ON THE APPLICATION OF SOLAR RADIATION MOMENTUM TRANSFER TO SPACE VEHICLE PROPULSION

by

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ERRATA

PAGE	FROM	TO
34 bottom line	φ	d φ
35 eq 22	$\sin \varphi$ / $\cos$	$\sin d\varphi$ / $\cos d\varphi$
38 bottom	P	ρ
39 nomenclature		add - - ϵ = emissivity σ = electrical conductivity
152 3rd line		add - - ref 50
155 2nd line	$2\pi r(rd\gamma)$	$(2\pi R)\ rd\gamma$
156 eq 120		add - - π on both sides
157 eq 127		add - - π
158 eq 134		add - - = .48 lb/foot length
159 eq 135		$T_b = B x_1 A_1 = B_0 \cos^2\alpha\ A_1 x_1$
162 3rd line from bottom	1 RPM	0.4 RPM
163 above eq 156	r dr	r + dr
164 eq 159 denominator		add - - π
168 eq 180	$\frac{1}{mr} = + \quad - e$	$= e + \lambda - e\lambda$

P. Villers
2/9/60

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ABSTRACT

This thesis is an evaluation of solar radiation propulsion, known as solar sailing, and its applicability to interplanetary propulsion. Consideration is also given to the use of solar sails for vehicle attitude stabilization. The thesis includes four parts.

Part I provides up-to-date information on pertinent aspects of the space environment. This part includes a survey of previous literature on the subject and a critical evaluation thereof. Possible solar sail missions are surveyed and new ones presented. A more systematic classification of solar sails, applicable terminology, and a suitable figure of merit are proposed. The new terminology "Solar Bounce" is offered to replace "Solar Pressure" because of significant errors resulting from existing implications of the latter term. All sail designs are found to fall into three classes, of which one is shown to be unstable.

Part II is a study of the "Self-Stabilized Stabilizer" which is offered as a new attitude stabilization design with interesting possibilities.

Part III is a preliminary design study on a "Centrifugal Sail" Mars Reconnaissance Probe. Included is a study of "Payload Shift", a new method of sail attitude control, employing inertial effects and solar forces to cause sail precession. Part III also includes results of experiments performed by the writer at M. I. T. and similar experiments at NASA on dynamic responses of the spinning sail.

Part IV compares the merits of a solar sail vehicle with alternative concepts and draws conclusions as to the present desirability of initiating research and development activities. Further related topics for future theses are also suggested. A possible Solar Sail development program is offered.

A bibliography and list of references, including 140 recent entries on space technology, is appended. It is organized by topic, and short summaries of each entry have been included.

Thesis Supervisors: Robert W. Mann
Associate Professor of Mechanical Engineering

Paul E. Sandorff
Associate Professor of Aeronautics and Astronautics

DEDICATED TO:

PROFESSOR GEORGE SUTTON

(now Chief Scientist of ARPA)
who suggested this investigation
and stimulated my curiosity
about Solar Sailing

ACKNOWLEDGMENTS

In a new field such as Solar Sailing a great deal of information has not yet been assembled and requires investigation in many different directions. It is with pleasure that I take this opportunity to acknowledge the assistance of some of the many people who have helped me in writing this thesis.

First I wish to acknowledge the help of my two advisors Professor Mann and Professor Sandorff, respectively of the department of Mechanical Engineering and of the Department of Aeronautics and Astronautics. They have spent many hours discussing, suggesting, pointing out weaknesses, and suggesting promising avenues to be investigated. Professors Brosens, Coons, Dahl, Dietz, Ezekiel, Den Hartog, and Paynter were especially helpful, as were Messrs. Carey, Baumann, and the staff of DACL Laboratory. At the MIT Instrumentation Laboratory, Messrs. Trageser, Dahlen, Bowditch, Magee, Scholten discussed various aspects and provided design approaches. Dr. T. C. Tsu of Westinghouse gave generously of his time and contributed valuable ideas. Dr. Cotter of the Los Alamos Scientific Laboratory and Messrs. Emmanuel Schnitzer and Jim Simmonds of NASA Langley Field gave me some of their data.

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NOMENCLATURE

A. U.	= astronomical units, (Earth-Sun distance = 1)
A	= area
A	= albedo of earth
a	= acceleration
A	= absorptivity
A_s	= A solar
A_e	= A Earth
A	= tangential acceleration
A_r	= radial acceleration
A_n	= acceleration in g_o units for $\alpha = 0$
B	= "Bounce" (as defined p 10) in psf, where $B = B_o \cos^2 \alpha$
B_o	= "Bounce" (as defined p 10) at normal incidence in psf
C	= speed of light
C_d	= aerodynamic drag coefficient
C_s	= solar constant
D	= distance
D	= diameter of earth
d	= moment arm
E	= energy
e	= emissivity
F_s	= solar radiation force on sail
F_g	= solar gravity force on vehicle
F'_g	= solar gravity force on sail alone
G	= sublimation rate in gm/cm2 sec

g_o = standard earth gravitational acceleration

g_s = solar gravitational acceleration

g = grams

H = angular momentum

h = altitude

K = $R_e/R_e + h$

L = lightness number of vehicle defined by Eq. 16
(see Eq.16a for conversion between L and acceleration a)

L = lightness number of sail alone defined by Eq. 17

M = molecular weight

M_a = mass of 1 atom

M_p = payload mass

M = mass

N = number protons/cm^3

P = vapor pressure in mm Hg

P_o = gas pressure in psf

R* = radial distance from sun in A. U. (dimensionless)

r = radius

$\bar{R}$ = reflectivity

R_e = radius of the earth $\approx$ 4000 miles

S = solar constant at 1 A. U.

S' = solar constant at distance specified

S_r = earth reradiation constant at surface of earth

S_e = earth constant including reflection and reradiation at surface of the earth

S'_e = earth constant including radiation plus reflection at a distance R + D from the center of the earth

Nomenclature (cont.) - 3

T	= temperature in 0^{o} K
t	= time
V	= velocity
W	= weight
	= density in lbs/cu ft
α	= angle of incidence, (measured between the incident light ray and a vector normal to the surface
ß	= angle between rod and normal to sail (Fig. 21)
ß	= angle of reflection
ß	= angle that normal to sail makes with respect to the ecliptic
γ	= wt/unit area in lbs/ft^2
γ	= angle between velocity vector and the local tangent to an orbit
ϵ	= wt/unit length (in lb/ft
ϵ	= efficiency
ϵ	= emissivity
ϵ_i	= inside
ϵ_o	= outside
σ	= electrical conductivity
μ	= central transfer angle, (solar)
ψ	= angle between instantaneous velocity and acceleration vectors
μ	= central gravitational constant where $\frac{\mu}{R^*}$ is the gravitational potential

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I. INTRODUCTION

Space vehicle propulsion systems have the unique property of operating in a high vacuum in the presence of extremely weak field forces. It is because of these conditions that propulsion systems frequently referred to as micro thrust propulsion are of real interest. In the last few years ion propulsion systems have attracted considerable study, and more recently, actual development effort. Problems of system weight, complexity, reliability and design are, of course, extremely severe and have prompted the continuing search for other propulsion methods.

As early as 1951 a propulsion system of utmost simplicity was suggested involving direct use of solar energy through momentum transfer to a lightweight reflecting surface. (Ref. 108). Such a system popularly known as " Solar Sailing" offers for reasonable payloads a vehicle accelerations of the order of 10^{-4}g, which is the value most frequently quoted for ion propelled systems.

Two unique properties of a " solar sailer" are: a mass ratio of unity and a thrust which, for a fixed geometry, varies uniquely as a function of orientation and radial distance from its power source, the sun.

At the present time the lack of extensive work in this field has fostered a widespread belief that Solar Sailing is an ingenious idea of limited practical interest. This point of view has been contributed to by the, in important respects, over-stretched analogy with the sailing ship on earth. This coupled with the very small solar radiation forces available has tended to discourage detailed investigation. It is the purpose of this paper to critically examine the problems raised by this novel propulsion approach with the avowed objective of determining to what extent the concept of solar radiation propulsion deserves practical development.

Part I of the paper is an attempt to organize available information from a wide variety of sources into a coherent framework covering: pertinent environmental data, propulsion and control design approaches, suitable missions and their limitations, and applicable trajectories. In Part I will also be found an explanation of why the writer suggests the use of the term "Solar Bounce" to replace "solar pressure," and an account of how the aerodynamic analogy suggested by the latter term has resulted in several incorrect conclusions in the existing literature.

Part II develops a new solar sail design to be known as the "self-stabilized stabilizer" for attitude stabilization and considers its merits.

Part III introduces a design deemed suitable for a Mars reconnaissance probe, and presents analytical and experimental data relevant to the proposed vehicle. Some of this data was developed by the writer in experiments conducted at MIT, and the remainder comes from several sources as is indicated in respective footnotes. The vehicle design presented draws on existing contributions to the field and goes on to develop several new design features proposed by the writer.

Part IV will draw on Parts I, II, and III to justify conclusions as to the present and future competitive potential of " Solar Sails" for space use.

PART I - Solar Sailing, A Survey of Applications, Techniques, and Feasibility - Pertinent Effects of Environment

1.0 Space Environment

Just a few years ago the term "perfect vacuum" was considered an adequate description of the space environment. Today intensive study, existing astronomical data backed up by rocket, satellite, and space probe data has provided some information and at the same time raised many new and unanswered questions. In Table I is found a list of important characteristics of the space environment taken from a paper by Dr. John C. Simons, Jr. of the National Research Corporation (Ref. 38).

This section will present available information on many of the above characteristics and also indicate areas of ignorance important to space vehicles, more specifically solar sails. First, however, we will comment on the Solar Flux and resulting solar momentum transfer.

1.1 Solar Flux as a Propulsive Source

The sun is a radiating source of energy equivalent to a black body at about $6000^{\circ}K$ and, following the inverse square attenuation law, it provides power at a distance of 1 A. U. of 1.36 to 1.4 erg/cm^2 sec = 92-95 ft lb/ft^2 sec. This power is distributed with a peak energy transmission in the visible range. Table II reproduced from an article by R. A. DiTaranto and J. J. Lamb shows energy as a function of wavelength. (Ref. 32). In astronomy, solar momentum transfer is considered responsible for several important effects including the streaming of comet tails and expulsion from

TABLE I

CHARACTERISTICS OF THE SPACE ENVIRONMENT

NATURAL	Low pressure and density
	Chemical composition Dissociated molecules Ions
	Thermal radiation, influencing vehicle temperature Infrared solar radiation Earth's albedo Infinite radiation sink ($0^{o}K$)
	Other solar radiation Visible Ultraviolet X ray
	Cosmic radiation Electromagnetic (gammas, X rays) Primary particles (protons, atomic nuclei) Secondary particles (electrons, positrons, mesons, Neutrons) Van Allen belt radiation (protons or electrons)
	Meteoric particles
	Force fields Electromagnetic Gravitational
OF HUMAN ORIGIN	Propulsion products
	Vehicle outgassing
	Acceleration
	Vibration
	Space debris
	Hostile action

TABLE II

DISTRIBUTION OF SOLAR ENERGY AS A FUNCTION OF WAVELENGTH

Region	Wavelength (in Angstroms)	Energy (in %)
Far Ultraviolet	1 - 2000	0.2
Near Ultraviolet	2000 - 3800	7.5
Visible	3800 - 7000	41
Infrared	7000 - 10,000	22
Infrared	10,000 - 20,000	23
Infrared	20,000 - 100,000	6

(From Ref. by R. A. DiTaranto & J. J. Lamb) the solar system of micrometeorites of certain weight/area ratio (Ref. 42). The equations relating the Energy E to the rate of momentum transfer/unit area are as follows.[1]

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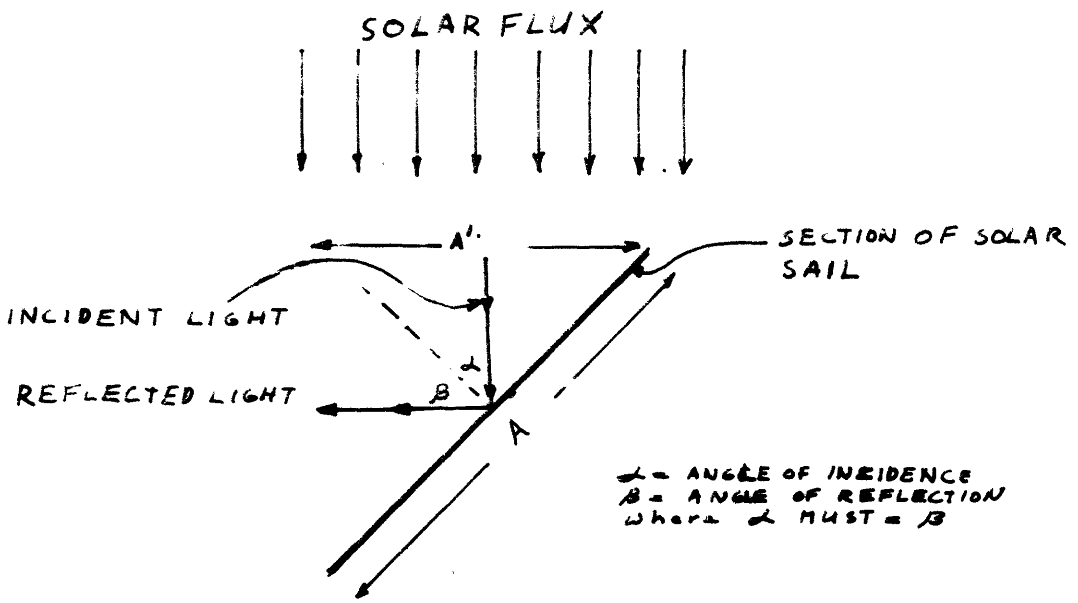
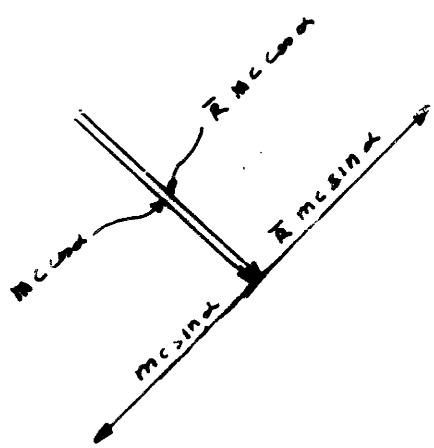
[1] The following derivation substantially follows that of Krafft A. Ehricke, Ref. 128, p. 62.

FIGURE 1

SOLAR FLUX ON SAIL AND RESULTING SOLAR BOUNCE



VECTOR RESOLUTION OF MOMENTUM TRANSFER TO SAIL



$$S = \frac{S}{R^{*2}}$$ where S is the solar constant at 1 A. U. (1)

where R* is the distance in A. U.

$$E = mc^2$$ where m is the relativistic equivalent mass. (2)

$$m = \frac{E}{c^2} = \frac{S' A' t'}{c^2} \qquad \frac{\text{Force} \cdot \text{time}^2}{\text{length}} \tag{3}$$

where m is the relativistic equivalent mass
and where A' is the projected area at normal
incidence shown in Fig. 1 such that with angle
of incidence α $A' = A \cos \alpha$ (4)
the momentum per unit area is:

$$mc = \frac{s' \, t \cos \alpha}{c} \qquad \frac{\text{Force time}}{\text{length}^2} \tag{5}$$

where mc is the momentum of the incident beam/unit area

From Newton's second law, the force/unit area for reflectivity is:

$$\partial/\partial t \, (mc) = \frac{S'}{c} \cos \alpha \left[\frac{\text{Force}}{\text{Length}} \right] \tag{6}$$

For a body of reflectivity, $\overline{R}$, the force unit area is obtained as the time derivative of the vector sum of the incident and reflected momentum transfer. As seen in Figure 1:

$$F/A = \quad \partial/\partial t \, (1 + \overline{R}) \, \vec{mc} \cos \alpha + (1 - \overline{R}) \, \vec{mc} \sin \alpha \tag{7}$$

For $\overline{R} \approx 1$,

$$\frac{\partial}{\partial t} \sum \vec{mc} = \frac{\partial}{\partial t}(2 \, mc \cos \alpha \tag{8}$$

$$= \frac{2 \, S'}{c} \cos^2 \alpha \tag{9}$$

$$\text{define } B_o \equiv \frac{2\,S'}{c} \tag{10}$$

$$\text{define } B \equiv \frac{2\,S'}{c}\cos^2 \alpha \tag{11}$$

$$\text{Therefore, } B = B_o \cos^2 \alpha \tag{12}$$

Where B_o is the rate of momentum transfer/unit area for an area normal to the source, and is equivalent to force/unit area.

Where B is the rate of momentum transfer/unit area whose normal is inclined at an angle α with respect to the source.

The use of the term "radiation pressure" which has been previously used has been avoided. This is to avoid the isotropic connotations of pressure which have resulted in misunderstandings in the existing literature. For instance, the "parachute" type of solar sail discussed in Section 2.1, and previously referred to in the literature appears to be inherently unstable in the absence of stiffeners or force fields. In the case of a true gas pressure or aerodynamic phenomenon it would, of course, be stable, and the apparent analogy between the two cases is undoubtedly responsible for the opinion that such an unstiffened sail configuration might be stable. Similarly, we find that "gas pressure" P_o acting on a sphere of radius r gives a force $P_o \pi r^2$ whereas, contrary to previously reported values, for a solar radiation of "pressure" P_o, the force on the sphere is only $P_o \frac{\pi r^2}{2}$. (See Appendix II.)

For the above reasons the writer has used the terminology "Bounce" to stand for what strictly speaking is "rate of momentum transfer/unit area." The symbols B and B_o will be used respectively for " Bounce" at an arbitrary incidence and at normal incidence. This is advocated as helpful in disassociating the vector radiation process which can be represented by the classical process of momentum transfer through elastic collisions from the stochastic and scalar process of kinetic theory of gases commonly associated with the term Pressure. Both Pressure and Bounce can be measured in lbs per square foot (PSF) but Pressure being a scalar has the same magnitude for all orientations. Bounce being a vector has a magnitude which is a function of orientation. The confusion between the above vector and scalar quantities is in many respects similar to the one in nuclear engineering between the dimensionally identical terms, Intensity (I) and Flux (ϕ) which are only identical for the special case of the collimated beam.

The values of Bounce at normal incidence are as follows:

Distance:	Mercury	Venus	Earth	Mars
B_o (lb/ft^2):	10^{-6}	3.55×10^{-7}	1.95×10^{-7}	9.10^{-8}

using in each case the mean distance of the planet's orbit.[1]

We recall that for angle of incidence α, the Bounce will be $B = B_o \cos^2 \alpha$.

A further important correction is that for other radiation sources. Radiation from other sources produces negligible Bounce as compared to sunlight with the sole exception of a satellite orbit where the combined effects of a planet's albedo and infrared reradiation produce a maximum $B_{o_{terrestial}}$. As

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[1] Ref. 128, p. 62

derived in Appendix I, this quantity varies such that at an altitude of 1000 miles, for example:

$$0.1\, B_{0_{solar}} \leq B_{o_{terrestrial}} \leq 0.33\, B_{o_{solar}} \tag{111}$$

which is of course a major correction factor for any satellite Bounce calculations. Clearly for many orientations since $B = B_o \cos^2 \alpha$ (Eq. 12).

$$B_{terrestrial} \geq B_{solar}$$

We shall therefore keep this in mind in discussing solar sailing near the earth.

1.2 Thermal Balance

As is well-known, thermal balance and heat rejection are among the more difficult problems in space operation. Although evaporation, ablation, and other material consuming heat dissipation techniques are of interest for a few space and re-entry operations, in virtually all cases radiation cooling is the dominant mechanism.

The applicable variables are then: geometry $\left(\frac{\text{emitting area}}{\text{irradiated area}}\right)$, orbital characteristics, orientation, and $\frac{\text{absorptivity}}{\text{emissivity}}$ or $\frac{A}{e}$ ratio, internal heat generation and, if transients are important, thermal capacity and conductivity.

For satellite operations an excellent treatment taking into account earth light will be found in Ref. 124.

For space vehicles a series of calculations by R. A. Cornog of the Space Technology Labs. (Ref. 122) shows how equilibrium temperature for plates and sphere vary as a function of orientation, $\frac{A}{e}$ ratio, distance and internal heat sources. Table III reproduced from the above reference lists $\frac{A}{e}$ for some materials of interest.

