kW/kg (including all the mass of the ship and its expellant), which might be an upper limit of technology, then the time to accelerate is

$$t = \frac{\frac{1}{2}MV^2}{P} = \frac{6.6\cdot10^9\ J/kg}{1{,}000W/kg\cdot0.333} = 2.0\cdot10^7\ s = 0.62\ years \qquad (3.9)$$

If each person needs 10 tons of mass, then each person's pro-rated power would be 10 MW per person. For a spaceship of 10,000 people, the total power would be 100 GW for the 10 tons per person case. This power is very large and can be decreased by prolonging the acceleration period from the 0.62 years to 62 years, for example, which would lower the power to a more practical 1 GW for the 10-ton case. This would be a practical 10 W/kg or 100 kW per person.

What expellant mass is needed to propel the ship? For constant P, the rate that mass must be supplied is

$$\dot{m} = \frac{P}{2V^2} = \frac{M}{4t} \qquad (3.10)$$

This expellant mass is supplied to the rocket at the time it is needed at speed V. (Note that $\dot{m}$ —>∞ as t——> 0.)

Integrate to find the total expellant mass:

$$\int dm = \int\frac{M}{4t}dt,\ \text{or}\ m = \frac{M}{4}\ln\frac{t}{t_0} \qquad (3.11)$$

This expression shows it takes a lot of expellant mass to get the spaceship started but $\dot{m}$ decreases rapidly as V increases, although P remains constant. The time to accelerate to the intermediate speed of 7 km/s, which is 2x 10^{-5} c, is 56 hr at the lower power of 1 GW. For comparisons, the approximate speed of a low earth orbit is 7 km/s (2x10^{-5} c) and speed of the earth around the Sun is 30 km/s (10^{-4} c) and 42 km/s to escape the Sun's gravity from earth's position. Using 56 hr for t_0, the expellant mass needed to accelerate to 115 km/s (3.8x 10^{-4} c) in 62 y is M/4xLn(100) =1.15 M. Again, this shows that it will be important to economize on expellant mass, especially in the early stages of acceleration. We must add a small amount to the above estimates of power cost and expellant mass needed to escape the Sun's gravity.

4. COURSE CORRECTIONS

The next question is, what is the amount of course correction practical with onboard power?

Assume that the correctional thrust is at right angles to the direction of travel so that no speed change occurs. Also assume that the expellant used is small compared to the ship's mass.

$$\dot{m}t << M$$

$$F = M\frac{dV}{dt} = v_{jet}\dot{m} \tag{4.1}$$

Where dV is the resultant perpendicular velocity, and v_{jet} is expellant mass speed relative to the ship.

$$M\Delta V = v_{jet}\dot{m}t \tag{4.2}$$

$$\frac{1}{2}\dot{m}v_{jet}^2 = \eta P \tag{4.3}$$

Where now P is the power consumed to produce the steering jet.

$$\frac{\Delta V}{V} = \frac{v_{jet}\dot{m}t}{MV} = \frac{\frac{1}{2}\dot{m}v_{jet}^2 t}{\frac{1}{2}MV \quad v_{jet}} = \frac{P\eta t}{\frac{1}{2}MV^2 \frac{v_{jet}}{V}} \tag{4.4}$$

$$\frac{\Delta V}{V} = \frac{10^8\,W\cdot 0.333\cdot 100y\cdot 3.15\cdot 10^7\,s/y}{6.6\cdot 10^9\,J/kg\cdot 10{,}000\,kg/person\cdot 10{,}000\,people\frac{1}{1}} = 0.16 \tag{4.5}$$

When v_{jet}=V and efficiency is 33%.

For 10 tons/person, 10,000 people, and 100 MW power for 100 years, this corresponds to a deflection of 9° for a 100-year "burn" course correction. The course corrections should be much smaller than this. The mass expended in this assumed 9° correction is:

$$\dot{m}\,t = \frac{2\eta P}{v_{jet}^2}t = \frac{2\times 0.33\times 10^8\,W\times 100\,y\times 3.15\times 10^7\,s/y}{(1.15\times 10^5\,m/s)^2}$$
$$= 1.6\times 10^7\,kg = 16{,}000\,tons \tag{4.6}$$

$$\frac{\dot{m}t}{M} = \frac{16\times 10^3}{10^4\times 10^4} = 1.6\times 10^{-4}$$

The mass used for steering is much smaller than M.

For 50 MW power, $\frac{\Delta V}{V} = 0.01$ and

$$\frac{\dot{m}t}{M} = \frac{2.2}{20}\times 10^{-4} = 1\times 10^{-5}$$

which might be adequate for course corrections and uses negligible mass as expellant.

A correction that brings the ship back to the star from which it just came would require

$$\frac{\Delta V}{V} = 2$$

Rather than 100 MW for 1000 years, we would need (2/0.16=13) 13 times this much rocket power or duration. The expellant mass would still be small. The point to be made here is the impracticality of speeding up then slowing down for the flyby with such a massive space ship in order to reduce significantly the cycle time.

5. RADIATION SHIELDING FROM THE SUN DURING FLYBY

The spacecraft will need shielding from the Sun's (or star's) electromagnetic radiation during the close encounter flyby. One way to do this is put up an umbrella-like thin sheet between the Sun and the spacecraft. The solar flux will intercept the shield, which in turn radiates from both sides. Then the spacecraft can be located some distance away in the shadow. If this reduction is not enough, another shield can be imposed to successively reduce the heating to any desired value, and is only limited by the weight of the shields. The limiting technology then is for the first shield to survive. We do not discuss shielding from stellar particle emissions during flare events, which could coincide with stellar flybys.

A NASA project called Solar Probe planned to use a graphite shield for flyby at four Sun radii [7]. The heat shield is discussed in several reports with emphasis on temperature limits set by radiation and evaporation [8, 9].

The total solar flux is 3.92x10^{26} W and the Sun's radius r_o is 6.96x10^8 m. The solar flux is then 64.4 MW/m^2 at the surface and falling as $(r_o/r)^2$.

The input power from the incident solar flux must be balanced by radiation from the thin disk of a shield assuming total absorption. The radiation formula is:

$$\frac{P}{A} = \mathrm{E}\sigma T^4 \tag{5.1}$$

The emissivity E is a property of materials, σ is the Stefan-Boltzmann constant (5.67x10^{-8} Wm^{-2}K^{-4}) and T is temperature of the surface in Kelvin.

The temperature of the surface of the shield is then:

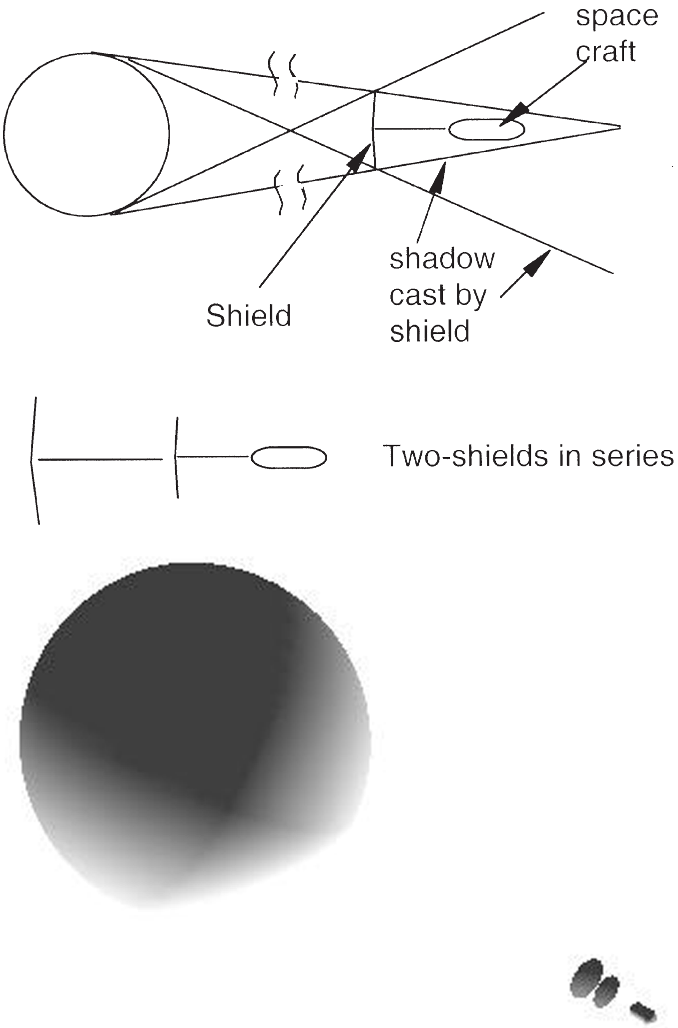

Fig. 5 Umbrella-like shadow shield from solar radiation (not to scale). The upper version shows a schematic of the idea and the lower one shows a perspective view.

$$T = \left[\frac{1}{2\mathrm{E}\sigma} \frac{P}{A} \left(\frac{r_o}{r} \right)^2 \right]^{1/4} \tag{5.2}$$

where the factor of 2 comes from radiation from both front and back. The emissivity is given from Kohl (p.165 and 274–p.158) [10] and plotted in fig. 6. The results show why graphite is the best of the candidate shield materials.

Radiation power using the data above and Eq. 5.1 is plotted in fig. 7. The arrows show the maximum practical temperature where loss of strength occurs; the curves end at the melting temperature.

The results of Eq. 5.2 are plotted in fig. 8.

There is a decided advantage in using graphite because of its high emissivity. The distance of closest approach for graphite would be 3 Sun radii and a little over 4 for tungsten.

The temperature of carbon fibers is limited by evaporation from the solid (sublimation). The theoretical evaporation rate is given by the expression

$$J = \frac{P}{17.14 \cdot (T/M)^{0.5}} \tag{5.3}$$

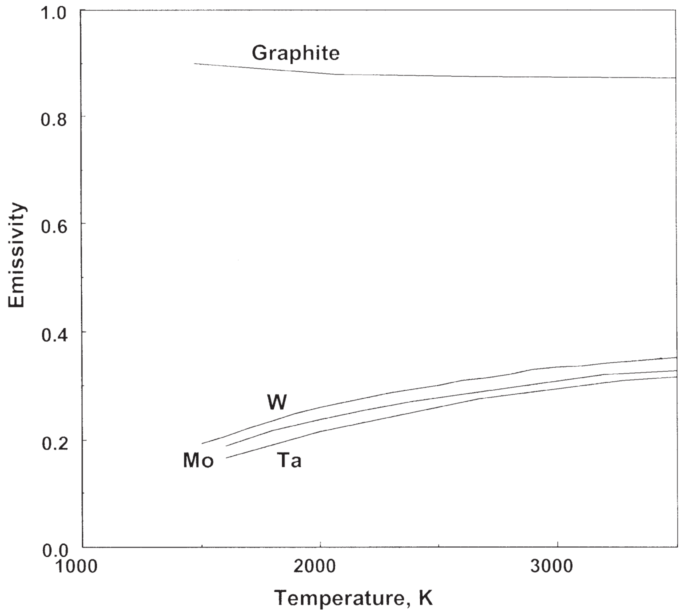

Fig. 6 Emissivity versus temperature.