Figure 2 through 5 also reproduced from Cornog's work shows equilibrium temperature as function of the pertinent variables.

Existing data on satellite vehicles indicates that through judicious choice of coatings, temperature has been successfully controlled between the predicted limits of 0 - 40^{o}C. (See Ref. 29 for information on coatings and results obtained.) Since $\frac{A}{e}$ is a function of temperature choice of materials with desirable $\frac{\partial A}{\partial t}$ and $\frac{\partial e}{\partial t}$, coupled with T^4 radiation law help provide adequate temperature stability. For space missions involving substantial changes in R^* (distance to sun) active control may be required. Such a system using thermostatic control to change the color of the surface is described in Ref. 125. A concrete example of a sophisticated technique involving control by regulation of waste heat from solar cells is described in Ref. 135. Thermal equilibrium for a proposed solarsail is treated in Section 2.5.

TABLE III[1]

OPTICAL PROPERTIES OF VARIOUS MATERIALS*

Material	*F	Absorption Number A	Emis-sivity E	E Ratio A/E
Silver	100	0.04	0.02	2.0
Aluminum, polished	100	0.10	0.05	2.0
	1000	- -	0.06	
Aluminum, 2024, buffed and polished[9]	100	0.34-0.37	0.03	12.0
Stainless steel, black	100	- - -	0.90	- -
	1000	- - -	0.90	- -
Stainless steel, polished	100	0.40	0.05	8.0
Fused quartz, bricks	100	0.1-0.4	0.90	0.2
Hard rubber, asbestos	1000	- - -	0.90	- -
Lamp black	100	0.95	0.95	1.0
	1000	- - -	0.95	- -
SiO on polished metal[7]	100	0.1	0.90	0.1
MgO[9]	100	0.15	0.97	0.15
Titanium, 6Al-4V[9]	100	0.8	0.18	4.4

* Buettner stated that the ratio of absorptivity of solar radiation to the low temperature emissivity may vary from ten for ideally polished metals such as aluminum and nickel, to one-tenth for ideal white.

1 Reproduced from R. A. Cornog, Ref. 122.

FIGURES 2 & 3

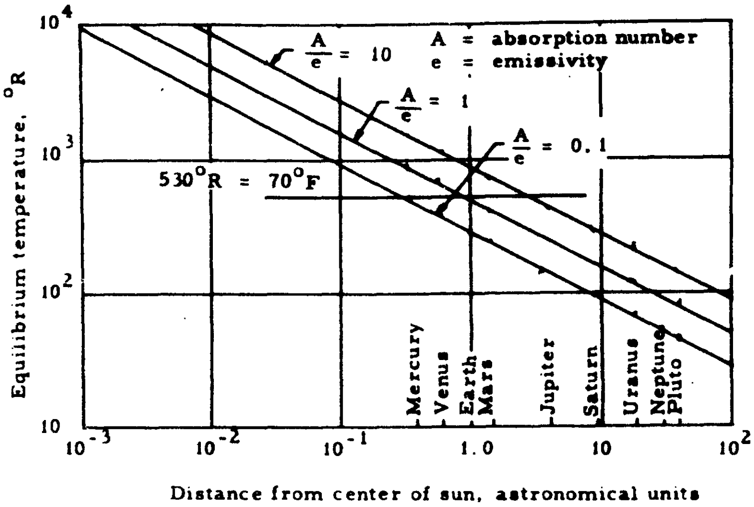

FIG. 2. Equilibrium temperature of an inert sphere

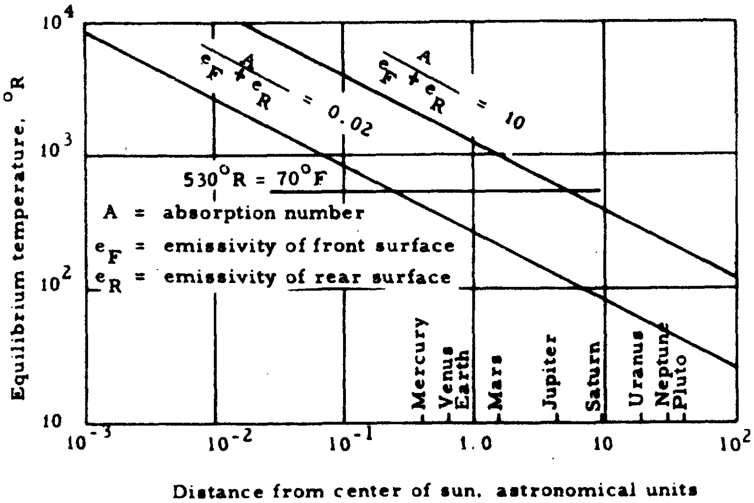

FIG. 3. Equilibrium temperature of a thin plate normal to the sun.

(From Fig 2 & 3 ref 122 by R.A. Cornog)

FIGURES 4 & 5

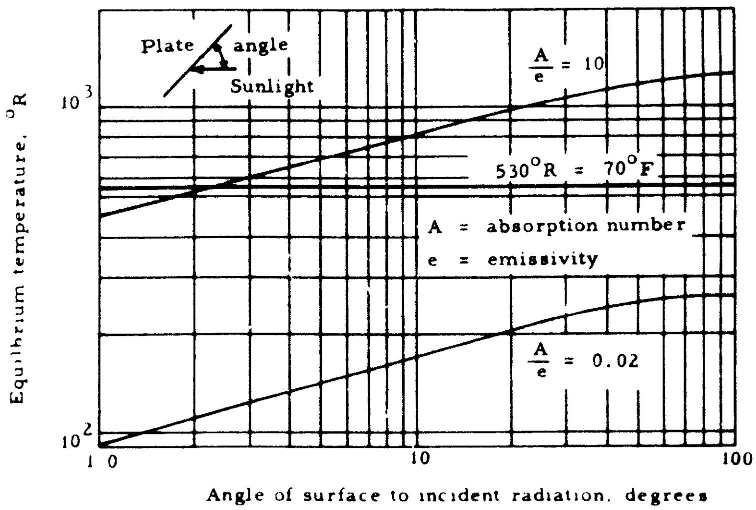

FIG. 4. Effect of attitude on equilibrium temperature of a thin plate located at one astronomical unit from the sun.

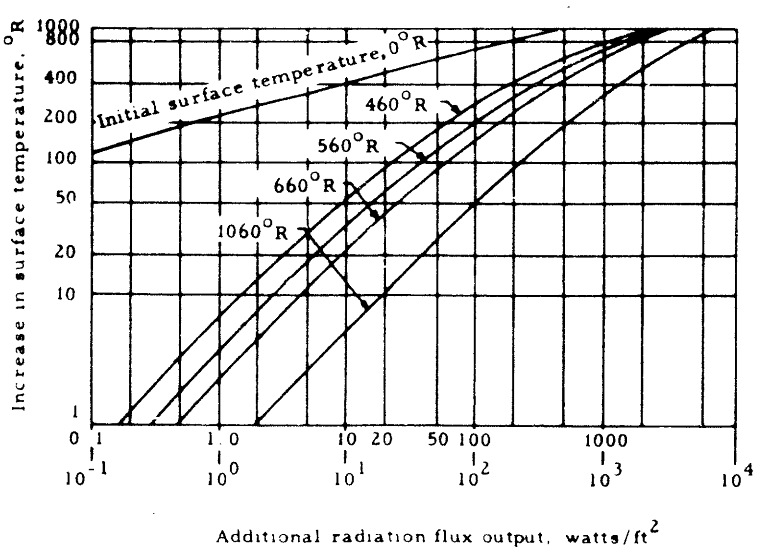

FIG. 5. Effect of added heat input on equilibrium temperature.

(From Fig 4&5 ref 122 by R. A. Cornog)

1.3 Radiation Effects

Among solar radiations listed in Table I two types in particular can be highly damaging to a number of materials. One such are soft X rays, the other are ultraviolet rays. It is to be recalled that most of the ultraviolet rays of the sun do not reach the earth and therefore represent an area of limited knowledge. According to recent experiments both the extreme ultraviolet and soft X rays regions of the solar spectrum can have detrimental effects on some materials. (See Ref. 29, p. 6 and Ref. 30.) This is especially true of organic materials with their weaker covalent and molecular (Van der Wahl) bonds. These effects as reported in Ref. 29, p. 6, include crosslinking of polymer chains and increased molecular weight, scission of the molecular chains and decrease in molecular weight, dehydrogenation and fragmentation of volatile products (methane, ethane, CO, CO_2), as well as double bond formation. The above referenced paper by Dr. Clauss of Lockheed also describes the practical consequences of the above effects. They include: raising or lowering tensile strength, embrittlement of polymers, decreased solubility, loss of adhesion, discoloration, and increased electrical conductivity. Current theories as to the mechanisms of degradation including the theory of Random degradation, the Weak link theory, and the reverse polymerization theory, are covered in Ref. 35, p. 2. A Substantial bibliography on the subject is included in the paper.
Ref. 39 brings out the interesting fact that degradation of polymers due to ultraviolet action are in some cases less severe in vacuum.

We will discuss these problems in the specific context of a solar sail in Section 1.6.

Of special interest to our discussion is the phenomenon known as sputtering. Quantitative values have been presented by Whipple (Ref. 42). They are, however, subject to large factors of uncertainty. Experiments on the subject are being conducted, but offer great difficulty. (See Ref. 40 for existing experimental data and experimental methods). "Proton sputtering" is the ejection of atoms from a surface by bombardment of high energy protons. Quite obviously the same effect can be caused by electrons or other ions both from solar and cosmic sources, but according to Whipple, outside of the immediate vicinity of the earth, proton sputtering from the sun is by far the dominant mechanism. A rate of 2×10^{-13} gm/cm^2/sec is given for the sputtering effect on aluminum which corresponds to approximately 1×10^{-6} in./yr[1]. Whipple's data are based on the equation:

$$\frac{d\,\text{mass}}{dt} = \frac{M_a \cdot \epsilon N V}{4} \tag{14}$$

where: M_a = mass of atom (for Al = 4.5×10^{-23})

ϵ = efficiency factor (atoms removed/proton)

N = protons/cm^3 (uses 600)

V = average velocity (uses 300 km/sec)

As for the efficiency of sputtering we note the interesting experimental result reported by Clauss that surface oxides reduce sputtering on metallic surfaces.[2]

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[1] Ref. 42, p. 120

[2] Ref. 29, p. 9

Whipple's above calculations are based on average values. In recent years interest in solar flares has been stimulated by data showing very large variations coinciding with increased solar activity. The incident of February 23, 1956 for instance increased the cosmic ray flux by three orders of magnitude in ten minutes and abnormalities lasted two days. (Ref. 135, Vol. III, p. 564).

The effect on materials of cosmic rays and any secondary radiations generated by them is another important consideration. Existing information consists of extrapolation from data on nuclear irradiation, and is not yet on a solid footing. It is likely that embrittlement of some metals will result, as is the case for nuclear bombardment. Sandorff discusses this problem briefly and proposes the term "radiation fatigue," postulating time and stress dependent effect on metals similarly to ordinary metal fatigue, (Ref. 119).

1.4 Cosmic Dust, Micrometeorites

Existing data on matter in space is generally acknowledged to be subject to an order-of-magnitude error. The most widely used values for meteoritic and micrometeoritic matter is based on a study by Whipple. His values are reproduced in Table IV. (Ref. 42). The values he indicates should be taken as "fairly high limits, representing high rates of puncture probabilities, because the penetration law ....almost certainly overestimates the powers of small particles to make holes in sheets of material." He further points out, "it is possible that a considerable error is made for the numbers of particles in the ranges of interest."

TABLE IV

METEOROID FREQUENCY AND VELOCITY DISTRIBUTION, CALCULATED PENETRATION

(from ref. 42, table 1)
(by F.L. Whipple)

(1) Meteor visual magnitude	(2) Mass (g)	(3) Radius (μ)	(4) Ass. vel. (KM/Sec)	(5) K.E. (ergs)	(6) Pen. in Al (cm)	(7) No. Striking earth per day	(8) No. striking 3 m sphere per day
0	25.0	49,200	28	1.0×10^{14}	21.3	--	--
1	9.95	36,200	28	3.98×10^{13}	15.7	--	--
2	3.96	26,600	28	1.58×10^{13}	11.5	--	--
3	1.58	19,600	28	6.31×10^{12}	8.48	--	--
4	0.628	14,400	28	2.51×10^{12}	6.24	--	--
5	0.250	10,600	28	1.00×10^{12}	4.59	2×10^{8}	2.22×10^{-5}
6	9.95×10^{-2}	7800	28	3.98×10^{11}	3.38	5.84×10^{8}	6.48×10^{-5}
7	3.96×10^{-2}	5740	28	1.58×10^{11}	2.48	1.47×10^{9}	1.63×10^{-4}
8	1.58×10^{-3}	4220	27	5.87×10^{10}	1.79	3.69×10^{9}	4.09×10^{-4}
9	6.28×10^{-3}	3110	26	2.17×10^{10}	1.28	9.26×10^{9}	1.03×10^{-3}
10	2.50×10^{-3}	2290	25	7.97×10^{9}	0.917	2.33×10^{10}	2.58×10^{-3}
11	9.95×10^{-4}	1680	24	2.93×10^{9}	0.656	5.84×10^{10}	6.48×10^{-3}
12	3.96×10^{-4}	1240	23	1.07×10^{9}	0.469	1.47×10^{11}	1.63×10^{-2}
13	1.58×10^{-4}	910	22	3.89×10^{8}	0.335	3.69×10^{11}	4.09×10^{-2}
14	6.28×10^{-5}	669	21	1.41×10^{8}	0.238	9.26×10^{11}	1.03×10^{-1}
15	2.50×10^{-3}	492	20	5.10×10^{7}	0.170	2.33×10^{12}	2.58×10^{-1}
16	9.95×10^{-6}	362	19	1.83×10^{7}	0.121	5.84×10^{12}	6.48×10^{-1}
17	3.96×10^{-6}	266	18	6.55×10^{6}	0.0859	1.47×10^{13}	1.63
18	1.58×10^{-6}	196	17	2.33×10^{6}	0.0608	3.69×10^{13}	4.09
19	6.28×10^{-7}	144	16	8.20×10^{5}	0.0430	9.26×10^{13}	1.03×10
20	2.50×10^{-7}	106	15	2.87×10^{5}	0.303	2.33×10^{14}	2.58×10
21	9.95×10^{-8}	78.0	15	1.14×10^{5}	0.223	5.84×10^{14}	6.48×10
22	3.96×10^{-8}	57.4	15	4.55×10^{4}	0.0164	1.47×10^{15}	1.63×10^{2}
23	1.58×10^{-8}	39.8*	15	1.81×10^{4}	0.0121	3.69×10^{15}	4.09×10^{2}
24	6.28×10^{-9}	25.1*	15	7.21×10^{3}	0.00884	9.26×10^{15}	1.03×10^{3}
25	2.50×10^{-9}	15.8*	15	2.87×10^{3}	0.00653	2.33×10^{16}	2.58×10^{3}
26	9.95×10^{-10}	10.0*	15	1.14×10^{3}	0.00480	5.84×10^{16}	6.48×10^{3}
27	3.96×10^{-10}	6.30*	15	4.55×10^{2}	0.00353	1.47×10^{17}	1.63×10^{4}
28	1.58×10^{-10}	3.98*	15	1.81×10^{2}	0.00260	3.96×10^{17}	4.09×10^{4}
29	6.28×10^{-11}	2.51*	15	7.21×10	0.00191	9.26×10^{17}	1.03×10^{5}
30	2.50×10^{-11}	1.58*	15	2.87×10	0.00141	2.33×10^{18}	2.58×10^{5}
31	9.95×10^{-12}	1.00	15	1.14×10	0.00103	5.84×10^{18}	6.48×10^{5}

*Maximum radius permitted by solar light pressure.

Whipple's quantitative data is based on extrapolation from the Harvard photographic meteor program, checked against radio astronomical data and deep sea meteoritic accression studies. It is interesting to note that presently available information from Explorer I satellite measurements is found to be "not inconsistent with" Whipple's figures.[1]

An area of important concern is the effect of micrometeorites on surface finish and in turn both on optical and thermal properties. Concern about effect on the absorption/emission ratio has led to sandblasting several of our satellites before takeoff with the object of insuring that the micrometeoritic effect on surface finish would no longer produce any further important changes. In the case of optical surfaces Whipple concludes, "the erosion cannot become important optically over a period less than a year."[2] For long lived vehicles such as Mars probes, the problem is a serious one and has resulted in the proposal to turn the lens of optical trackers towards the vehicle when not in use and to cap photographic lenses until the time a photograph is made.[3]

As for the penetration of micrometeorites given size and velocity, no generally accepted theory has yet evolved. Diamond in Ref. 123, p. 12 gives some equations on the cratering process based on the theory that at sufficiently high velocities the process is analogous to deceleration in a fluid media. He also discusses

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[1] Ref. 33, p. 44

[2] Ibid, p. 121

[3] Ref. 135, Vol II, Section 6-16, and Ref. 135, Vol. I, p. 37.

other penetration modes. Ref. 41 reports experimental results at NASA on 11,000 fps impacts. The subject is still widely debated since the relative impact velocities involved in the 10 - 80 km/sec range have not been successfully simulated.

The distribution of meteorites, micrometeorites, other particles and radiation is very probably not uniform in space, but little is known about this important subject. From observation of zodiacal light and radar results it is believed meteor orbits may follow the planetary pattern in being concentrated in a plane roughly that of the earth's ecliptic. (Ref. 42, p. 121. This raises the interesting possibility of reduced penetration probability for trajectories sharply inclined to the ecliptic, however, no experimental confirmation exists as of now. A high concentration of radiation, is trapped around the earth in the so-called Van Allen belts and this may turn out to be true with other planets. As for micrometeoritic surface erosion, Whipple feels that it may well follow some inverse law of distance decreasing with distance from the sun.[1]

1.5 Vacuum Effects

The gas pressure in space at some distance from the earth is estimated to have density equivalent pressures of 10^{-12} to 10^{-16} mm Hg.[2] Vacuum effects include increased friction, outgassing (sublimation), and of course modes of heat transfer. Also "surface effects" - crack propagation, emissivity characteristics, surface oxides, chemical reactivity, alloying activity.

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[1] Ibid, p. 121

[2] Ref. 38, p. 6

The importance of oxygen and other adsorbed layers on frictional characteristics are well know. Most common lubricants will either not lubricate in a space environment or volatilize (many oils), and with the absence of normal oxides and other surface films cold welding may result. According to Ref. 31, M_oS_2 and WS_2 seem to perform satisfactorily. Ref. 31 cited above includes a discussion of current theories on the mechanisms of friction and data on experiments in progress. Also of interest are Teflon, Nylon, Sapphire, and Pyroceram, indicated as good bearing materials in Ref. 29. An experimental program on frictional properties in a vacuum is in progress at Litton Industries in California on a number of materials and some results are already available (Ref. 34).

The second problem, namely outgassing, is just beginning to be fully appreciated. The expression relating loss in gm/cm^2 sec of material to vapor pressure is:

$$G = \sqrt{\frac{M}{T} \quad \frac{P}{17.4}} \tag{15}$$

where G is loss per unit exposed surface area in gm/cm^2 sec.

M = molecular Wt

T = temperature in 0^o K

P = is vapor pressure at T in mm Hg

The above equation quoted from Ref. 39, p. 21, is based on kinetic theory of gases. It assumes that no liberated atoms return to the material. Experimental work conducted at NRC is reported in the above reference, and provides the vapor pressure of many plastics and some metals. Figure 6 (based on a figure in the above reference) presents a plot of vapor pressure vs. temperature for a large number of metals and a few plastics. A further table of values is presented in Ref. 35 and provides vapor pressure for organic coatings as a function of temperature. The practical significance of the findings in the above three references is that thin coatings of many plastics and some metals are subject to excessive sublimation at even moderate temperatures, in some cases, ambient temperature. However, one should note that equation 15 may not be valid for complex polymers in that plasticizers may sublimate more easily than the parent plastic thereby giving deceptive high value to vapor pressure in short and medium length experiments as reported in the above experiments This is apparently the case with the polyester film known as Mylar which is discussed further in the next section.