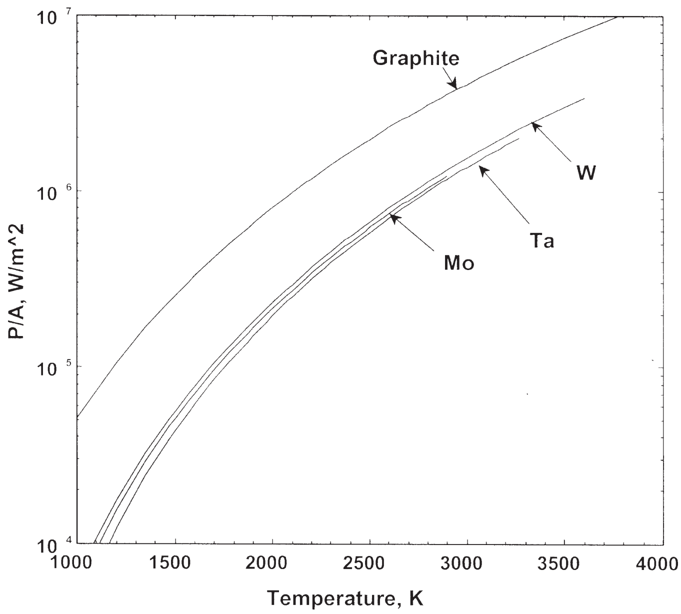

Fig. 7 Calculated radiated power.

$$P = e^{A - B/T} \tag{5.4}$$

where P is the vapor pressure in Pascals, T in Kelvin, M is mass per mole, which for carbon is 0.012 kg and J is in kg/m^{2}s. A reasonable fit to the data for vapor pressure from Kohl [10] for electrographite is A=37.5 and B=99,500. The sublimation rate from Ref [7,9,11,8] for the Solar Probe was taken as 0.0015 mg/m^{2}s at 2204 K and 0.0046 mg/m^{2}s at 2242 K. They note that sublimation rates were measures "…. about an order of magnitude lower than for graphite presumably due to surface energy effects…" of fibers.

The surface material loss rate is J/ρ in units of m/s. The density of graphite is about 1400 kg/m^3. We plot in fig. 9 the theoretical loss rate based on measured vapor pressure from graphite in units of micrometers/hr. Also shown are two measured loss rate data points and the theoretical curve adjusted to pass through the measured points (dashed curve).

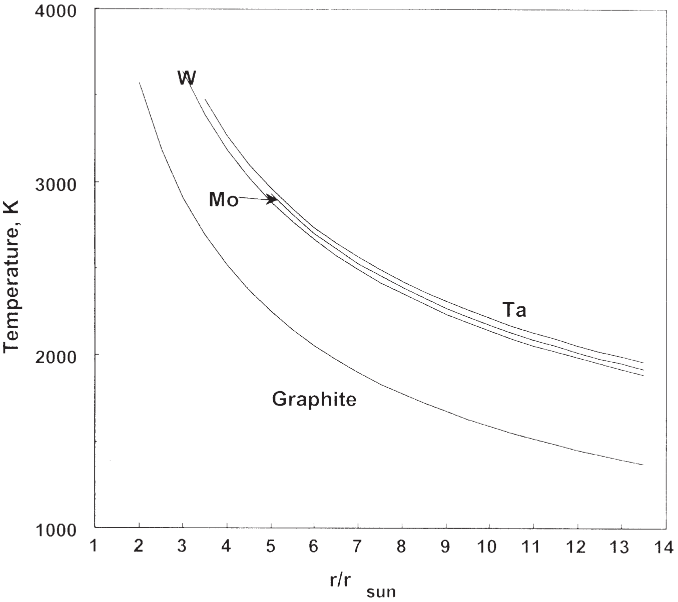

Fig. 8 Calculated temperature of the shield normal to the Sun versus distance from the Sun.

Based on use of fibers of tens of microns thickness and a few hours exposure time, the temperature is limited to about 2600 K.

A somewhat higher heat flux is tolerable if the shield is made of fabric, which is the most likely construction material anyway. A thin porous weave would expose single fibers to the incident flux proportional to its diameter, but it radiates from its perimeter, or 3.14 times its diameter, compared to a disk of twice its diameter for both sides. Thus a factor of up to π/2 can be put into Eq. 5.2. Adjacent fibers will shield each other somewhat, so this correction would have to be reduced. The same temperature will be reached at a reduced radius by a factor of $(\pi/2)^{0.5}$, which is 1.25. At the same radius, the temperature would be reduced by 12%. Use of fibers can make a flat surface of 2800 K become 2500 K. The shield could be tilted to spread the heat load over more area. For example, tilting 60° from that shown in fig. 6 would reduce P/A in Eq. 5.2 by a factor of 2 giving a temperature reduction of 19%.

A practical fabric might be woven from fibers of 10-µm diameter. If the weave were equivalent to a single layer side by side touching, then the mass would be:

$$\frac{\frac{\pi}{4} D^2 L \rho}{D L} = \frac{\pi}{4} D \rho = \frac{\pi}{4} 10\,\mu m\ \ 3400\,kg/m^3 = 0.0267\,kg/m^2 \tag{5.5}$$

A 100-m radius shield would then have a mass of 840 kg.

If a carbon fiber shield of 100 m radius were radiating at 2500 K appropriate to three Sun radii or about 3 MW/m^2 from each side, then the radiant flux on the spacecraft would be

$$\frac{P}{A} = \frac{\pi 100^2\, 3}{4 \pi x^2} = \frac{7500\,MW}{x^2} \tag{5.6}$$

where x is the distance from the shield to the spacecraft and x_o is a typical distance taken here to be 100 m.

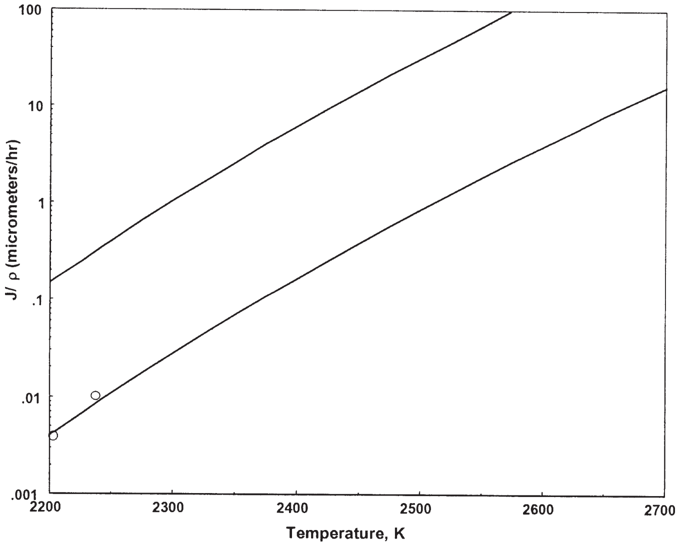

Fig. 9 Evaporative loss rate of the carbon fiber shield versus temperature.

$$\frac{P}{A} = 0.75\,MW/m^2 \left(\frac{x_o}{x}\right)^2 \tag{5.7}$$

For example, at 500 m, the flux rather than being 7 MW/m^2 from the Sun would be only 0.03 MW/m^2 or a factor of 100 reduction. A second shield could further reduce this heat load.

The Solar Probe [7] planned to fly by the Sun at 4 Sun radii. The present work is consistent with the Solar Probe work. A somewhat higher temperature owing to 3 Sun radii apogee is allowed because a 100 times higher evaporation rate is allowed during the flyby.

5.1 Ablative Shield

The flux at three solar radii (Case 4 of Table 1) peaks at 7 MW/m^2 and the total energy is 1.3×10^{11} J/m^2 during the flyby. This can be handled using a two-sided radiation shield made of carbon fibers. For higher heat fluxes, consider an ablative shield similar to that used to shield re-entry vehicles. The cohesive energy of carbon is about 100 MJ/kg to heat up to about 3000 K from near zero and then to evaporate. For a 100 m radius shield the amount of material evaporated during a flyby of the Sun would be:

$$\frac{1.3 \times 10^{11}\ J/m^2 \times \pi \times 100^2\ m^2}{100 \times 10^6\ J/kg} = 4.1 \times 10^7\ kg \tag{5.8}$$

The assumed spacecraft's mass is 10^8 kg (10,000 people x 10 tons per person) so this loss on each pass would be 40% of the ship mass. This is too much but could be cut down by designing a smaller radius ship. For example, a radius as small as 10 m might be a limiting case that would result in an ablated mass on each flyby to 0.4% of the ship's mass. For comparison our earlier radiative shield of mass 840 kg was only 10^{-6} of the ship mass. The ablative shield is possible but much more massive than the radiative shield and therefore considered impractical.