1.6 Effects of Space Environment on Solar Sailing

The preceding five sections have developed some of the important characteristics of the space environment. We wish now to relate them to a solar sail which we shall assume is a thin plastic film such as Mylar with an aluminum or silver coating.

1.6.1 Outgassing

The average molecular weight of Mylar polyester films is approximately 20,000.[1] Using this value in Equation (15), using

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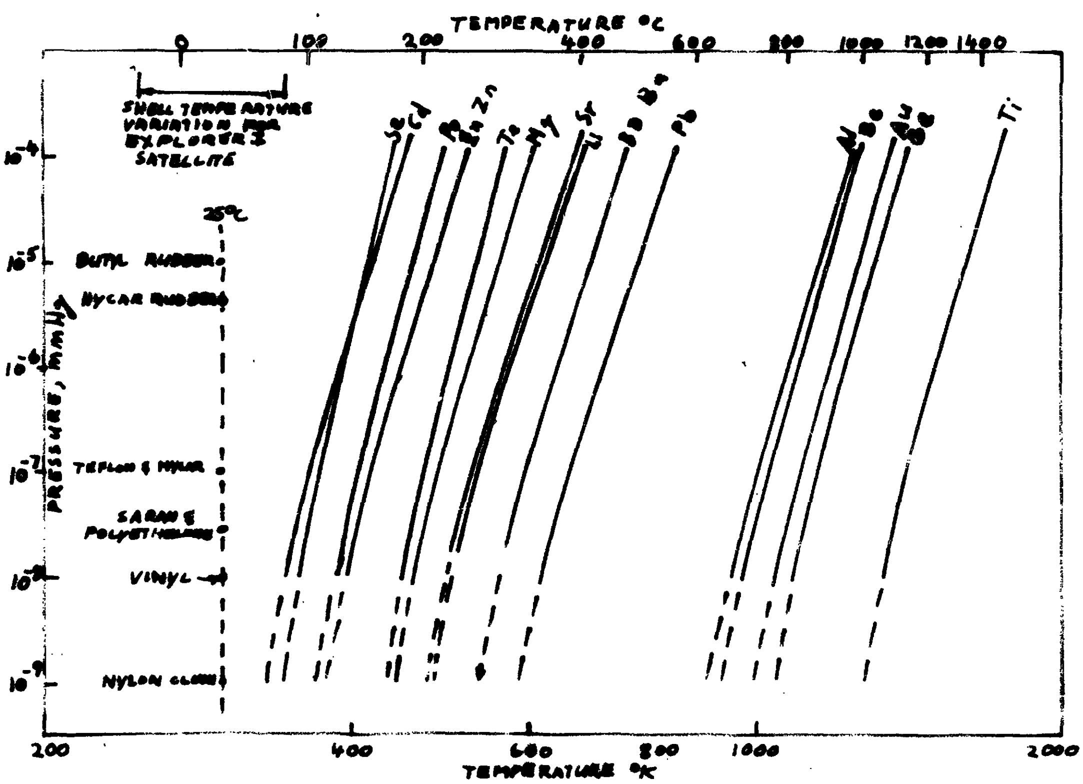
[1] Private communication, Mr. Emmanual Schnitzer, NASA, Langley Field.

FIGURE 6



VAPOR PRESSURE OF VARIOUS SOLIDS AS A FUNCTION OF TEMP.

(FROM REF 29 FIG 3 by F. J. CLAUSS)

$300^{\circ}K$ for temperature, with a pressure P of 10^{-7} mm Hg from Figure 6 we find that:

$$G = \frac{10^{-7}}{17.14} \sqrt{\frac{20,000}{300}} \approx 5 \times 10^{-8} \ gm/cm^2 \ sec \qquad (16)$$

or $\quad 2 \times 10^{-8}$ in. $/cm^2$ sec for 1 surface exposed.

Therefore, a thin film of 10^{-4} inches would volatilize in 5,000 seconds?

Experimental data on a polyester coating, namely "Paraplex P-43" of the Rohm and Haas Company with 2% "Luperco ATC" added (made by the Novadel Agene Corporation) is given in Ref. 35, Table 2. Results at ambient temperatures, 4×10^{-5} mm Hg pressure and 24 hours exposure are a weight loss of only 2.5% (19.7% for a run at $300^{\circ}F$. Obviously then the assumptions using Eq. (15) must be wrong. The probable answer was provided the writer by Mr. Emmanual Schnitzer[1] of NASA who explained that Mylar molecules are not homogenous and that a small percentage of the component molecules are of low molecular weight and are weakly bonded. Their volatilization provides the relatively high vapor pressures found in vapor pressure measurements, and very probably longer runs would find decreasing pressure. In the case of many plastics with appreciable plasticers, fillers and unreacted material this effect may be even more pronounced. It therefore seems that, at least for many organic materials , Formula (15)

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[1] Private communication, Mr. Emmanual Schnitzer, NASA, Langley Field.

must be rejected. In the specific case of Mylar (a promising material for a solar sail) according to the bast available information at NASA an approximately 1% weight loss is anticipated with very low outgassing rates afterwards.[1] NASA's forthcoming Mylar passive communication satellite should hopefully corroborate the above estimates.

1.6.2 Space Drag Effects

As will be shown later, Solar Bounce may provide up to approximately 10^{-4} g acceleration. We are interested in comparing this to probable space drag due to the 10^{-12} or higher dynamic pressure in space. For order of magnitude values we shall use the results of Krafft Ehricke's work based on a $C_D A/W$ ratio = 500 for a 10^{-4} inch thick Mylar sphere and gas weight small as compared to the weight of the Mylar. With a relative velocity of 100,000 fps, he finds a drag produced deceleration at 50,000 miles from the earth of 5.10^{-12} g. At 300 miles this figure is 6.10^{-4} g, at 1,600 miles deceleration it is 3.4×10^{-8} g.[1] The 300 mile value is in good agreement with Ref. 17, Fig. 2 which shows altitude vs. dynamic pressures and solar bounce.

1.6.3 Erosion, Puncture and Radiation Damage to a Solar Sail

Given a thin solar sail of Mylar coated with aluminum or silver we can predict some but not all the effects of environmental factors discussed in preceding sections. Meteoritic effects of all particles reported by Whipple in the case of a 10^{-4} inch thick coated Mylar sail will produce penetration rather than surface

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[1] Ibid.

erosion. Reference 123, p. 20, suggests that "meteroids larger than the wall thickness would make a hole equal in diameter to the material." Meteroids whose diameters are of the same order or smaller than the wall thickness would penetrate explosively. Treating Whipple's data as of the first category and summing all the cell intervals we arrive at the interesting result that the meteoritic effect would be to pierce .003% of the sail area/year. Certainly not an alarming result for microholes and low stress levels.

Proton sputtering is more serious, again using Whipple's data, we arrive for aluminum at an erosion rate of $\approx 10^{-6}$ in./yr. For a space vehicle requiring 1 - 3 year life we obviously need therefore a minimum aluminum coating of $\approx 10^{-5}$ in., or $\approx$ 3000 Angstroms. Compared to 10^{-4} in Mylar this represents a coating weight of approximately 20% above that of the film alone, or 17% of the overall weight.

NASA experiments on 2200 A^{o} aluminum vapor coated Mylar concludes that:[1] Ultraviolet protection by the aluminum is probably satisfactory.[1] U. V. transmission is 1 part in 10^{13} at between 4000 - 833 A^{o}, then increases to 10% at 500 A^{o}. At the latter wavelength "the intensity of the sun's radiation is not yet known," Nor is the "effect of prolonged exposure of the plastic to about 10% of the intensity in this wavelength range...." Again this year's communication satellite experiment should give some answers.

Based on the previously quoted experimental results of Ref. 30, p. 10, the ultraviolet effects in a vacuum may not be as serious as in an oxygen atmosphere. As for radiation damage by X rays, cosmic rays and secondary emissions no detailed information be-

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[1] Ref. 109, p. 4

yond the information given in Section 1. 3 was found, and the remarks on the forthcoming NASA satellite apply here too.

2.0 TOWARDS A SYSTEMATIC DEVELOPMENT OF SOLAR SAIL DESIGN

In any new field one of the necessary steps is the development of logical classifications, symbols, and terminology as a prerequisite for further systematic development. The writer feels that solar sailing is ready for such a step to follow the creative work of its early contributors: Wiley (Ref. 108), Ehricke (Ref. 100 and 100A), Garwin (Ref. 102), Sohn (Ref. 104), Tsu (Ref. 106), and Cotter (Ref. 98).

A first question is what is a useful index, or figure of merit, to replace the mass ratio or specific impulse indexes which apply to chemical, nuclear, and ion propulsion systems. Solar sails, we recall, have a mass ratio of unity and a specific impulse of infinity. A suitable index has been proposed by Cotter at Los Alamos Scientific Laboratory. He defines a dimensionless parameter, "Lightness" for which we shall use the symbol "L". Lightness is defined by him as "the sun's maximum radiation-pressure force divided by the sun's gravity force on the whole device."[1] In our notation this would be:

$$L = \frac{\int_{o}^{A} B \, \partial A}{F_g} \qquad \text{for } \alpha = 0 \qquad (16)$$

We note that a simple conversion between "L" and vehicle acceleration "a" exits, namely,

$$.0194 \, L = a \text{ in ft/sec}^2 \qquad (17)$$

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[1] Ref. 98, p. 5

$$6.2 \times 10^{-4} L = a \text{ in units of } g_o \text{ (earth gravity)} \qquad (18)$$

$$L = a \text{ in units of } g_S \text{ (solar gravity)} \qquad (19)$$

where:

A = sail area

B = bounce

F_g = solar gravity force on vehicle

α = angle between normal to the sail and the solar radius vector.

L has units of lb force/lb force

Since Bounce and solar gravity both follow the inverse square law, the Lightness number of a vehicle is invariant for any distance from the sun and " is a measure of the inertia of the device. " Obviously, the higher the Lightness number "L", the greater the acceleration capability of the system.

To separate payload and controls from sail properly we shall define:

$$L_o = \frac{\int_o^A B \quad A}{F'_g} \qquad \text{for } \alpha = 0 \qquad (20)$$

where $F'g$ is the gravity force on the sail alone. This introduces "L_o" as a measure of merit of the sail design irrespective of the payload. We thereby have a figure of merit for the sail irrespective of the choice made in tradeoff between more payload and greater vehicle acceleration.

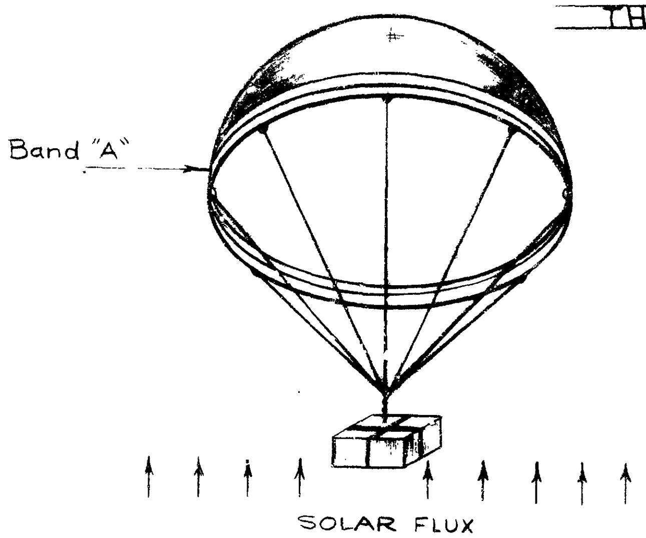
Given the above figure of merit to evaluate sail designs, the writer wishes further to propose the division of all solar sailing devices into three broad design classes. We will then proceed to demonstrate the impossibility of one entire such class in its pure form as a payload carrying design. The proposed three classes are, respectively: Self-Supporting Membranes, Rigidized Sails, and Field Effect Sails. We further propose to break down the latter two classes into sub-classes for which we will develop concrete illustrations, and whose important aspects we will briefly describe.

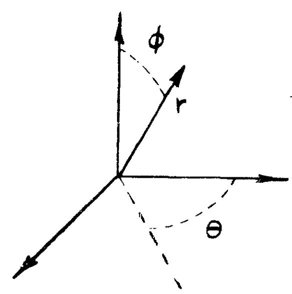
2.1 The Self-Supporting Membrane

As the name implies, a "self-supporting membrane" would be a structure which carries a load in essentially pure tension and whose equilibrium position would be due to balance between Solar Bounce Forces and tensions in the sail. If one excludes the special case of a homogenous surface with either no payload or a uniformly distributed payload, one must assume that the connection between payload and sail may be either in the middle, at the edges or at some intermediate point. Let us first consider a configuration similar to a parachute and we shall then suggest the generalization of our results. Figure 7 shows such a device. We shall, for the moment, make no assumptions as to the equilibrium shape of such a sail except for circular symmetry. First let us consider an intuitive approach to this problem. In a parachute, the sides of the parachute are held out by gas pressure which according to well-known principles is a constant, normal to the level surface. For light, however, as the sides approach an angle of incidence of 90° the forces opposing collapse of the sail towards the sail axis

FIGURE 7

THE "SELF-SUPPORTED" MEMBRANE

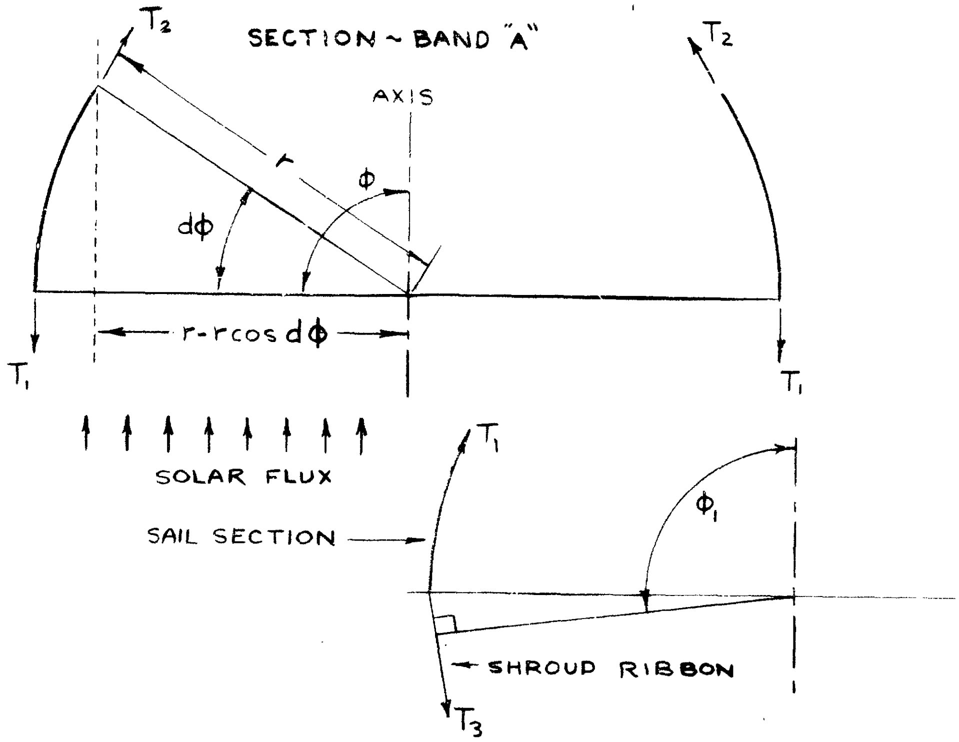




COORDINATE AXES

FIGURE 8

FORCE ANALYSIS-"SELF-SUPPORTED" MEMBRANE



approach 0 as $\cos^2 \alpha \rightarrow 0$. The horizontal components of tension, however, do not have a $\cos^2 \alpha = \cos^2 \phi$ term and therefore do not go to zero as rapidly. This means that in the absence of sail stiffness, a small element cannot start to curve (giving a local finite radius of curvature) without violating equilibrium. It is this rapid decrease of available bounce forces near the edge that prevents an equilibrium solution for finite curvature.

Now we shall proceed to demonstrate the above statements in a more rigorous fashion. Figure 8 shows a section view of a particular ring in Figure 7. We shall consider a band (using spherical coordinates) located by the angle ϕ.

Clearly if there is a payload there is a ϕ such that $\phi = \frac{\pi}{2}$ since without stiffness the sail elements immediately adjacent to the tension shroud ribbon connecting the payload must assume an angle parallel to the shroud, or have $\phi_1 > \frac{\pi}{2}$. Therefore, such a band as that in Figure 8 must exist where $\phi = \frac{\pi}{2}$ whether at the edge of the sail or some other point.

Let us write an equilibrium expression in the θ direction (outward). We note that for the case of the sail axis directed towards the sun and with 100% reflectivity,

$$B_o \cos^2 \alpha = B_o \cos^2 \phi \tag{21}$$

where α is angle between sun and normal to the sail at any point. Denoting the tension in the sail at ϕ by T_2, and a very small width of sail at $\phi = \frac{\pi}{2}$ by ϕ,

$$-T_2 \sin \phi[2\pi(r - r\cos \phi)] + B_o \cos^2\left(\frac{\pi}{2} - \frac{\phi}{2}\right)[2\pi r(r\, d\phi)] = 0 \qquad (22)$$

using:

$$\sin d\phi \approx d\phi \qquad (23)$$

$$\cos\left(\frac{\pi}{2} - \frac{d\phi}{2}\right) = \sin\frac{d\phi}{2} \qquad (24)$$

$$-T_2\, d\phi\, (2\pi r) + B_o \left(\frac{d\phi}{2}\right)^3 \qquad (25)$$

$$T_2 = B_o r \left(\frac{d\phi}{2}\right)^2 \qquad (26)$$

Therefore, for:

$$\begin{array}{l} \phi \to 0 \\ T_2 \to 0 \\ \text{unless } r \to \infty \end{array} \qquad (27)$$

Equation 22 then tells us that equilibrium cannot be satisfied for $T_2 > 0$ if r is finite.[1] Therefore, the sail will collapse inward at this ring and obviously this mode of failure will propagate. We note that the only specification on the load we have assumed is that its diameter be less than the sail diameter. The only specification we have made on the sail shape is that it be symmetrical and have a circular cross section normal to its axis. By using an element dA instead of a ring $2\pi r dr$, as in our analysis, one can show that

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[1] Technically this statement is not correct as was proved to the writer by Prof. Sandorff. If one were to have a sail 200,000,000 miles in diameter making a hemisphere around the sun then, of course pure tension members across the bottom of the sail would hold it in equilibrium. This is the only exception known, and is mentioned as an amusing example of the difficulty in extrapolating boundary conditions to infinity.

this latter restriction is not a necessary one. If one considers a case where the sail is not normal to the solar flux, one can show that for 180° of the ring circumference the $B_o \cos^2 \alpha$ term reverses its sign and helps to produce collapse, therefore changing the orientation will not improve stability. In the above equation we see that if "B" was a scalar constant as is the case for a gas pressure we would find an equilibrium position and, in fact, would be solving the problem of a parachute supported by dynamic gas pressure. Since our analysis considered only local equilibrium on a ring without assumption as to shape it can be readily seen that only slight modification of our treatment would lead to the same conclusions for shroud ribbons located near the center or at any intermediate point.

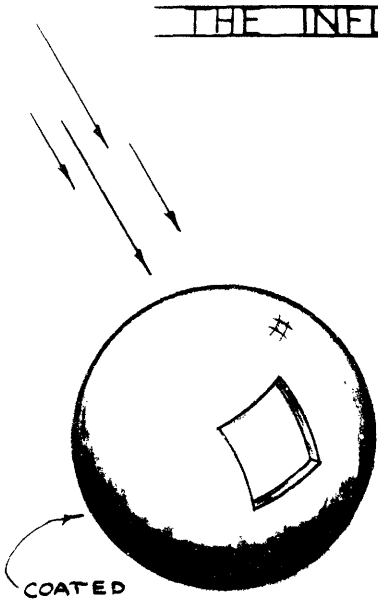
2.2 Rigidized Sails

This class represents designs utilizing compression members of appreciable stiffness. Subclasses include: Inflated Sails, Peripherally Stiffened Sails, and Spider Web Sails.

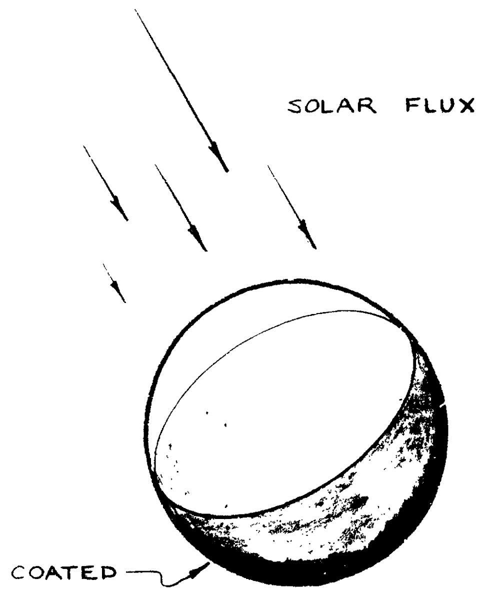
2.2.1 Inflated Sails (Figure 9)

The inflated sail can, in principle, have many shapes. It is characterized by being an essentially three dimensional configuration as compared to the primarily two dimensional shapes considered elsewhere. The first proposal of this type was by Krafft Ehricke of Convair-Astronautics (Ref. 100 and 100A). He suggested silver coated spheres which would have a small amount of hydrogen inside to provide a 400 psi skin stress in the 10^{-4} in. thick Mylar film. He suggested the provision of small windows of clear

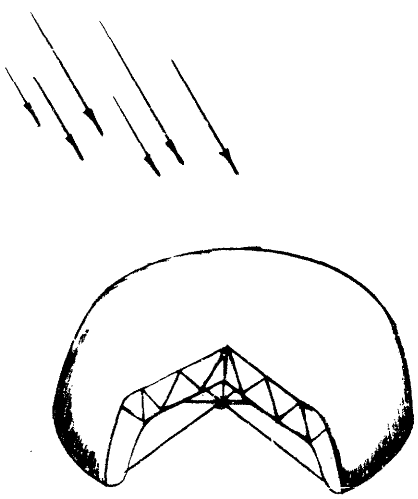
FIGURE 9
THE INFLATED SAIL



9a. FULL SPHERE



9b. HEMISPHERE



9c. FLYING "SAUCER"

Mylar to keep the gas at an acceptable equilibrium temperature. He felt that the Mylar's own rigidity and the broad weight distribution would be sufficient after it was once inflated to keep it essentially spherical. With a payload, of course, this last assumption is open to question, and some stiffners would be required thus making a combination vehicle with the "Spider Web Sail."

It is very interesting to note that Ehricke's vehicle is almost exactly the same as the NASA passive communication satellite to be sent up this year, and this raises the point that Bounce will be an important, although in this case unintentional, effect. Bounce will undoubtedly strongly contribute to changing the satellite's orbit.

As a solar sail, a fully silvered sphere lacks orientation control. A half-silvered one would have control (one hemisphere only coated), but at the price of serious ultraviolet exposure problems, for which Mylar would probably not be suitable. More serious however, is the sphere's poor Lightness number.

Total radiation force is only $B_o \frac{\pi r^2}{2}$ (119)

as derived in Appendix II. While solar gravity force is :

$$4\pi r^2 \frac{t \rho g}{g_o} \qquad (28)$$

where:

g = solar gravity
g_o = earth gravity
t = thickness in feet
ρ = density (lb/ft^3)
B_o = Bounce in psf
L = Lightness number

so that L =

$$\frac{B_o g_o}{8 \rho t g} \qquad (29)$$

An improved lightness number is obtainable with the "flying saucer" (Fig. 9c) which has controllable orientation, but suffers from the same puncture problems.

2.22 Peripherally Stiffened Sails

This type rigidized sail involves essentially a stiffening ring at its periphery. Such a ring need only provide enough stiffness to prevent the mode of failure of the membrane of Section 2.1. The stiffness required to avoid collapse can easily be calculated as a function of sail diameter, sail shape (how taut), size and vector orientation of inertial reaction forces of the payload on the sail. This has been done in Appendix III. The stiffener involved could be an inflated ring with independent sections to minimize the danger of collapse, but a study of Whipple's data as it applies to thin films (Section 1.63) does not encourage such an approach. A more realistic approach is to use "tubular pockets" filled with a plastic foam. It might be possible to delay the foaming process until the vehicle was in space, thus allowing the sail to fit into a compact payload for launching into orbit. Such an idea has been proposed by Cotter.[1] An alternative is the use of a permanently stiff material such as a heavier plastic or light metal stiffeners (Piano wire). These might raise serious launch and deployment problems for reasonable size sails. (Useful sails are of the order of 150 - 1500 feet in diameter.)

No proposal using the peripherally stiffened sail has been made to date. It offers potential passive static stability (see Section 2.6) and a good lightness number for small [redacted] diameters, as is apparent from Appendix III. NASA is said to be planning to investigate an inflated ring design for a communication satellite. (Ref. 136, p. 143

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1 Ref. 98, p. 12.

FIGURE 10
PERIPHERALLY STIFFENED SAIL

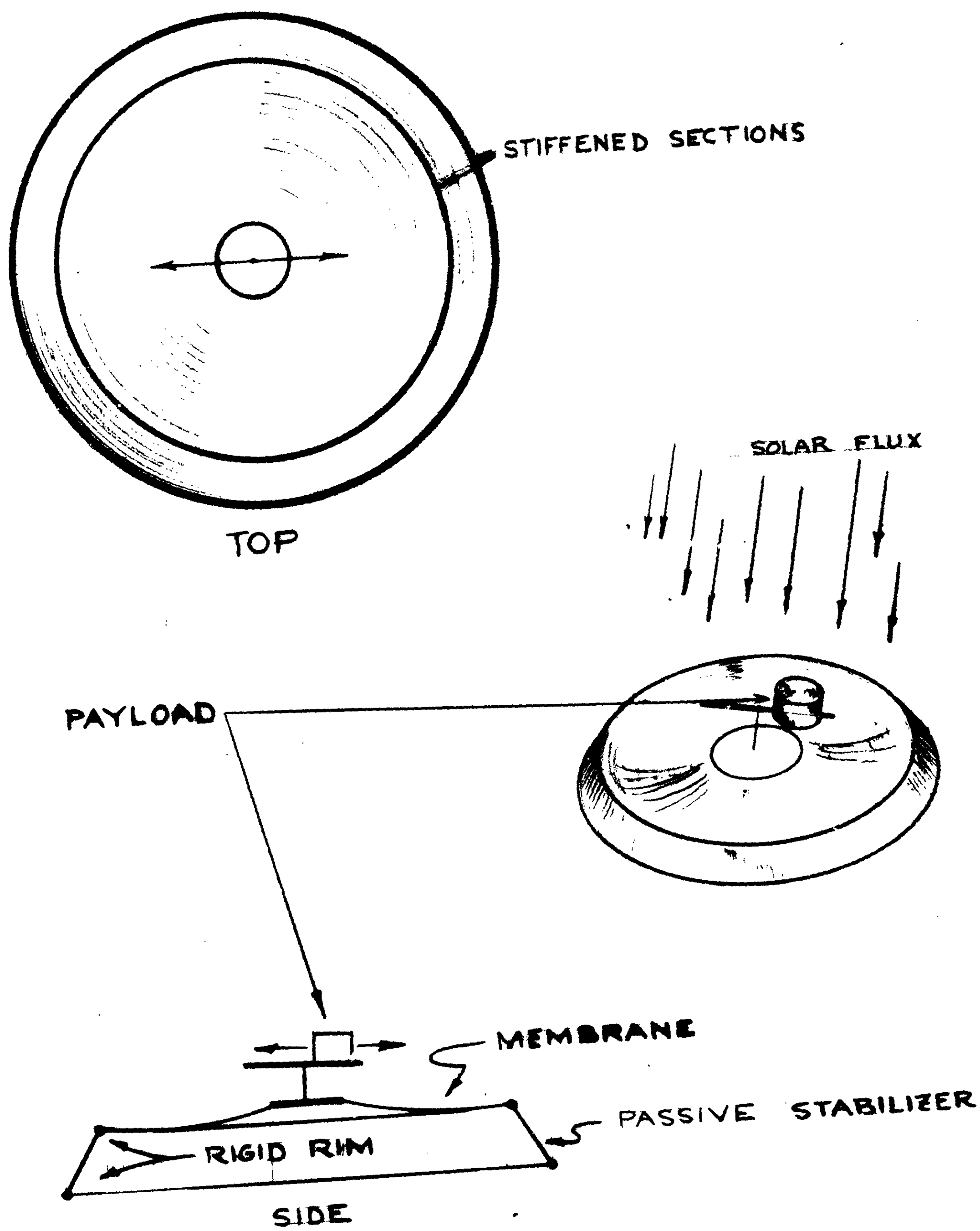
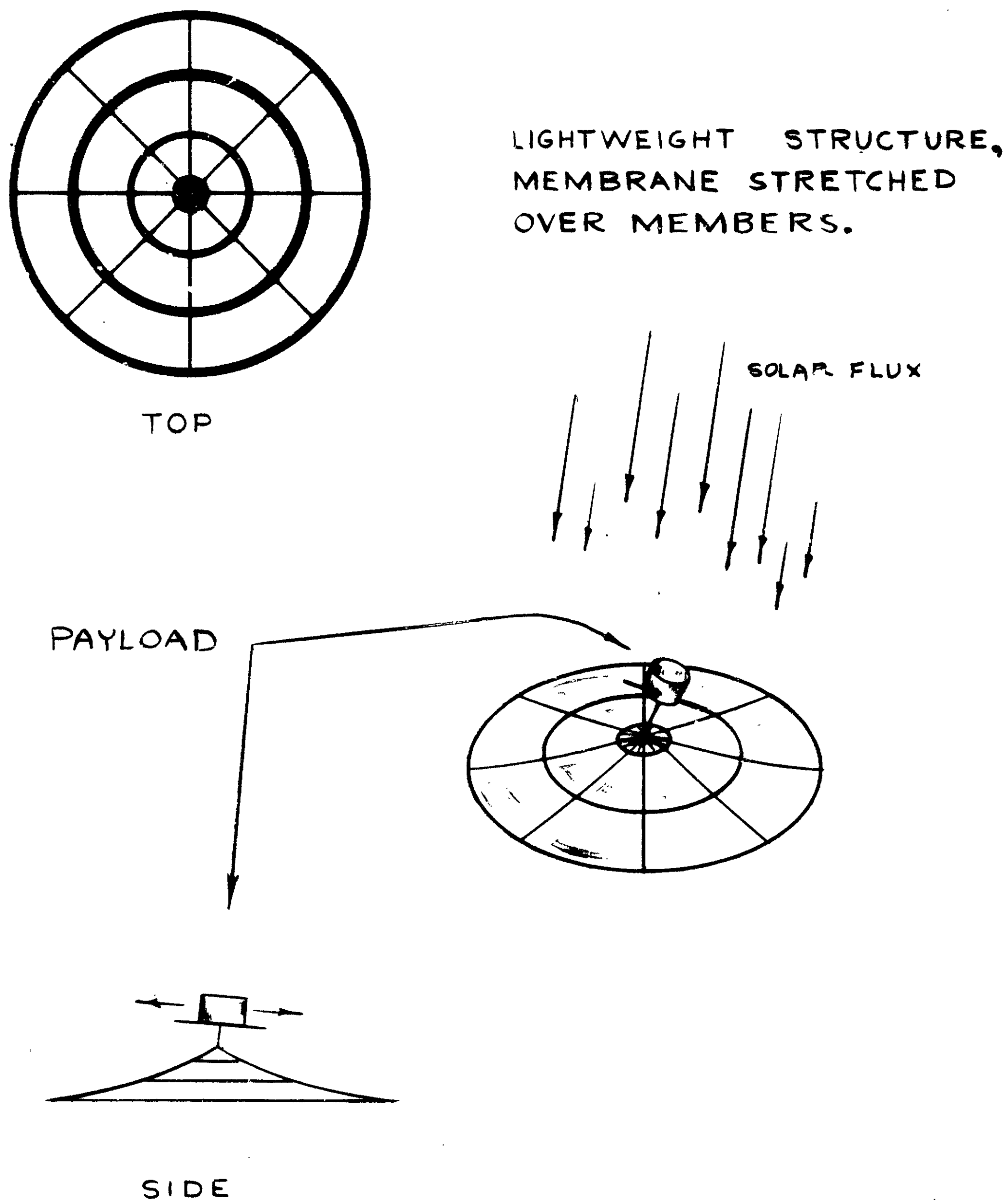


FIGURE II
THE "SPIDER-WEB SAIL"



2.2.3 The Spider Web Sail (Fig. 11)

This type of construction involves a panel and stringer construction. Panels might be coated Mylar or even thin aluminum. The stringers could be of the foamed in place Mylar tube variety, or given the low forces involved, (total pressure forces on a 1000 feet sail 1/10 lb) some piano wire type of structural matrix, or light honeycomb construction.[1] (See Section 2.4 for a discussion of materials and structural problems.)

The web type construction offers good potential lightness numbers since a virtually flat disk is possible, in addition, passive static stability becomes possible as brought out in Section 2.6.[1]

2.3 Field Effect Sails

As we have seen the central problem of solar sail design can be reduced to the catch phrase " more Bounce to the ounce. " We wish to optimize the lightness number L_o, which is related to but yet (as proven for the case of the sphere in Appendix II) is different from the criteria of optimum projected area/weight ratio.

Available field forces include electrostatic, magnetic electro-magnetic (including light), gravitational (in the form of gravity gradients) and inertial. The first two are handicapped by our requirement for a surface as flat as possible. Repulsive forces are a vector effect along the line joining the individual charge or dipole pairs and at the limit for a coplanar surface the repulsive force

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[1] For an excellent generalized comparison of the merit of various materials and structural configurations for space missions, see Ref. 117 by George A. Hoffman of the Rand Corporation, and for a comparison of pressure stabilized vs. stringer and sandwich constructions, see Ref. 116 by George Gerard of NYU and Ref. 118 by Paul Sandorf of M. I. T.

preventing bending goes to zero. As for electromagnetic effects in the case of solar flux, we have shown the collapse caused by its action (Section 2.1). Any eddy currents in a satellite orbit would of course also have less and less "Rigidizing" effect as the sail became flat. Gravity gradient effects are small as compared to solar bounce. Inertial effects in the form of centrifugal acceleration are of great potential interest. They have been proposed by Cotter (Ref. 98, p. 14) in the form of a counter rotation between payload and sail. We will propose a different variant in Part III and will consider this very promising approach in great detail.

2.4 Materials for a Solar Sail

As is shown later (Appendix V) stresses on a solar sail are a few psi, with total solar force of $\approx$ 0.01 lbs for a 250 ft diameter sails. Therefore, the limiting factor on Lightness number is the ability to produce thin films, and to package them on earth. Such factors as meteoritic effects and erosion (Section 1.63) have been shown to be less restrictive.

The polyester film Mylar is at the present time the lightest continuous film available in appreciable sizes. It is now produced in 1/4 mil. thicknesses with a weight of .0018 lb/sq. ft. At present rolls 58 inches wide and 20,000 feet long have been made. Mylar cannot be heat-sealed, but 1/2 inch wide 1/4 mil. tape is available at .0036 lb/sq. ft. with a tensile strength of 1500-2500 psi, temperature range of -184°C to 138°C (melting point), and satisfactory environmental and aging properties.[1] Samples of 1/10 and 1/20 mil Mylar exist

[1] Private communication, Mr. Ronald L. Larsen, Schjeldahl Company who are manufacturing the 100 ft. Mylar NASA satellite.

and given adequate demand could undoubtedly be produced.[1]

For our further calculations, we shall assume 1/10 mil. Mylar and use .001 lb/sq. ft. to account for bonding, non-uniformity in thickness, etc. Mylar service temperatures are quoted by Dupont as -60 to 150°C. Further properties of interest are reproduced in Table IV. No data on damping properties or stiffness is presently available. Mylar is susceptible to degradation under ultraviolet exposure.

Other plastic films that might be of potential interest include: Polyethelene (less strength, better ultraviolet properties, can be heat sealed but not available in 1/4 mil thickness), Polyvinyl Fluoride (good ultraviolet resistance), Makro Fol and Isolier Folie DO-202 (German polycarbonate films), M X D-6 (metaxylylene adipamide), polyvinyl alcohol, and Kel-f.[1]

A very interesting, if somewhat speculative possibility, (see Ref. 108) is to choose a film backing that would sublime in space leaving a 3000 A° layer of aluminum on a grid matrix of a non-volatile material such as Fiberglas. Stress calculations show that this possibility is an interesting one. The major obstacle is to develop such a backing (required to handle the sail prior to launch). Because of earth gravity forces large scale testing of this approach may prove impossible. However, experiments at NRC on lifting thin vacuum deposited films from the surface of a solvent have been successfully conducted.[2]

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[1] Private communication, Dr. T. C. Tsu, Westinghouse Research Laboratories.

[2] Private communication, Dr. J. Simon, National Research Corporation, Cambridge, Mass.

TABLE V
PROPERTIES OF MYLAR*

PHYSICAL

Property	Typical Value (1 mil film) 10,000 lb/sq. in. at 150°C	Test Method
Tensile strength	20,000 lbs/sq in.	Instron tensile tester
Tensile modulus	550,000 lbs/sq in.	Instron tensile tester
Break elongation	75%	Instron tensile tester
Impact strength	60 kgm.-cm	DuPont falling ball impact tester
Bursting strength	45 lbs.	Mullen
Tear strength	15 gms.	DuPont single sheet tear tester
Flex life	20,000 cycles	DuPont flex tester at 25°C.
Bending recovery	43%	Immediate 180° bend recovery
bending recovery	51%	60 second bend recovery
Density	1.39 gms/cc.	Density gradient tube
Area factor	20,000 sq. in./lb	Calculation
Refractive index	1.64 ^{n}D-25°C.	Abbe refractometer
**Light transmission	about 90% in visible > 4000 A°, zero at < 3000 A°	
CHEMICAL		
Moisture absorption	Less than 0.5%	1 week immersion at 25°C.
Oxygen permeability	0.90 gms/100 sq m/hr	General Foods tester
THERMAL		
Thermal coefficient of expansion	15×10^{-6} inches/inch/°F	From 70° to 120°F
Conductivity coefficient	3.63×10^{-4} cal/cm/sec/°C.	Cenco-Fitch method

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* From Ref. 100, Table 2.
** From Ref. 120

According to Mr. Carey of the Civil and Sanitary Engineering Department at M. I. T., a development cost of around $500,000 could produce a practical solution. He feels that a highly plasticized vinyl coated solution of 1/10 to 1/20 mil could be edge sealed and aluminum coated on rolls and would not only be practical to handle on earth but would quickly sublime in space.[2] In any such device since the stress in the sail near the hub varies approximately as $2\pi rt$, a central section of increased thickness would be required. Potential improvements over a 10^{-4} in Mylar 3000 A^{o} aluminum coated sail could increase L_o by a factor of 5 to 6. The limiting value is that of the aluminum coat alone whose minimum thickness as seen in Section 1.64 is dictated by considerations of proton sputtering losses.

As for use of thin metallic films alone without a plastic backing, the thinnest aluminum films known are of the order of 2×10^{-6} inches and were used in the German Fabry-Perot interferometers in the 1930's.[3] A web structure using a titanium mesh of 20 seconds/meter and .001 in. diameter with 2×10^{-6} in. aluminum panels has been suggested by H. J. Gale of M. I. T.[4] While this proposal produces the very low weight of $.42 \times 10^{-4}$ lb/ft^2, the problem of erection seems extremely difficult and the aluminum thickness is below the one probably necessary to have a reasonable life in the face of sputtering. The Titanium grid, however, can be considered as an alternative to the Mylar or Fiberglas mesh proposed earlier.

[2] Private communication

[3] Ref. 107, Section 2.10

[4] Ibid.

Reference 117, Figure 3, seems to indicate glass fibers as offering a more favorable strength/wt ratio. For the stresses involved in a solar sail it is likely that the minimum available wt/length is probably more important since the thinnest strands available are probably strong enough for the forces involved.