A more in depth analysis of ablation (evaporative) cooling can be seen from a power balance. The input power to the shield is the solar flux that is balanced by radiation cooling and evaporative cooling. See fig. 10 for the relative strength of each cooling process.

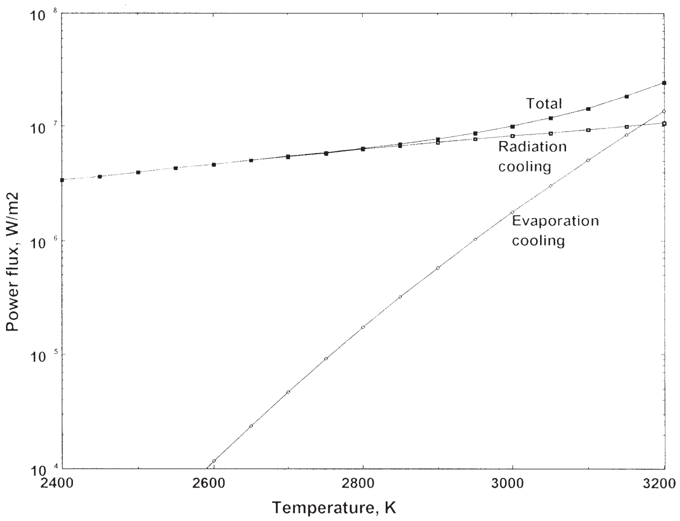

Fig. 10 Radiation and ablation cooling versus temperature.

$$H \cdot A = 2\mathrm{E}\sigma T^4 + 2\mathrm{E}_{cohesive} J \quad (5.9)$$

$$H = solar\ flux$$
$$A = absortivity \approx 1$$
$$\mathrm{E} = 0.9 = emissivity$$
$$\sigma = 5.67 \times 10^{-8}\ J \cdot m^{-2} \cdot s^{-1} \cdot K^{-4}$$
$$\mathrm{E}_{cohesive} = energy\ to\ heatup\ and\ evaporate \approx 100 MJ / kg$$
$$J = evaporate\ rate\ kg \cdot m^{-2} \cdot s^{-1}$$

(5.10)

The factor of two in Eq. 5.9 accounts for two-sided radiation and evaporation.

6. DISCUSSION OF SITCHIN'S HYPOTHESIS

This study started with speculation as to whether periodic visitations to the solar system as proposed by Sitchin [12], could be the result of cycling interstellar ships. In his book "The 12th Planet," Sitchin [12] proposed that the earth had been visited by extraterrestrial life on a cyclic basis every 3,600 years. The cycle was the result of the orbit of their planet. The present analysis was initiated partly to see how this might be possible. The first possibility examined was elliptic orbits like comets and planets. A period of this duration means the turnaround point is 230 AU (one AU is the distance from the earth to the Sun) from the Sun or six times further than the distance to Pluto. This is difficult to understand with our current understanding of solar system evolution. If we extend Sitchin's idea to include artificial habitats of the sort featured in the Arthur C. Clarke novel "Rendezvous with Rama" [13] and discussed as world ships by Martin [1] it would be more understandable. This second possibility examined was cyclic orbits to the near stars. However, as already discussed, the shortest cycle time found was 41,000 years with a realistic heat shield, and this only reduces the cycle time to 24,000 years if the Sun's surface is just skimmed with some futuristic heat shield during the flyby. So the present analysis is off by about an order of magnitude from explaining the 3,600-year period proposed by Sitchin.

7. DISCUSSION AND CONCLUSIONS

Cyclic trajectories were found that would return a large spaceship to the solar system periodically with minimal propulsion for course corrections only. The cycle time was 41,000 years for a course including three stars and 57,000 years for four stars. The cycle time was determined by fairly hard requirements: distance of closest approach, which was about 3 Sun radii on the flyby, set by radiation cooling and the need to have a cumulative deflect of 360°, which was on average 90° for each of 4 flybys of the nearest four stars in the circuit or 120° for three stars.