2.5 Thermal Equilibrium of a Solar Sail

As previously mentioned (Section 1.2) temperature equilibrium is a function of configuration, orientation and material. A study at NASA has been made of temperature equilibrium of the 100 foot diameter sphere with encouraging results (Ref. 109). For circular orbit they calculate a maximum temperature on the exposed side of 148^{o} - $160^{o}C$, respectively, for altitude of 800 miles and earth albedo of 0.36, and altitude of 1000 miles with an albedo of 0.52. The minimum temperature of the coldest point: for 800 miles, $-104^{o}C$ and 1000, $-108^{o}C$.

The equations used which are derived in the above paper consider both solar radiation, earth reflection and earth reradiation. We quote their results:

$$C_S A_S + \frac{1}{4}\frac{\epsilon_i}{\epsilon_o} C_S A_S \alpha \left\{ 1 + 2\left[a + \frac{1-a}{4}\frac{A_E}{A_S} \right] \left[1 - (1-k^2)^{1/2} \right] \right\} \tag{30}$$

$$= (\epsilon_i + \epsilon_o)\sigma T_{HS}^4$$

$$\epsilon_i \quad T^4 = \frac{1}{4} \frac{\epsilon_i}{\epsilon_o} C_S A_S \left(\frac{1-a}{2}\right) \frac{A_E}{A_S} \left[1 - (1-k^2)^{1/2}\right]$$

where:

- a = albedo of earth
- A = absorptivity (fraction of incident radiant energy which is absorbed)
- C_S = solar constant
- h = altitude
- k = $R_E/(R_E + h)$
- ϵ = emissivity
- σ = electrical conductivity

For a flat disk type of sail of aluminum coated Mylar, the equilibrium temperature far away from the earth has been calculated by Tsu[1] based on estimated emissivities as 50°C to 60°C.[2] This is, of course, a function of angle, but variations from this value can easily be estimated from Figures 3 and 4, and do not appear serious for reasonable angles. We note that such a sail's conductivity and heat capacity is so low that temperature changes can be very rapid. Tsu calculates that the sail should not be allowed to be on edge with respect to the sun for more than 5 seconds because of the excessive resulting drop in temperature.[1]

2.6 Orientation Control

A solar sail, unless special design features are incorporated, is of neutral static and dynamic orientation stability. Forces in

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1. Private communication, Ibid.

2. He estimates that the range between -50°C and +120°C is the temperature for all conceivable emissivities of the sail surfaces.

space are low, but conversely so is damping. If we are interested in obtaining a desired attitude of the sail and maintaining it, several methods are available. On Class 2, or rigidized vehicles, we can change attitude by moving the equilibrium position of stabilizers shown in Figure 10. These stabilizers as shown in Figure 10 produce a restoring torque against any random motion and if moved by a control device would create torque about a new equilibrium angle. Analysis of all torques around the center of mass show the long lever arms involved. Designwise the above approach is difficult of execution and we will consider others, noting that the stabilizing fins shown in Figure 10 may be desirable for passive stability in any case in vehicles of this type. As we shall show later, they are not necessarily advantageous in spinning sails.

A second approach is to either pull or alternatively release the shroud ribbons or other connections between the payload and one-half of the sail thus creating a torque on the sail, and by reaction shifting the position of the payload. Approximately half way before the desired position is reached the process must be reversed to recenter the payload and avoid overshoot. This process is required to have zero resultant angular momentum. Since angular momentum transfer from Solar Bounce to the sail is non-linear as a function of angle, an equal and opposite momentum transfer must in general be initiated at some point other than midway through the orientation change. The above analysis follows from requirements for a final steady state, and from conservation of angular momentum between the sun and the system.

A different arrangement proposed by Cotter[1] involves first orienting the payload in a suitable manner and then spinning the payload, which by conservation of momentum will cause the sail to rotate. After the midway point is reached, the maneuver is reversed and both payload and sail return to zero angular velocity at the desired sail orientation. Of course, the difference in inertias between sail and payload is such as to require a relatively high angular velocity for the payload. A typical ratio between the two inertias might be 10^5 to 10^6 (for a 1000 lb. payload and 1000 feet diameter sail). This system in turn would produce appreciable centrifugal stresses on payload components, and might thereby affect reliability and structural weight.

A possibly more attractive solution also taken from Cotter's work[2] is that of a class 3 sail - where the sail is a spun disk. Cotter suggests spreading the payload so as to increase its inertia and spinning the payload in an opposite sense to the sail. With equal angular momentum for payload and sail he suggests that merely by applying a torque between the two "gyroscopes" (payload and sail) we will induce a mutual precession about their common axis. An important advantage of this method is that the system has gyroscopic stabilization and is therefore relatively insensitive to small disturbances. For a 1500 foot sail rotating at 1/2 rpm Cotter calcu-

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[1] Ref. 98, p. 12

[2] Ibid, p. 14

lates a precession rate of approximately 1 rev/2hrs. Control is simplified since precession has no overshot and the sail will stay in whatever orientation it finds itself when the mutual torque is removed.

A final, and as far as the writer knows an original method, is that of payload shift. It is felt that its simplicity may be valuable in terms of reliability of control. Payload shift, as the name implies, consists in shifting the center of mass with respect to the center of Bounce (pressure). This is done by placing the payload on a movable arm such that by changing the moment arm r and an angle θ (where θ is a polar angle in the sail's plane measured with respect to the sail) we can introduce a torque in any desired direction.

This system is suitable for two position "on-off" control, i.e., payload either extended with relative angle and distance r or retracted. The application of the method of payload shift to a centrifugally supported sail will be extensively treated in Part III (Section 8.0).

To generalize the preceding survey, attitude control for a sail can then be accomplished through four basic approaches:

1. Asymmetrical Bounce by configuration change
2. Asymmetrical payload application
3. Reaction forces (including reaction torques)
4. Cross-coupling of forces (precession)

We did not discuss, as of little general promise for solar sail attitude control, field forces: gravity gradient, electromagnetic, etc.

3.0 Possible Application of Solar Sailing

We have classified potential solar sailing applications into promising, possible, and impossible or impractical. This list includes both applications suggested in the literature and a few new ones. The list is not meant to imply any completeness or definitiveness, but should be considered as what it is, both a survey and an expression of opinion.

3.1 Promising Applications

In general, one can start by ruling out all solar sailing applications below an altitude of 400 miles where Solar Bounce first becomes larger than aerodynamic pressures. No applications except for attitude control should be considered below 800 - 1000 miles at which point aerodynamic drag ceases to be of major importance.[1] Three very different types of applications are considered as promising. They are "primary propulsion", "trajectory perturbation" and "attitude control".

3.1.1 Solar Sailing as Primary Propulsion

A solar sail powered vehicle is well-suited for modest payloads (up to 1000 or 2000 lbs) to the near planets. It simplicity, inherent reliability, unlimited energy, reasonable weight, simple payload orientation, and excellent midcourse correction capabilities, as well as relatively relaxed navigation requirements are some of the reasons that make it of interest. Part III of this paper which discusses a solar sail propelled Mars probe will develop those advan-

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[1] Ref. 56, p. 111

tages as well as problem areas in considerable detail.

3.1.2 Solar Sailing for Trajectory Perturbations

Solar sail vehicles to get an acceleration of 10^{-4}g or L of 0.16 are of very large diameter. There are a number of interesting ballistic trajectories, however, where only relatively minor perturbation capabilities are required. An "auxiliary" sail of $10^{-5} g_o$ or less capability might be very adequate for these purposes. We note that $10^{-5} g_o$ gives a ΔV of 128 ft/sec/day and that most interplanetary ballistic trajectories speak of auxiliary correction capabilities of the order of 2000 - 3000 fps over trips of two or more years durations. For example, the MIT Instrumentation Laboratory Mars probe design has a ΔV capability of 2,250 ft/sec (Ref. 135).

Specific applications include orbit adjustment maneuvers for high altitude satellites, such as the 24-hr satellite. There the unlimited energy capability of the sail, and its physical size as a good radar, radio and optical reflection would be substantial advantages. In addition, the absence of large areas vulnerable to micrometeorite penetration (as is the case with radiators) is an important asset. A 10^{-5}g sail incidentally might weigh only 10% or less of the payload weight.

Another interesting if somewhat speculative idea is to use a solar sail as a means of orbital correction capability for lifeboats, as will undoubtedly be of eventual interest in manned vehicles (For a design study on space lifeboats, see Ref. 127.) Desirable characteristics include long storage life, low cost, compactness, dual function as attitude stabilizers (human movement in such a light vehicle might otherwise cause large angular velocities.) Further, these relatively large sails would be very valuable for optical and radar tracking by rescue vehicles. A small perturbation capability is desired to allow for instance to deviate from a re-entry course which a "lifeboat" might not be designed to sustain.

3.1.3 A Solar Sail as Attitude Control Device or as an Angular Momentum "Sink"

Many vehicle attitude control proposals employ flywheels, (Ref. 6, 8, 18), spin stabilization, (Ref. 3, 7, 9), or microthrust rockets (Ref. 14) (for comparisons between different methods see References 1, 2, 12, 13). All the above systems have the advantage of relatively fast, orientation insensitive, response. They have already been used in various applications (according to Soviet data flywheels were used on their lunar probe). All the above systems, however, have the limitation of being eventually saturated in the presence of small but persistent unbalanced torques which for an earth satellite include Solar Bounce, and earth (Ref. 3) gravity (Ref. 104), and centrifugal gradient effects (Ref. 4, 10), aerodynamic forces (Ref. 5, 17), and interaction with the earth's magnetic field (Ref. 21, 134). For a space probe, of course, all systematic unbalance effects other than Solar Bounce become negligible. Faced with the problem of stabilizer saturation, several solutions are possible. For low orbit satellites (up to about 300 miles) aerodynamic stabilization systems are interesting (Ref. 18), for many other satellite missions magnetic field interaction attitude control has interesting possibilities as a method of "dumping" angular momentum (see Ref. 134 for a specific design proposal, involving the generation of eddy currents).

However, for many high altitude satellites and all ballistic space probes, a solar Bounce altitude control has unique advantages. In the case of a proposed 340 lb. ballistic trajectory Mars probe with microthrust jets for midcourse correction, a series of four " solar vanes" have been proposed. In this careful four volume treatment of a Mars photographic reconnaissance by M. Trageser and associates at the M. I. T. Instrumentation Laboratory, (Ref. 135) flywheel altitude control is supplemented by four " solar vane " sinks. For simplicity, Solar Vane control is restricted to " on - off" control. The metal vanes are either extended or retracted. Four vanes, one on each side, are set at 45^{o} tilt with respect to the vehicle axis. Each vane is of 1600 sq. cm and has a B_o of approximately 6 dyne cm = 42×10^{-8} ft. lb. They can be operated in pairs to produce a couple in any desired axis. By using the flywheels to hold them in a fixed attitude, angular momentum is " dumped" at a fixed and calculated rate, and the flywheel is allowed to slow down correspondingly.

A general treatment of a solar sail as a stabilizer and dynamic damping system is given by R. L. Sohn of the Space Technology Laboratories (Ref. 104). Sohn evaluates the sail for use on a lunar probe. For a derivation of the appropriate equilibrium equations for Solar sail stabilizer see Appendix IV. The derivation in Appendix IV takes into account the previously discussed vector nature of Bounce, B_o, as opposed to the scalar nature of a gas pressure Po. Sohn's calculations, taking onto account the above comment, still show very reasonable stabilization times for representative gravitational and solar bounce asymmetries.

A special feature of a solar sail attitude control system not previously mentioned in the literature is that contrary to other stabilization torques, it is primarily a heliocentric effect. For such projects as an orbiting satellite telescope the value of this is obvious, inasmuch as the telescope orientation must be maintained fixed in inertial space and not with respect to earth. Using the expression $B = B_o \cos^2 \alpha$ where B_o near the earth is 1.95×10^{-7} psf for a stabilizer of area A sq. ft., and mean distance to c.g. of d we have available torque of 1.95×10^{-7} A d $\cos^2 \alpha$.

For a purely passive system where other disturbing torques are of the same order as the sail the restoring force is small, for small disturbance $\alpha \lessapprox \frac{\pi}{2}$, and for a permissible oscillation of β^o

Where β is a small angle

an average restoring force of approximately

$$\frac{1.95 \times 10^{-7}}{3} \cos^2 \beta \, (A \text{ [redacted] } d) \qquad (32)$$

is available. However, we also note that $\frac{\partial \cos^2 \alpha}{\partial \alpha} = -2 \sin \alpha \cos \alpha$ and therefore maximum sensitivity occurs for $\alpha = 90^o$ and minimum sensitivity for $\alpha = 0^o$. In the case of an active system with an on-off mode acting as an angular momentum dump in conjunction with a flywheel altitude control system a much smaller sail can be used since the available torque becomes approximately:

$$1.95 \times 10^{-7} A d \text{ psf} \tag{33}$$

When the stabilizer is oriented by the flywheel normal to the sun. This is the advantage of the extendable type of sail as developed in the MIT proposal.

Of obvious interest is a dual mode where gross motions are reduced by flywheel and stabilizer, after which the stabilizer is allowed to assume its neutral orientation. As mentioned by Sohn, dynamic damping can be then introduced into the stabilizer system, by providing motion to the stabilizer in such a way as to provide a proportional restoring force.

An entirely different mode of operation providing attitude stabilization, an angular momentum sink, and damping in a single integrated mode of operation is that of the "Self' Stabilized Stabilizer." This approach, as far as the writer knows, has not yet been suggested. It appears to offer a number of substantial advantages and for that reason is discussed in detail as Part II.

3.2 Secondary Applications of Solar Sailing

Deemed of less interest than applications in Section 3.1 but none-the less possible are two further types of application, namely radiation tracer bodies and relative motion control.

Radiation tracer bodies as developed by Ehricke (Ref. 100 & 100A) are of interest in obtaining data on drag, in the neighborhood of planets and the moon (interplanetary drag he calculates would, however, be too small to observe). Ehricke proposes silver coatings predicting a low equilibrium temperature of -170°k -200°k at which temperature he expects the empty Mylar sphere will have sufficient rigidity. He further suggests that a sphere could serve as a measuring device for micrometeoritic penetration and silver coating erosion through optical measurements (i.e., gradual or sudden fading process). With the rapid evolution of more complex vehicles and larger boosters it is very possible that the above purposes may well be served with multi-purpose vehicles.

We recall that Ehricke's radiation propelled tracer bodies are capable only of ballistic trajectories with a fixed Solar Bounce superimposed in a purely radial direction. Such a body, if launched in a given solar orbit, has an easily calculated path depending uniquely on initial condition and lightness number (Ref. 66, 100). As noted earlier, the NASA communication satellites are a special case of the radiation tracer bodies and Solar Bounce effects are very significant on them.

As for the other type of sail mentioned, a "relative motion control", this can be thought of as an intermediate between attitude control and trajectory perturbation devices. A "relative motion control" would apply to such a vehicle as a nuclear powered ion engine with a payload located (as has been frequently suggested) at the end of a very long tow cable (to take advantage of $\frac{1}{r^2}$ nuclear radiation attenuation and thus economize on radiation shielding). A relatively small sail might then be used much as aerodynamic controls are employed on a glider towed by an airplane. This sail would be utilized to allow limited control of relative position with respect to the ion propulsion unit, and probably more important to damp out any tendency towards pendulous oscillation of the vehicle at the end of a long cable. Because such a device is not of apparent interest in the very near future no attempt has been made to explore it carefully, and it is merely mentioned in the interests of a more complete coverage.

3.3 Impossible or Impractical Applications of Solar Sailing

This section has been included both as a convenient way of bringing out some of the limitations on solar sailing, and also because mere mention of what appears to be an impractical application may lead others to a modified and useful concept. The above classification includes low orbit satellites, "solar tugs", cislunar vehicles, manned vehicle applications except as small perturbation, and space relay stations.

Low orbit satellite sails as we have seen earlier in Section 3.1 are undesirable because they lead to ~~excess~~ excessive drag. A minimum altitude of 800 miles insures fairly negligible drag, but attitude control applications can in some cases be considered at 400 miles.

The "solar tug" is a superficially attractive idea for "ferrying" cargo from a low or medium altitude orbit to a higher one with no expenditure of stored energy, using a solar sail tow vehicle. Considerations of size, transfer time and payloads relegate this idea to the realm of the impractical (see Section 4.1 for appropriate equations and relevant calculations).

Cislunar vehicles in general cannot use a solar sail to great advantage because of the unfavorable performance in the presence of a central force field, such as the earth's, greatly stronger than the available Solar Bounce effect. This combined with the relatively short transfer times make the sail as a means of propulsion (as contrasted to perturbation) far less attractive than nuclear-electric or even chemical systems. As for use as midcourse correction, low transit time to the moon make this process also unattractive.

Solar sails for manned vehicles have been listed as unattractive (especially as primary propulsion) because of weight and (except when used as a perturbative force effect) time factors. A solar sail of approximately 250 ft. diameter will give an acceleration of the order of 10^{-4}g to a 200 lb. payload. For much larger payloads of which manned vehicles are an example either accelerations become excessively low or sail sizes become so great as to raise serious technological questions of present day feasibility. In fact, even a 250 ft. diameter sail requires a certain amount of development effort, but it is generally considered as of manageable proportions.

The space relay station concept involves the use of solar sailing to station one or more communication relays in space with controllable attitudes and position. With the recent development of Maser technology the advantages in terms of signal to receiver noise in using a Maser installation on earth are such as to make even a directional rebroadcast system in space appear of little present interest.

4.0 Interplanetary Trajectories for Solar Sailing

In this section we shall consider separately the two key problems of a potential mission: earth escape and subsequent interplanetary transfer maneuvers. Inasmuch as no fundamentally new ideas will be offered in this section except for inclusion of earth Bounce effects, derivation of the appropriate equations will only be discussed and the reader will be referred to the original references for a more detailed presentation. In any case the material presented here will no doubt be superseded by a more precise computer aided study now being conducted by Dr. Theodore Cotter at the University of Michigan where he is currently a visiting lecturer in Physics. Dr Cotter plans to present his findings in a paper to be delivered at the New York annual meeting of the Institute of the Aeronautical Sciences (IAS), January 25 - 29, 1960.[1]

4.1 The Earth Escape Maneuvers

In some ways escape from earth can be said to differ from escape from the solar system only in the magnitude of the parameters, and hence the time scale. For a solar sail four significant differences stand out. First is that Solar Bounce for earth escape acts as a heliocentric vector. This is to be used to furnish the desired accelerations in terms of geocentric coordinates and this forces use of relatively rapid orientation changes. Second, solar gravity is of the same order of magnitude as Solar Bounce effects, for a reasonable design, but Solar Bounce is at least 4 orders of magnitude less than earth gravity for much of the escape spiral. An analytically important

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[1] Private communication, Dr. Theodore Cotter.

factor is that for interplanetary travel the ratio of Solar Bounce effects to solar gravity effects are a constant, hence the use of a lightness number. For earth escape this of course is not true, instead Solar Bounce effects are a constant, Earth Bounce effects and earth gravity are decreasing. The fourth difference is of course the non-negligible Earth Bounce effect whose distribution is a varying function of position, season, etc., and whose magnitudes are not accurately known.

Its estimated fluctuations as per Appendix I, Equation 112 of between:

$$0.16 \left(\frac{R}{R + D}\right)^2 B_{o_{solar}} \leq B_o \leq 0.52 \left(\frac{R}{R + D}\right)^2 B_{o_{solar}} \qquad (112)$$

does not permit its omission in any but order of magnitude estimates (except for the case of the polar orbit).

In general we can say that an optimum time earth escape program must at least initially optimize (geocentric) tangential acceleration. The impossibility of escape using purely radial acceleration by the use of forces less than 1/8 those of the central force field has been proved by Copeland in Reference 66. Optimization studies for low thrust acceleration devices have shown tangential acceleration to be either the fastest or in some uses close to the fastest method of escape for a given available thrust (Refs. 63, 69, and 55 Sections 8 and 10).

In the case of a solar sail pure tangential acceleration is impossible but we note the proof by Tsu (Ref. 106, p. 9) that only small errors are introduced by neglecting the effect of the radial component of acceleration (for velocity vectors close to circumferential). We also note Ehricke's proof (Ref. 55, 8-113) that the results for a low mean acceleration are approximately the same as for a constant acceleration equal to the mean acceleration. In doing this we sidestep, possibly with insufficient justification the effect on escape time of the degeneration of a circular or elliptical trajectory in the presence of small but repeated unsymmetrical acceleration patterns over many revolutions. Perkins says in his paper that "whether or not the thrust is constant, if it remains less than 1% of local vehicle weight, the mean path of these trajectories will maintain the familiar circular velocity-altitude relationship for some time."[1]

Having assumed the foregoing, we are able to use directly the computer derived plots of position, time and velocity for a constant 10^{-4} tangential acceleration as shown in Figure 13 and 14 which are reproduced from the paper by F. M. Ferebee of Rocketdyne (Ref. 69).[2]

From Figure 13 we see that for a 10^{-4} g constant acceleration starting at a 300 nautical mile orbit about 85 days are required for escape after 337 turns. From 1000 nautical miles using Ref. 74

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[1] Ref. 74, p. 237

[2] For a representation of a large family of such curves for varying accelerations see reference 55, Sections 8 and 10. For non-dimensionalized solutions from which all the pertinent parameters can be easily calculated for cases of interest, see Ref. 74.

Figure 13

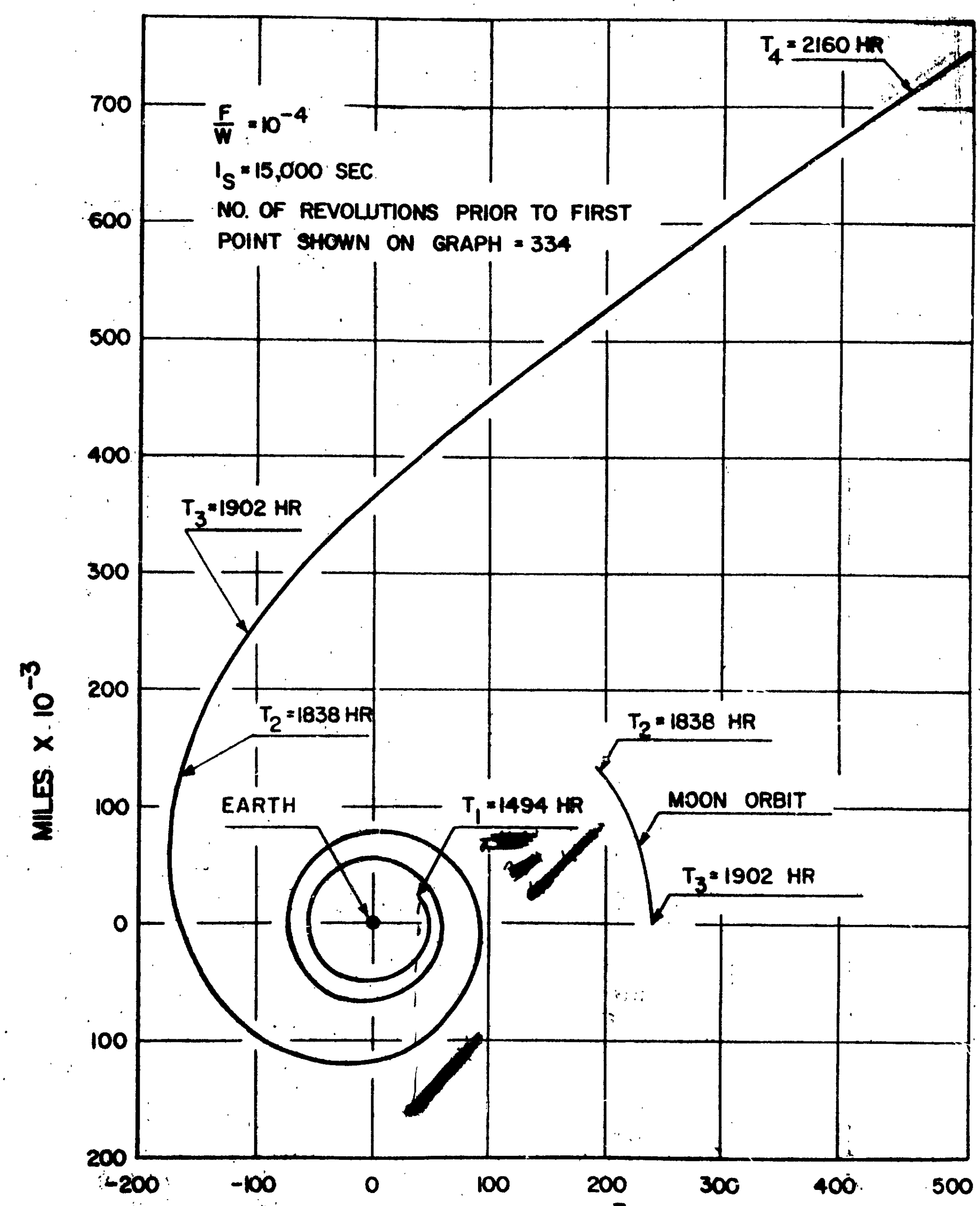
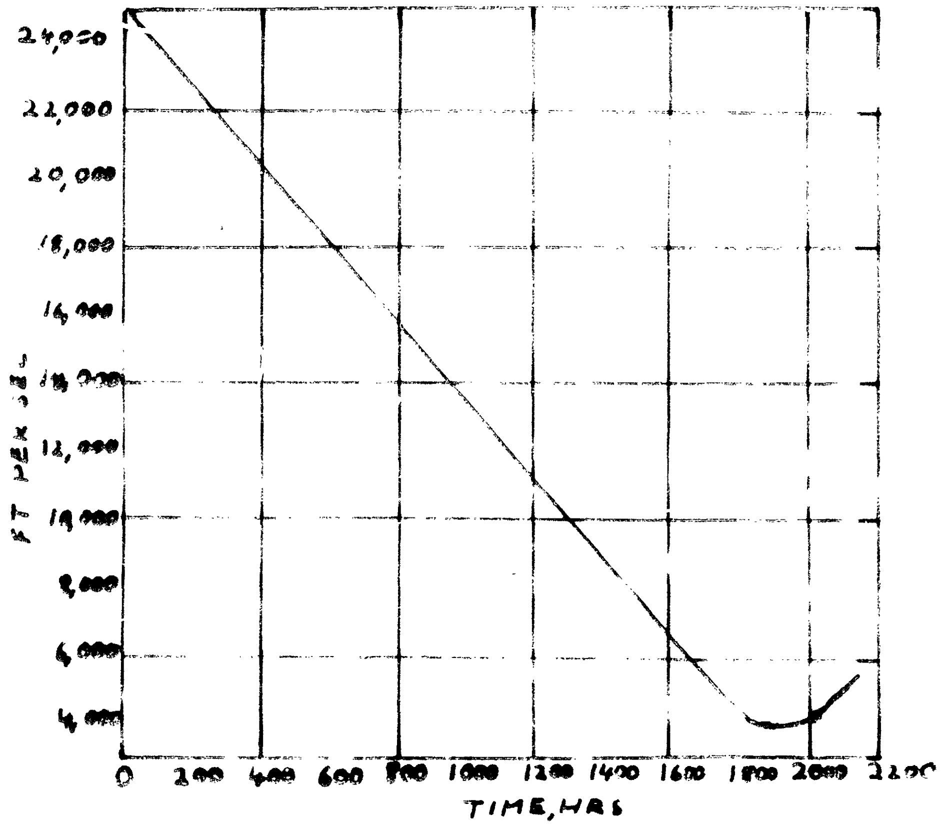


FIG.13. Mars Mission Earth Escape Path

(From Ref. 69 by Ferebee)

FIGURE 14



VELOCITY PROFILE DURING $10^{-4} g_0$ ESCAPE
(FROM REF 69 FIG 10 by Forebee)

we find that only 80 days are needed. We recall that for a solar sail an 800 to 1000 nautical mile launch altitude is needed. If our average acceleration is less than ~~at~~ 10^{-4}g, ~~At somewhat less~~ the time to escape becomes, according to Ehricke, (Ref. 55, p. 8-112):

Constant Tangential Acceleration	Time from 300 N Mile Orbit to Escape
10^{-4}	85 days
$.5 \times 10^{-4}$g	8 months
$.3 \times 10^{-4}$g	12 months

Clearly then, mean accelerations less than .5 or $.6 \times 10^{-4}$g are of very limited interest for earth escape. Given this approximate value, we can at least arrive at some order of magnitude answers. (Noting previous reservations on the accuracy of the answer and leaving moot any gains that might be made through the use of an elliptical orbit.)

Two basic modes of escape offer themselves: One is a polar orbit, (Ref. 98) the other is an orbit in the ecliptic (Ref. 98, 102, 107, 108), with an obvious infinite number of intermediate orbits between the above two.

The polar orbit offers the advantage of having as possible orbit one in which the tangential Solar Bounce component is a constant. As can readily be seen by taking the derivative of $\cos^2 \alpha \sin \alpha$, its maximum is obtained for $\alpha = 35^{\circ}$.

For convenience, let us use $\alpha = 30^{\circ}$ as an approximation, then:

$$B = B_o \cos^2 \alpha = 3/4 \, B_o \qquad (34)$$

$$B_{tangential} = B_o \cos^2 \alpha \sin \alpha = .37\ B_o \qquad (35)$$

$$B_{radial} = B_o \cos^3 \alpha = .68\ B_o \qquad (36)$$

Where B is normal both to the ecliptic and to the polar orbit. The effect of B_{radial} is to gradually push the escape spiral to an escape spiral at slightly larger radius from the sun and, therefore, slightly greater altitude. This effect is of secondary importance, however, and escape is primarily due to the tangential component. We note that a polar orbit escape is also possible with a geocentric sail orientation using only earthlight with:

For α = angle of earthlight incidence

$$B_{tangential} = B_{terrestrial} \cos^2 \alpha \sin \alpha \qquad (37)$$

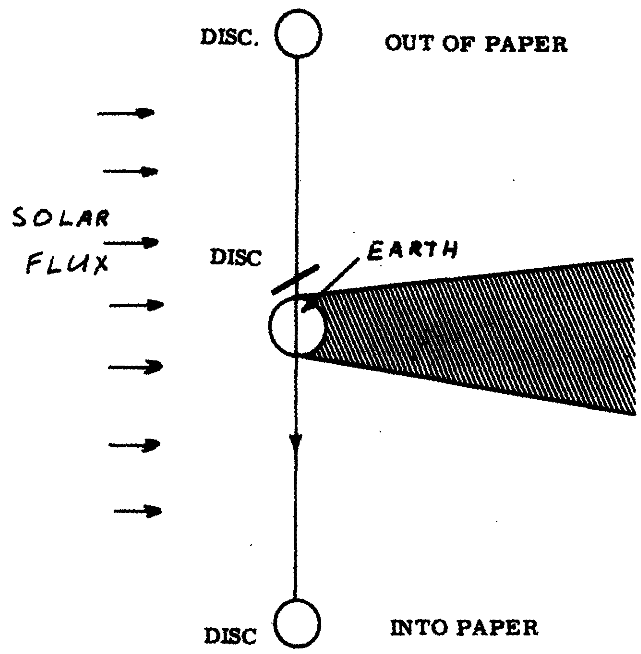
for the above $B_{tangential} = 3/8\ B_{(terrestrial)}$ which from Eq. (112) is seen to have a maximum value of $.06\ B_{solar}$. From the above we see without computing the numerical value that an intermediate compound angle which would use both earth and solar bounce is not advantageous for a polar orbit, and that, (assuming a flat sail), an optimum polar orbit is the one with a $B_{terrestrial} = 0$ and with solar angle of incidence of 35^o.

For an equatorial orbit as shown in Fig. 16, earth bounce will no longer have value zero. As shown in Fig. 16, (modified slightly from Cotter, Ref. 78), we see that tangential solar bounce has a four position average of:

$$1,\ 3/8,\ 3/8,\ 0 = .44\ B_o \qquad (38)$$

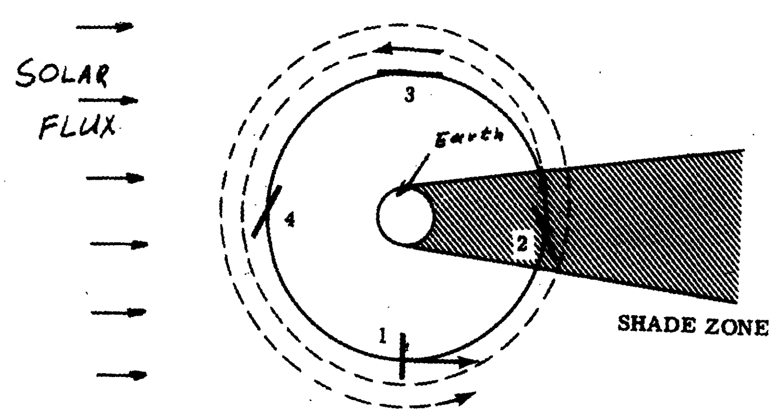
for the case of no earth shadow. For the case of earth shadow,

FIGURE 15



A POLAR ORBIT FOR AN ESCAPE SPIRAL FROM EARTH
(From Fig 26 ref 19 by T. Cotter)

FIGURE 16



A HIGH LATITUDE EARTH ESCAPE SPIRAL
(From Fig 29 ref 18 by T. Cotter)

$B_{tangential}$ has an everage of $.34\ B_o$. In addition, point 2 has a positive tangential contribution of $3/8\ Bo_{terrestial}$, point 4 has a negative tangential contribution of $3/8\ Bo_{terrestrial}$ where $Bo_{terrestrial}$ for point 4 is approximately $.56\ Bo_{solar}$ and $Bo_{terrestrial}$ for point 2 is only $.16\ Bo_{solar}$ as per appendix I, Equ. 112. Points 1, 2, 3, 4 all have a positive radial $Bo_{terrestrial}$ which, as previously stated, we neglect in a first approximation.

A calculation by Cotter of which the results only are quoted by him, (Ref. 98), gives earth escape in months from a 1000 N mile orbit as approximately 1/L where L is the lightness number. Since we considered B average to be between: $.34\ B_o$ and $.44\ B_o$ our approximate calculation gives us a result slightly higher than Cotter's, which given our crude approximation of a 4 point average and our neglect of radial acceleration is not surprising.

The polar orbit versus the equatorial orbit requires further analysis to establish which is more practical. The equatorial orbit would have a lower average B_o if too large a fraction of the orbit was spent in the shadow of the earth, but as pointed out by Cotter, this effect can be minimized by orbiting in the ecliptic around a relatively high latitude, this would also reduce the negative tangential contribution of $Bo_{terrestrial}$. In addition, a slight tilting of the orbit might be found to be justified because of the resultant further reduction in earth shadow time, and $Bo_{terrestrial}$ effect. As mentioned earlier, an analysis of the ellipse might provide a limited improvement. However, for a non-polar orbit the circularizing effect of greater Bounce at the apogee than at the shadowed perigee also must be taken into account.

A point not so far mentioned, but probably noticed by the reader, is that because of the earth rotation around the sun an 80 to 90-day escape would require a polar orbit to change its plane by almost 90^o. To a limited extent, use of a slightly asymmetrical sail angle would change the plane of the escape spiral. However, a calculation made in Ref. 80, Eq. (29) shows that given a 10^{-4} g acceleration, where g is the acceleration of the central force field involved, a 5^o change in plane of the orbit requires 200 revolutions. Since we would only have a fraction of a_r available for this purpose, and a_r is as we calculate equal to about $3/8 \sqrt{3} \times 10^{-4} g_o$, obviously the forces available can only cause a negligible small (for our purpose) change of orbit. From this we conclude that for reasonable lightness numbers (0.5 or less) the polar orbit would degenerate ultimately to an orbit almost normal to the starting one, at which point the escape orientation pattern would be the same as for the equatorial orbit shown in Fig. 16. In addition, of course, launch into the proper polar orbit requires both more energy and more accuracy.

An approximate polar launch might be an orbit whose normal is 30 - 45^o away from the solar radius vector so that after launch the 80 - 90^o orbital change would still provide an orbit whose normal is 45^o or less on the other side of the new solar radius vector, while maintaining the orbit normal to the ecliptic. As for the final velocity at escape, as shown in Fig. 14, it is small as compared to the earth orbital speed and, therefore, is not as serious a disadvantage of the polar launch orbit as would otherwise be the case.

With the difficult orientation problems involved and long escape times, one should seriously consider the launching of a solar sail vehicle into a hyperbolic path with respect to earth (escape) by use of a larger booster and using the solar sail only in space where it is more effective. However, such a step involves considerable additional energy over the 1000-mile orbit. For escape from earth we require: 673×10^6 ft/lb/slug (disregarding coriolis, drag, and other "non-ideal" effects) for launch to a 1000 mile circular orbit we require only 413 x 16 ft lb/unit mass. We, therefore, need over 50% more energy than is the case if we use a solar sail for earth escape.[1]

Alternatively, Dr. Tsu has suggested a solar heating rocket booster unit for escape from satellite orbit which would be sufficient to bring the payload plus sail to the same acceleration as the sail alone.[2] This booster which might then be dropped off would improve escape time, (assuming payload mass = sail mass)

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1 Based on $E = \frac{GM}{2r} - \frac{GM}{r_o}$ where $\frac{GM}{r_o}$ is gravitational potential energy on the surface of the earth = -673×10^6 ft lb/slug and E at altitude h = $\frac{GM}{r_o + h}$

Ref. Table 1.1, Eqs. 2, 27, 2.45, P. E. Sandorff, "Orbital Vehicle Notes," Chapter 2, March 1959.

2 Private communication, Dr. T. C. Tsu

by a factor slightly greater than two.

4.2 Interplanetary Transfer

Once the solar sail propelled vehicle has escaped we have seen in Fig. 14 that its specific orbital velocity is approximately the same as earth's. For an interplanetary mission we must either add or subtract specific energy from the orbit (for missions to inner and outer planets, respectively). For a planetary probe we are also interested in the velocity matching problem. The two pertinent parameters are then specific energy E and specific angular momentum H, both referenced to the sun.

First let us quote a few pertinent results from the application of celestial mechanics. In Fig. 17, we have reproduced an E-H diagram of the solar system from Ref. 75, by Edward Rodriquez of Autonetics. One of the mose important tools to understanding the behavior of a space vehicle in a central force field is the E-H plane analysis.

Rodriquez derives, the following key relations, (with slightly different nomenclature).[1]

Given instantaneous specific energy:

$$E = \frac{V^2}{2} - \frac{\mu}{r} \qquad (39)$$

$$H = r^2\dot{\theta} \qquad (40)$$

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[1] Ref. 79, p. 6-8.