The maximum deflection required was the Sun encounter of 112°. The assumption of distance of closest approach to the star on flyby (about three Sun radii) is based on one design of shielding using radiation cooling at 7 MW/m^2. A better shield design would allow closer approach. The smallest distance of closest approach–skimming the surface of the Sun–would shorten the roundtrip to 24,000 years for three stars and 33,000 years for the four star case. This closest approach distance would result in a heat flux of 60 MW/m^2, making for a difficult shielding problem. However, owing to a shorter exposure time the energy to be absorbed is only twice that at 3 Sun radii.

The cycle times might be prohibitively long from the human viewpoint. However, space-faring civilizations evolving on planets circling widely-separated members of multiple-star systems or living in galactic regions of higher stellar density might well apply the techniques described in this paper.

8. ACKNOWLEDGEMENTS

We want to acknowledge Edward Cheng for his interest and suggesting this topic and Gerald Nordley for a number of useful suggestions and discussions. The many useful suggestions by the referees and editor of JBIS are appreciated. W. L Barr, a long time friend and colleague in the fusion research program at Lawrence Livermore National Laboratory contributed many calculations of orbit mechanics. He passed away April 24, 2004.

REFERENCES

1. A.R. Martin, "World ships—Concept, cause, cost, construction and colonization", *JBIS*, **37**, pp.243-253, 1984.
2. Buzz Aldrin, Dennis Byrnes, Ron Jones, and Hubert Davis, "Evolutionary space transportation plan for Mars cycling concepts", AIAA Space 2001 Conference and Exploration, Albuquerque, NM. Aug 28-30, 2001.
3. Victor G. Szebehely, "*Adventures in Celestial Mechanics*", U. of Texas Press, 1989.
4. E. F. Mallove and G. L. Matloff, "*The Starflight Handbook*", John Wiley & Sons, Inc., New York, 1989.
5. C. E. Singer, "Interstellar propulsion using a pellet stream for momentum transfer", *JBIS*, **33**, pp.107-115, 1980.
6. G.D. Nordley, 'Momentum Transfer Particle Homing Algorithm' presented at NASA/JPL/MSFC/AIAA Annual Tenth Advanced Space Propulsion Workshop, Huntsville, AL, April 5-8, 1999, also cited on p.98 and 100 of G. L. Matloff, "*Deep-Space Probes*", Springer Praxis Publishing, Chichester, UK, 2002.
7. G. Gloeckler, Chairman, Solar Probe Science Definition Team, "Solar Probe: First Mission to the Nearest Star—Report of the NASA Science Definition Team for the Solar Probe Mission" 1999.
8. J.E. Randolph, W. Imbriale, R.N. Miyake, E.A. Pierson, R.B. Dirling,

Jr., "The solar probe heatshield development", unpublished report about 1999.
9. J. Randolph, J. Ayon, R. Dirling, W. Imbriale, R. Miyake, D. Le Queau, G. Olalde, E. Pierson, S. Rawal, B. Rivoire, J. F. Robert, C. Royere, R. Taylor, P. Valentine, W. Vaughn, "The solar probe shield/antenna materials characterization", *Carbon*, **37,** pp.1731-1739, 1999.
10. W.H. Kohl, "Handbook of Materials and Techniques for Vacuum Devices", Reinhold Publishing Corporation, New York, 1967.
11. P.G. Valentine, P.W. Trester, R.O. Harrington, Jr., "Mass Loss Testing of Carbon-Carbon at High Temperatures," GA-C22515, 1997.
12. Z. Sitchin, "The 12th Planet", Avon Books, Harper Collins Publishers, New York, 1976.
13. A.C. Clarke, "Rendezvous with Rama", Bantam Books, New York, 1973.

(Received 13 September 2004; 10 January 2005; 18 March 2005; 8 July 2005)

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