Figure 17

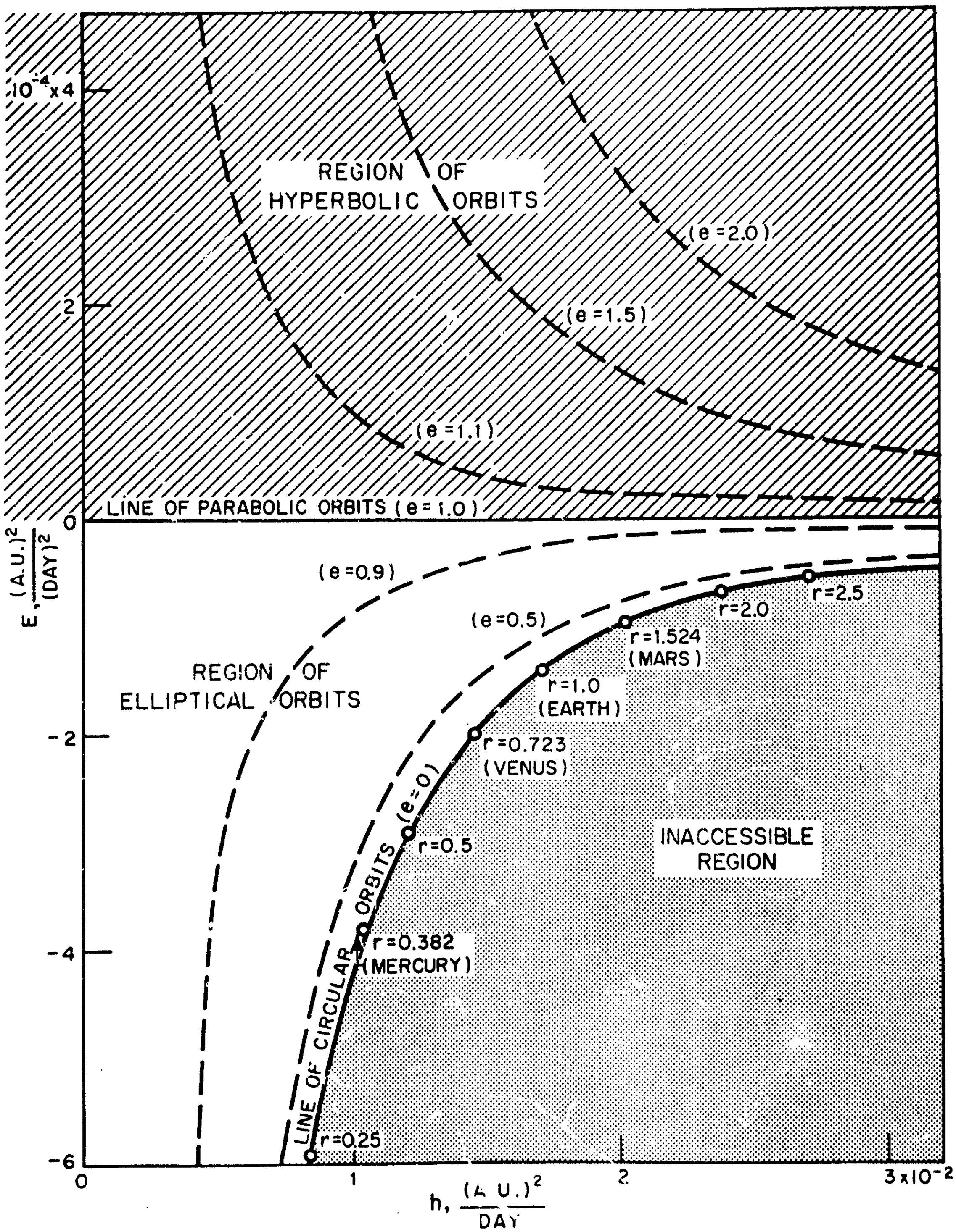

Fig 17. Specific energy vs specific angular momentum for two-body orbits

(Reproduced from ref 75 by Rodriguez)

We obtain:

$$\dot{E} = a\,V\ \cos \psi \qquad (41)$$

$$\dot{H} = ar\ \cos(\psi + \gamma) \qquad (42)$$

$$\frac{\partial E}{\partial H} = \frac{V \cos}{r \cos}(\psi + \gamma) \qquad (43)$$

where:

a is the acceleration vector

a_r is radial acceleration

a_θ is tangential to the orbit

V is the instantaneous velocity vector

r is the radial distance to the sun

ψ is the instantaneous angle between velocity and acceleration vectors

γ is the angle between the velocity vector and the local tangent to an orbit

E is the specific energy

H is the specific angular momentum

θ is the central transfer angle, (solar)

$\frac{\mu}{r}$ is the gravitational potential

From the preceding equations we see that a_r ($\psi\gamma = 90^\circ$) produces $\dot{E} = 0$ and $\dot{H} = -ar$ for $\psi = 90^\circ$. In general an acceleration tangential to the velocity vector ($\psi = 0$) produces:

$$\dot{E} = aV \quad \dot{H} = ar\cos\gamma \qquad (44)$$

Similarly for $\psi + \gamma = \pm 90^\circ$ $\dot{H} = 0$ $\dot{E} = av\cos\psi$ (45)

For low thrust vehicles since γ initially is ≈ 0 we have that initially radial thrust produces $\dot{E} = 0$ and $\dot{H} = ar$. By controlling the sign of the tangential component as well as the ratio of the two components of thrust we can control our E and H (since the radial acceleration can only be positive for a solar sail, we can only change the sign of a_θ). Applying this approach Cotter finds that to match Mars orbital velocity the initial portion of the trip must be with a_θ = negative so as to produce a negative $\dot{H}$ and thereby avoid excessive velocity at Mars. (Ref. 98, Fig. 3.) Cotter's calculations show a feasible Mars trip (including velocity matching) for a lightness number of L = 0.5 of 10 months with a 6-month wait in the vicinity of Mars (spent orbiting) and 10 months return, (ibid, p. 10). The same velocity matching objective as shown in Reference 69 or 86, Fig. 17 can be achieved more efficiently for an ion engine by using a negative radial acceleration during a portion of the trip.

In general one concludes that the effect on E and H of radial and tangential component are a function of past history, (as manifest in the instantaneous velocity vector), and that both time and energy optimization , therefore, is a highly complex problem.

One very obvious method of getting to Mars is, of course, a purely radial transfer, this has been examined in Ref. 66, but both in terms of velocity matching, flexibility, and time does not seem to be of any advantage. Such a transfer mode cannot be used for a return to earth . Radial acceleration towards the sun by use of radial thrust is of course impossible for a solar sail.

A general description of the effect of A_r and A_θ in the r, θ plane is more intuitive than the E, H plane, but the E, H plane as has been shown is very useful. For an analysis of Mars trajectory possibilities in the r, θ plane see the above reference by Irving, (Ref. 86).

For a solar sail, in particular, as was pointed out by Dr. Tsu at Westinghouse, an attempt to optimize travel time without insisting on exact velocity matching, (although a close approach to it does in fact result), leads to a very simple and interesting result. This result is that, (assuming initial velocity and position that of the earth (which is approximately true as seen in Fig. 14) and further assuming coplanar, circular orbits (also a very useful first approximation) minimum transfer time is achieved by use of a "Logarithmic Spiral". The logarithmic spiral involves a constant angle of incidence. Further this angle, Tsu proves, is purely a function of what is here termed the lightness number L.

From this simple solution he is then able to answer a number of pertinent questions: transfer time, angle of incidence α, and the "spiral angle φ" which is, as before, the angle between the velocity vector and the local tangent to the circular orbit.

In his solution a further approximation is made. Namely that the effect of radial acceleration is negligible. The effect of a_r is later compared to that of a_θ and it is shown that for the accelerations in question the error is small. We recall that $a_\theta = B_o \cos^2 \alpha \sin \alpha$ and $a_r = B_o \cos^3 \alpha$.

Results of Dr. Tsu's calculations are here reproduced as Figures 18-20. These show sail setting α, spiral angle ψ, and trip time to various planets.[1]

An interesting possibility for return to earth from Mars without a waiting time was reported by Battin at the MIT Instrumentation Laboratory, (Ref. 62). He showed that for a ballistic vehicle a close encounter of Mars of the order of a few thousand miles could be used to modify the return journey so as to make feasible non-stop round trips for most transfers. (Where it proved impossible, this manifested itself as a requirement for grazing with radial distance such as to be inside the surface of Mars.) The approach derived by Battin is to treat grazing encounters as impulses taking place at the minimum miss distance. Under these conditions the change in the magnitude of the velocity vector is small, but the angle change, ($\Delta\gamma$), may be large.[2]

Battin's work includes a 3-dimensional perturbation analysis and he reports on a whole family of ballistic encounters. Of special interest is his comparison of similar transfer orbits treated first as circular, coplanar transfers and then as non-coplanar, non-circular. More of his work including computer runs is included in Ref. 135, the MIT Instrumentation Laboratory Mars Probe Study. Another interesting study showing all possible

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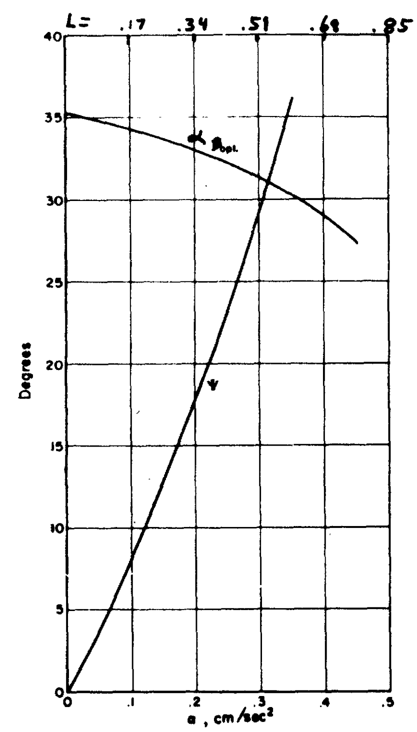
1 For a different derivation leading to the same transfer orbit conclusions in slightly different form, see Ref. 61.

2 For another interesting application of grazing encounter where Ehricke shows that a vehicle may gain a ΔV of 547 ft/sec on a close passage of the moon, see Ref. 100, p. 22.

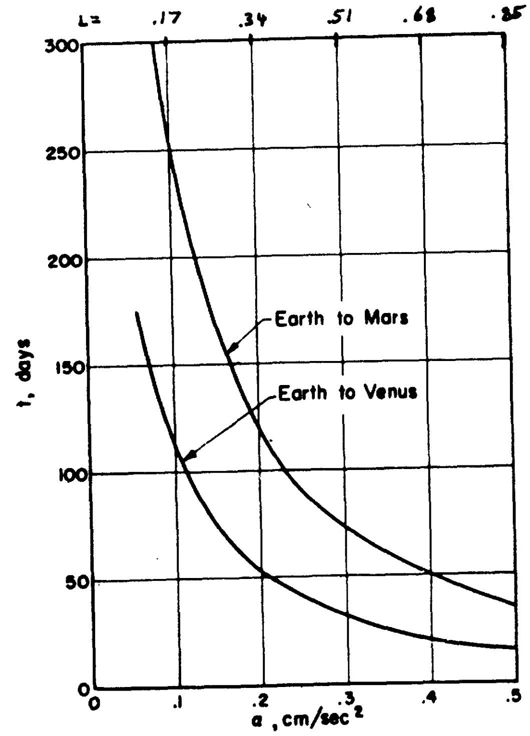
FIGURE 18



Optimum sail setting and spiral angle of ship's path

(From Fig 3 ref 106 by T.C. Tsu)

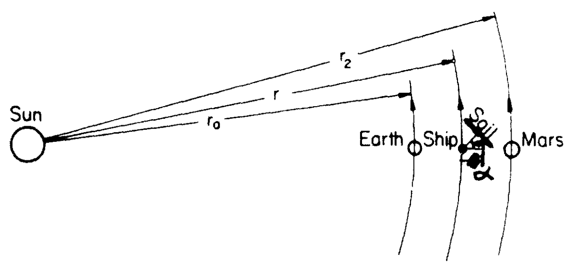
FIGURE 19



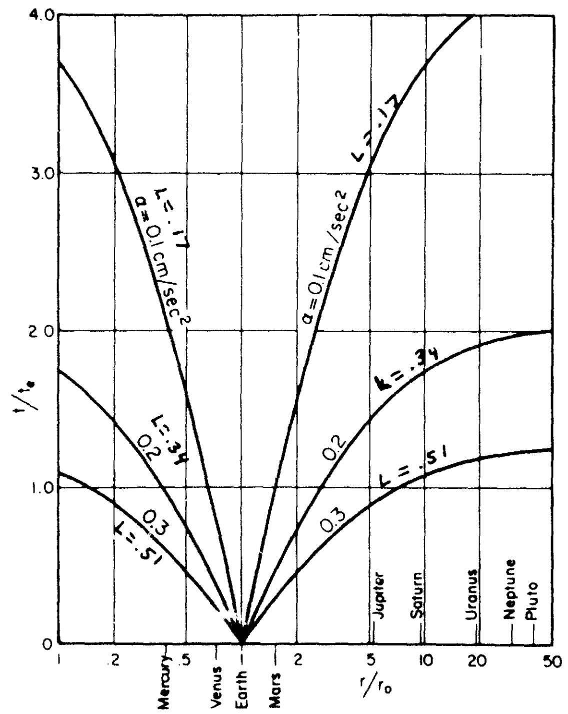
Trip time with optimum sail setting

(From Fig 6 ref 106 by T.C. Tsu)

FIGURE 20



Sail setting for Earth to Mars trip



Trip time along logarithmic spiral (t) compared with that along transfer ellipse (t_e)

(From Figs 8,9 ref 106 by T.C. Tsu)

transfers in the form of a topological mapping of date vs. transfer time with equi-energy lines serving as contour lines is found in Ref. 64, and is reproduced herein as Fig. 27.

4.3 Relativistic Effects

Quite obviously solar sailing in the solar system does not involve velocities where relativistic mass corrections are directly important, but what of any pertinent relativistic corrections on the behavior of the impinging photon and its momentum transfer. Ehricke in Ref. 100 treats the relativistic force correction due to the Poynting-Robertson effect. This effect, a relativistic force, is based on a doppler shift such that "radiation emitted in the direction of azimuthal motion has a slightly higher frequency than the radiation emitted in the opposite direction". Further, since radiation energy is a function of frequency we have an energy change. A similar radial effect exists. For a light 100 ft diameter sphere Ehricke finds, (p. 62), an azimuthal deceleration of 2.3×10^{-12} g_o due to the above effect. Thus relativistic corrections are of no practical importance.

PART II - The "Self' Stabilized Stabilizer" for Satellite Attitude Control

5.0 Mode of Operation

In Section 2.6 we discussed the various possible methods whereby a solar sail or vane can be used for attitude control, angular momentum "dumping" or oscillation damping. The purpose of Part II is to describe a novel approach to the above function that the writer believes provides a more efficient method of performing any or all of the above named functions.[1] For the purpose of discussion we shall consider as a useful application a 2000 lb earth satellite of cylindrical configuration and 10 ft in length. Such proportions might be those of an orbiting telescope, not including its auxiliary functions.

The Self-Stabilized Stabilizer consists basically of a very small Class 2, or rigidized sail, with provision for inherent passive stability to obviate the need for active attitude control of the attitude control device itself. In using a Class 2 sail we reduce our sail lightness from $L_o = .32$ to about between .032. This corresponds to a sail whose wt/sq ft is instead of .001 lbs about .01. The above sail typically of 10 ft diameter is to be attached to the payload

---

[1] Although the Self-Stabilized Stabilizer here presented is believed to be a new idea the writer wishes to acknowledge the influence on his thinking of a paper on Aerodynamic Stabilization by John K. Wall of Douglas, (Ref. 9). The viscous damping technique proposed in this paper is based on a similar device of Wall's.

through a lightweight rod of sufficient stiffness to prevent collapse under the forces involved. Clearly, as will be seen, an optimum ratio exists between sail size and the above moment arm. However, this is a function of design and material selection and optimization has not been attempted here.

All previous proposed stabilizers follow the expression derived in Appendix IV. They have a restoring force:

$$A \ell \cos^2 \alpha \tag{46}$$

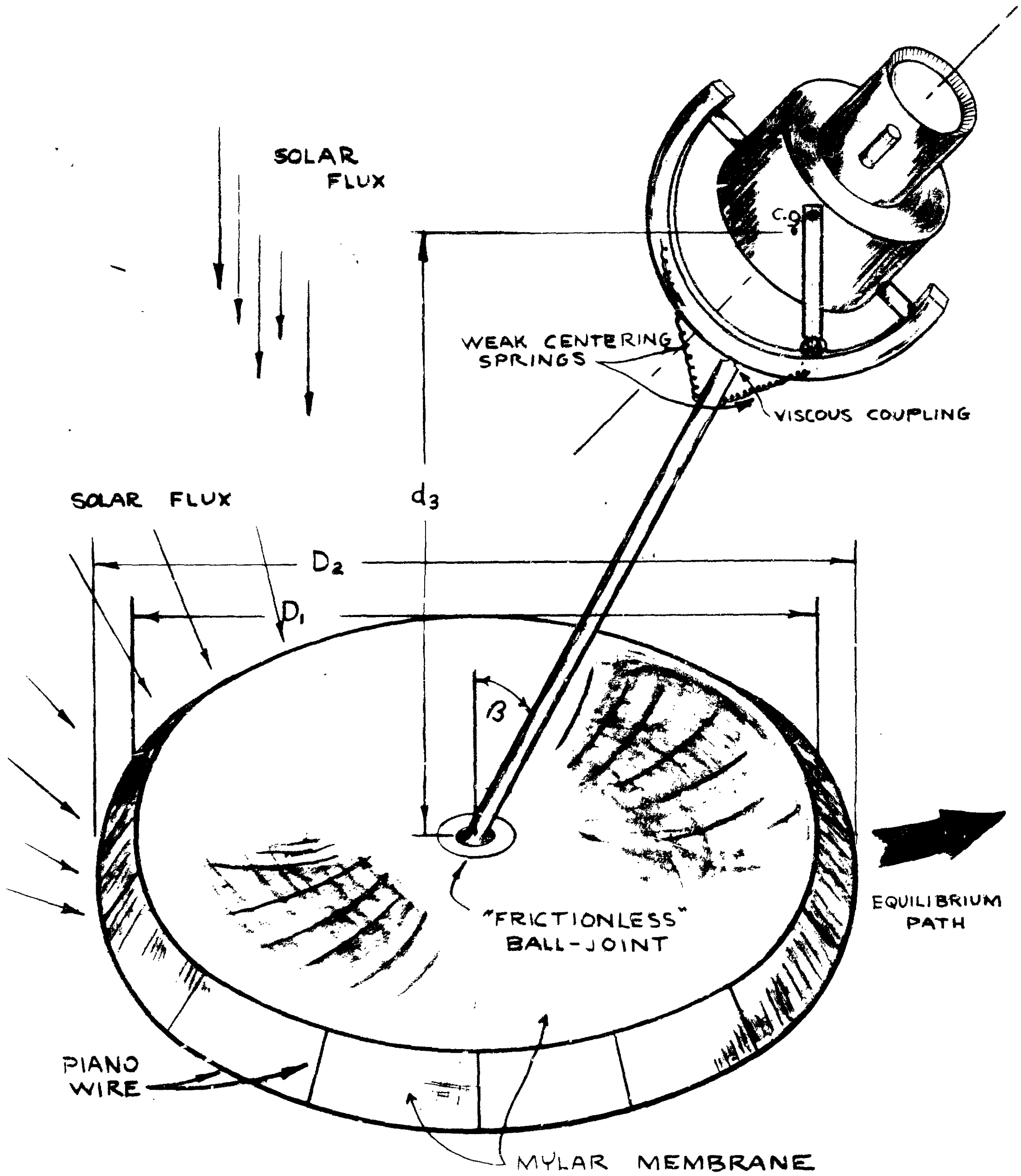
where:

> α is the angle of incidence
>
> A is sail area
>
> ℓ is moment arm between c.g. of vehicle and c.b. (center of bounce) of sail

Momentum transfer devices, (which are essentially the above stabilizer rotated 90°), such as the MIT solar vane, (see Section 2.6) have a torque of $A \ell \cos^2 \alpha$ where $\alpha \approx 0$. However, the solar vane has no steady state stabilizing effect, for if it is used the vane will exert a torque tending to cause the vehicle to rotate until it reaches the condition of the first mentioned stabilizer, unless a second and equal solar vane is placed on the opposite side of the vehicle in which case clearly restoring torque is zero for all α.

The Self-Stabilized Stabilizer shown in Figure 21 has a restoring force of Ad sin ß for all ß.

FIGURE 21
THE SELF-STABILIZED STABILIZER



where:

β = angle between rod and normal to sail

A = area of sail

d = distance between end of connecting rod and c.g.

We further note that for equal weight stabilizer area "A" can be much larger without collapse than for other designs since its behavior in bending is in general that of a uniformly loaded plate with a central support, or for the design of Fig. 21, the case of a hoop with a membrane supported at its center, Other stabilizers are either cantilevers or membranes in a cantilevered frame. For maximum angular momentum transfer the stabilizer can be held, (by a flywheel system in the payload) at $\beta = 90^o$.

For damping action we propose two alternatives depending on damping requirements. The simplest method is to coat the rod with a rubber suitable to the environment, (possible a silicone derivative). As is well known, rubber is a very effective damping media and flexural oscillations of the rod would provide a small but for some systems adequate damping. Here again an optimization study should be made. An optimum damper is a function of available stabilizing torque (which limits useful stiffness) and a balance between the damping and spring constants. We leave such an investigation to a later study.

A more efficient but also more complex and heavier approach to damping is to use a viscous coupling between rod and vehicle with a weak centering spring, somewhat similar to Wall's device mentioned earlier. Such a device should not be used between sail and rod. The bearing at that point should on the contrary reduce friction to a minimum to prevent variations in stabilizer orientation.

As for the stabilizer itself as seen on the figure we propose a 1/4 mil Mylar disk perpherally stiffened by a spring wire hoop and with a reinforced center section where the bearing connection with the rod is made. The rod itself for optimum weight should be built in several concentric sections of decreasing diameter so as to closely approximate a tapered rod of uniform stiffness. For a sufficiently long rod the base (vehicle) sections would be hollow tubing.

6.0 Calculated Characteristics of Stabilizer

Given:

$B_o = 2 \times 10^{-7}$ psf (at 1 A. U.)

$d_1 = 10$ ft

$d_2 = 11$ ft (per Fig. 21)

$d_3 = 15$ ft

1. Force due to solar bounce

$$F = B_o \left[\pi \left(\frac{d_1}{2}\right)^2 + \cos^3 \beta \, \pi \left(\frac{d_1+d_2}{2}\right)\left(\frac{d_2-d_1}{2}\right)\right] \quad (47)$$

$$F = 2 \times 10^{-7} \pi \left[\left(\frac{10}{2}\right)^2 + .55 \left(\frac{10+11}{2}\right)\left(\frac{11-10}{2}\right)\right] \quad (48)$$

$$F = 6.3 \times 10^{-7} [25 + 2.9] = 1.7 \times 10^{-5} \text{ lbs} \quad (49)$$

2. Maximum stabilizing torque

$$1.76 \times 10^{-5} \text{ lbs} \times 15 \text{ ft} = 2.64 \times 10^{-4} \text{ ft lbs} \quad (50)$$

$$= 3580 \text{ dyne cm} \quad (51)$$

3. Restoring torque for any angle ß

$T_r \approx 2.64 \times 10^{-4} \sin \beta$ ft lbs (52)

$T_r \approx 3580 \sin \beta$ dyne cm (53)

4. Self-stabilizing torque (available to preserve stabilizer Orientation

$$T_{restoring} \approx 2 B_o [\cos^3 \beta - \cos^3 (\beta - d\alpha)] \, 2\pi (r_2^{\,3} - r_1^{\,2}) \quad (54)$$

for $d\alpha$ = small change in angle of incidence

$$T_r = \frac{4\pi}{3} \left(\frac{1320-1000}{8}\right) [B_o \cos^3 \beta - \cos^3 (\beta - \delta\alpha)] \quad (55)$$

$$= 400 \times 10^{-7} [\cos^3 \beta - \cos^3 (\beta - \delta\alpha)] \text{ ft lb}$$

for special case of $\delta\alpha = 2^{\circ}$

$= 13 \times 10^{-7}$ ft lbs (56)

$= 17$ dyne cm (57)

Weight of Self-stabilized stabilizer

Wt. of rod 0.5 lbs (assuming rod ten feet long at .05 lb/ft (58)

Weight of stabilizer

(aluminum coated 1/4 mil Mylar) plus frame

1. Wt. Mylar $\pi r_2^2 = \pi \left(\frac{11}{2}\right)^2 (.002)^{\text{lb/sq ft}}$ (59)

$= .19$ lbs (60)

2. Frame (2 wire hoops)

$\epsilon [\pi d_1 + \pi d_2] = \epsilon \pi (10 + 11) = 66$ ft. (61)

where ϵ = wt/ft

assume 16 gauge wire 0625" dia. then $\epsilon = .0105$ lb/ft (62)

Frame wt. = .66 lbs (63)

Total weight (not including damping mechanism) = 1.3 lbs (64)

As seen in Figure 21, a second and slightly larger hoop concentric with the first one provides stabilization to the stabilizer by offering a restoring torque for any $\delta\alpha$ perturbation, (such as bearing friction). In this manner the stabilizer is kept normal to the sun. Stabilizer equilibrium clearly occurs at $\alpha = 0$. The optimum angle ß for maximum restoring torque of the stabilizing section is easily computed by finding the minimum of the restoring torque expression for small displacement on unit areas at opposite ends.

For small α:

$$T = \ell \cos^3 (\alpha - ß) - \ell \cos^3 (\alpha + ß) = 0 \qquad (65)$$

$$\frac{\partial T}{\partial \alpha} = 3\ell \, (-\cos^2 (\alpha - ß) \sin (\alpha - ß) + \cos(\alpha+ß)^2 \sin(\alpha-ß) = 0 \qquad (66)$$

specializing to $\alpha = 0$

$$\frac{\partial T}{\partial \alpha} = 3\ell \, [(\cos^2 ß \sin ß) + \cos^2 ß \sin ß] = 0 \qquad (67)$$

$$\frac{\partial\left(\frac{\partial T}{\partial \alpha}\right)}{\partial ß} = \frac{\partial^2 T}{\partial ß \, \partial \alpha} = -2 \cos ß \sin^2 ß + \cos^3 ß = 0 \qquad (68)$$

adding $\sin^2_ß$ to both sides: $3 \sin^2 ß = \cos^2 ß + \sin^2 ß$ (69)

$$\sin^2 ß = \frac{1}{3} \qquad (70)$$

$$ß = 35^\circ \qquad (71)$$

It is this feature which is the basis for the suggested name of the whole stabilizer, the "Self-Stabilized Stabilizer".

Part III The Centrifugal Sail, Its Evaluation, And Its Application To A Mars Probe -- An Integrated Vehicle Concept.

In Part I we considered both general pertinent information and specific proposals pertinent to the use of solar sails. We also tried to classify these proposals in a significant manner, with appropriate terminology. In this part we shall focus our attention on a particular mission and develop a solar sail vehicle concept which, the writer feels, can be seriously considered for this purpose. Part III will include a brief description of design alternatives and a brief explanation as to why a particular selection is made.

7.0 THE "CENTRIFUGAL SAIL" AS A DESIRABLE CONFIGURATION

By "Centrigugal Sail" we mean a class 3 or "Field Effect Sail" which derives its pseudo-stiffness from the inertial forces induced by spinning the circular sail about its own axis. In appendix V, we derive the appropriate equations for determining a suitable RPM with acceptable stresses and deflections. It is shown in that appendix that at suitable velocitities corresponding to less than 0.4 RPM, deflections from a perfectly flat disk of the order of 1 to 2 ft. and maximum centrifugal stresses less than.5 psi result for our proposed 250 ft. diameter sail with its 200 lb. payload, (as more fully described in section 11.).

7.1 Centrifugal Sail Design Aspects

The choice of a centrifugal design is based on its high lightness number ($L_0 = 0.32$ for our proposed design), its simplicity of construction, its freedom from serious micrometeoric damage, (as indicated in section 1.6), its ease of erection in space and the special advantages derived from its gyroscopic behavior.[1] The benefits of gyroscopic properties are the well known ones of "spin stabilization" which minimizes the effects of disturbance torques, and controlled precession which, by preventing overshoot, greatly simplifies the dynamics of the application of control torques to cause attitude change. This is especially true in space where, in the absence of gyroscopic action, any impulse will cause a certain angular velocity which will persist until an equal and opposite impulse or corresponding integrated torque is provided. The value of eliminating the need for a constant control

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[1] The idea of spinning a sail is not presented as a new idea. The writer first thought of the idea after studying Hoffman's excellent paper on structural design for space vehicles, (ref. 117), and has since read Cotter's paper, (ref. 98), which uses a centrifugally supported solar sail as part of one of his design alternatives.

"dither" to maintain the orientation vector, (and thereby for a solar sail the acceleration vector) within prescribed narrow limits cannot be over emphasized.

The use of aluminium vapor coating is based on its satisfactory reflectivity which is over 98% in the infrared region of interests, (ref. 155), and 90% in the visible (ref. 109). Its low transmission of ultraviolet radiation to the mylar backing, (ref. 109), and its A/e ratio which leads to satisfactory thermal equilibrium temperatures (ref. 122), are among desirable characteristics.

Of possible alternative coatings silver (suggested by Ehricke) is of special interest. Its A/e of 2.0 as compared to 12.0 for aluminium would lead to a substantially lower equilibrium temperature, (see Figs. 2, 3, 4). This in turn would increase the stiffness of the mylar coated surface and improve self damping of the sail. Silver has the further advantage of a higher reflectivity, 96% in the visible and 99% in the infrared.[1] The coating removed by proton sputtering according to Whipple is proportional to both atomic weight and an efficiency factor. Thus the mass of the coating would be a constant if sputtering removal efficiency turns out to be the same. No experimental data on this point has been found.

A detailed solar sail design study should determine optimum temperature and consider a mixed aluminium, silver

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[1] Ref. 155 & 100, p. 58

coating in a ratio suitable for providing the desirable temperature range. Use of mixed coatings for this purpose has already been made on existing satellites.[1] Another property of films which requires further investigation is conductivity and its effects. During an earth escape manoeuver interaction of the spinning sail with the earth's magnetic field might cause eddy current damping. Such a consequence may therefore have to be guarded against in any design involving earth escape.

The choice of the polyester film Mylar is based on its availability, wide service temperature, high tensile strength, (primarily valuable for handling on earth), negligible outgassing, (discussed in section 2.4), and excellent flexural tolerance, plus current availability in $\frac{1}{4}$ mil. thickness and potential availability in 1/10 mil.[2] No thinner or equally thin film is known to be available. Mylar has three known disadvantages: poor ultraviolet resistance, an adhesive is required for bonding, and relatively high specific gravity, (1.38). The above it will be recalled are elaborated on and alternative films are discussed in section 2.4. It will be recalled that the interesting alternative of sublimating the plastic film in space and allowing the 3000A$^{\circ}$ coating, (with a grid type support), to carry the load was also

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[1]Ref. 29, p. 2

[2]Private communication, Dr. T. C. Tsu

discussed in section 2.4. The potential improvement in the sail lightness number (even assuming a 20% sail weight allowance for a supporting grid), from 0.32 to 1.5 is very attractive. With payload of mass equal to that of the sail this would give a vehicle L of 0.75 and would allow earth escape in about 40 days. This, as seen in section 4.1, would radically change a presently somewhat marginal situation. Because such a device requires a substantial development effort, however, we have not made it the basis of our proposed "first generation" solar sail. We note that such a device by being a practical Earth escape vehicle would substantially improve the launch mass ratio.

7.2 Dynamic Behavior of a Centrifugal Sail

The reader may have already asked himself what is the dynamic response of a sail to attitude changes, will it precess like a rigid body and because of its low stiffness and damping properties what types of vibratory modes may be expected, what amplitude might be found both for steady state and transient conditions? The above questions turn out to be extremely difficult ones to answer in a satisfactory manner.

Because of their importance, considerable effort was expended on them and, after a number of false starts, three fundamentally different approaches yielded useful information:

Experimental -- Experiments were performed at the MIT Dynamic Analysis Control Laboratory using spinning plastic and cloth disks of varying stiffness and weight. Response to various perturbations was observed using a strobotac and photographs were taken using an E.G.G. strobolume Model #1532A.[1] Results are shown in Figures 22, . Because of lack of funds, (i.e. none) it was not possible to arrange to run the experiment in a vacuum. Nonetheless some interesting observations were made.

Fortunately (after the above program was well under way) Mr. Emmannuel Schnitzer of NASA told the writer of a similar experiment conducted at NASA in a vacuum of 0.02 mm mercury. Through the courtesy of Mr. James G. Simmonds of NASA,Langley Field Operations, their findings were made available and will be discussed here.

Analytical - Simplified Model -- A simplified three domain model was used to predict the precessional behavior and follow through of the sail.[2]

Analytical Treatment - Vibration Patterns -- A careful mathematical analysis of the general case of a rotating disk of negligible stiffness is found in Applied Elasticity by John Prescott, (ref. 133). Pertinent results

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[1] The courtesy of Dr. Edgerton of Edgerton, Germeshausen & Grier Inc., in loaning this piece of equipment is gratefully acknowledged.

[2] The contribution of Dr. Robert Hufnagel of the Perkin-Elmer Corp. Norwalk, Conn. in suggesting the model is gratefully acknowledged.

of this very thorough mathematical analysis will be quoted and compared with experimental findings.

7.21 Experimental Information

The experiments performed by the writer at the MIT Dynamic Analysis Control Laboratory were designed to provide experimental information on behavior of thin spinning membranes. It was also intended to study modes of vibration as a function of sail stiffness, angular velocity, and sail diameter. Any transients due to small pertubations was to be observed. Finally, "follow through" of the sail during a rotation normal to the spin axis was also to be observed, and precession noted. Test mounts included a $\frac{1}{4}$ hp electric drill as shown in figure 22 with a 6" diameter hub. A Variac control was provided for speed variations.

A second mount consisting of a 1/20 hp motor on a 2 degree of freedom gyroscope mount was also used. Provision had been made for small unbalance torques to be provided to initiate sail precession. These did not turn out to be large enough to observe precession and torquing by hand was successfully substituted.

Conduct of Experiment

4 sails were employed consisting of:

1 - light cloth sail of 15" diameter.

2 - Two thin plastic disks of low stiffness with 30" dia. and 20" dia. respectively.

FIGURE 22

DYNAMIC RESPONSE OF A SPINNING SAIL

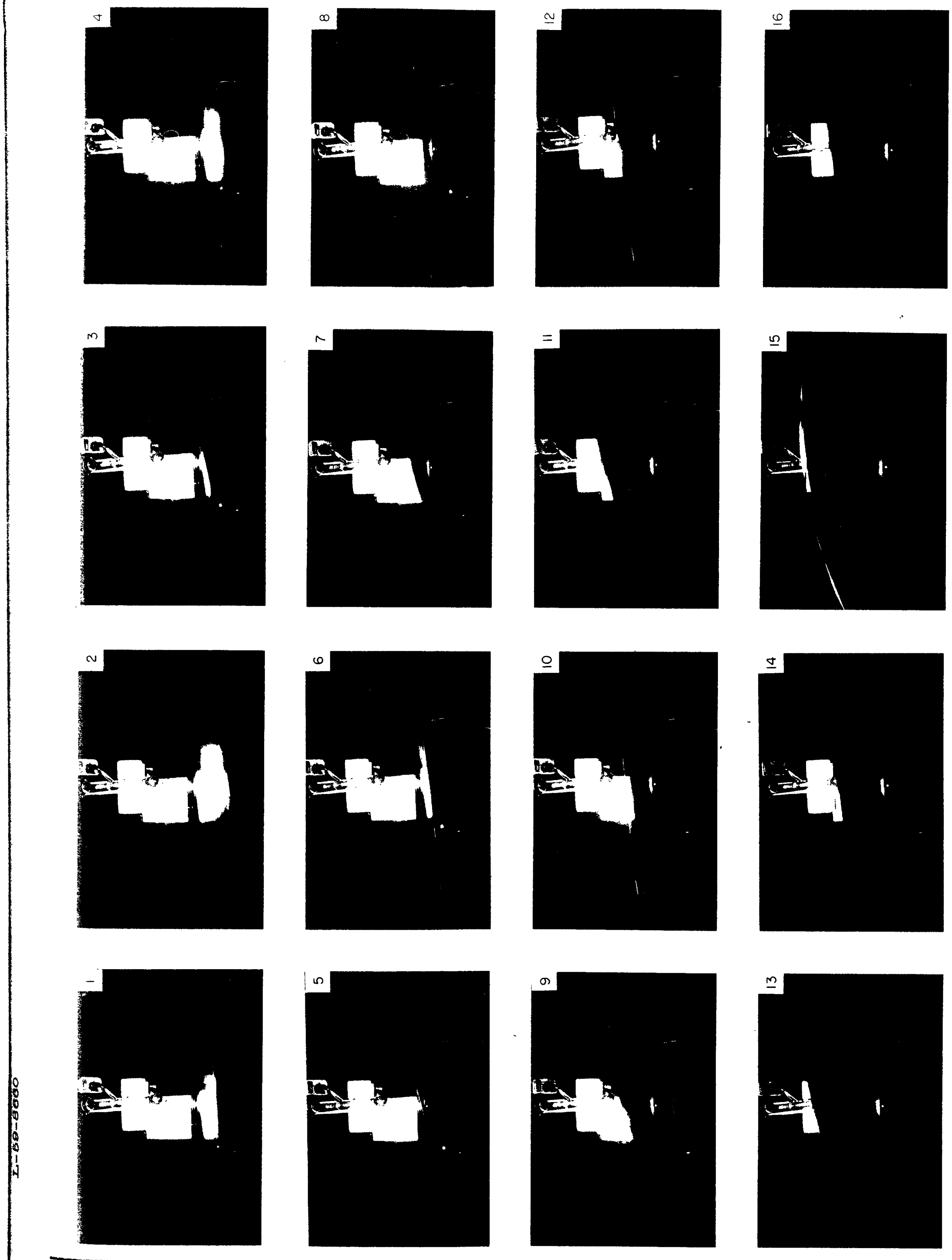
20" DIAMETER PLASTIC
2000 RPM
370 RPM NODE ROTATION

20" DIAMETER PLASTIC
680 RPM
115 RPM NODE ROTATION

20" Diameter Plastic
1340 RPM
200 RPM NODE ROTATION

15" DIAMETER CLOTH
2150 RPM
(NOTE NODE REVERSAL

FIGURE 23



FLUTTER IN A SPINNING MEMBRANE AT 0.02 mmHg PRESSURE
(courtesy of Mr. James Simmonds, NASA)

3 - One thin plastic disk of greater stiffness and weight with a diameter of 30". (Type and exact gage of plastic film not known.)

The disks were circular and homogeneous in all cases, a 6" hub supported them at the center and the sail was taped at the edge joint between the edge of the central hub and the sail at r = 3". Experiments on both mounts were conducted at speeds ranging from about 200 rpm through 2000 rpm. After preliminary observations were recorded the experiment was performed at night with a Strobotac which was used to observe the sail patterns and to determine both sail speed and speed of rotation of nodes. The Strobulume unit mentioned early with a 100 microsecond flash was used for photography. In all some 20 different combinations of speed and sail diameters were observed and the following was found:

1. In all cases and on both mounts, irrespective of method of deployment, only 3 nodes were observed. They were equally spaced radially as seen in figure 22. No other vibration patterns were observed. Deployment was done at varying accelerations and both with sail folded on the hub (centrifugal self deployment) and with sail edges supported by hand.

2. Cusps occured either on the bottom (more frequent) or on the top, the cusp once "set in" on a particular run would not reverse, but might for the same sail be

different on another run.

3. No suitable equipment was available to measure oscillation amplitude and it was therefore estimated. Amplitude was a function of speed, and increased non-linearly with angular velocity.

4. Oscillation amplitude was a function of diameter of sail, and increased with increasing diameter.

5. No combination of speed and diameter resulted in a standing wave. Node rotational velocity varied between 15 and 26% of sail angular velocity.

6. Ping pong balls were thrown at the sail at various speeds to simulate small pertubations. In no case was an observable transient produced.

7. The sail mount (electric motor) was rotated by hand about an axis normal to the spin axis to observe follow through. This was invariably smooth and monotonic. The experiment was done with both low and high angular accelerations. When a 20" sail was run on the gyroscopic mount at around 100 rpm it was possible to observe the beginnings of sail precession. This precession was of course prevented by constraints on the hub and therefore appeared only as a transient, but very perceptible mode. Precession torque was provided by rapid hand rotation in one axis of the mount normal to the spin axis.

Before discussing the results we shall consider the experiments run at NASA. Experiments at NASA, Langley Field were performed by Mr. James G. Simmonds[1]. They were performed in a 0.02 mm Hg vacuum. A thin circular film of 4 foot diameter was spun at between 500 and 2500 rpm. At a speed of 800 rpm a simultaneous rotation of 2.4 rpm about a second axis was also employed. The resultant oscillations on the latter test were recorded on film at 24 frames/second. Both distortions and precessions were observed. As described they appear to be similar to the ones obtained in our tests except of lower amplitude. The photographs taken by them, fig. 23 (in copies 1 to 4 of this thesis only) are not distinct enough to be readily interpreted by someone who did not actually see the experiment.

Through analysis of the aerodynamic forces, they conclude that the ratio of aerodynamic forces to centrifugal

$$\text{forces}^{1} = \frac{P_a\, b}{P\, h} \tag{72}$$

Where P_a = air density
b = radius of disk
P = density of film
h = film thickness.

They conclude that "even for low vacuums the ratio of $\frac{b}{h}$ may be so large as to make the aerodynamic forces significant."[2]

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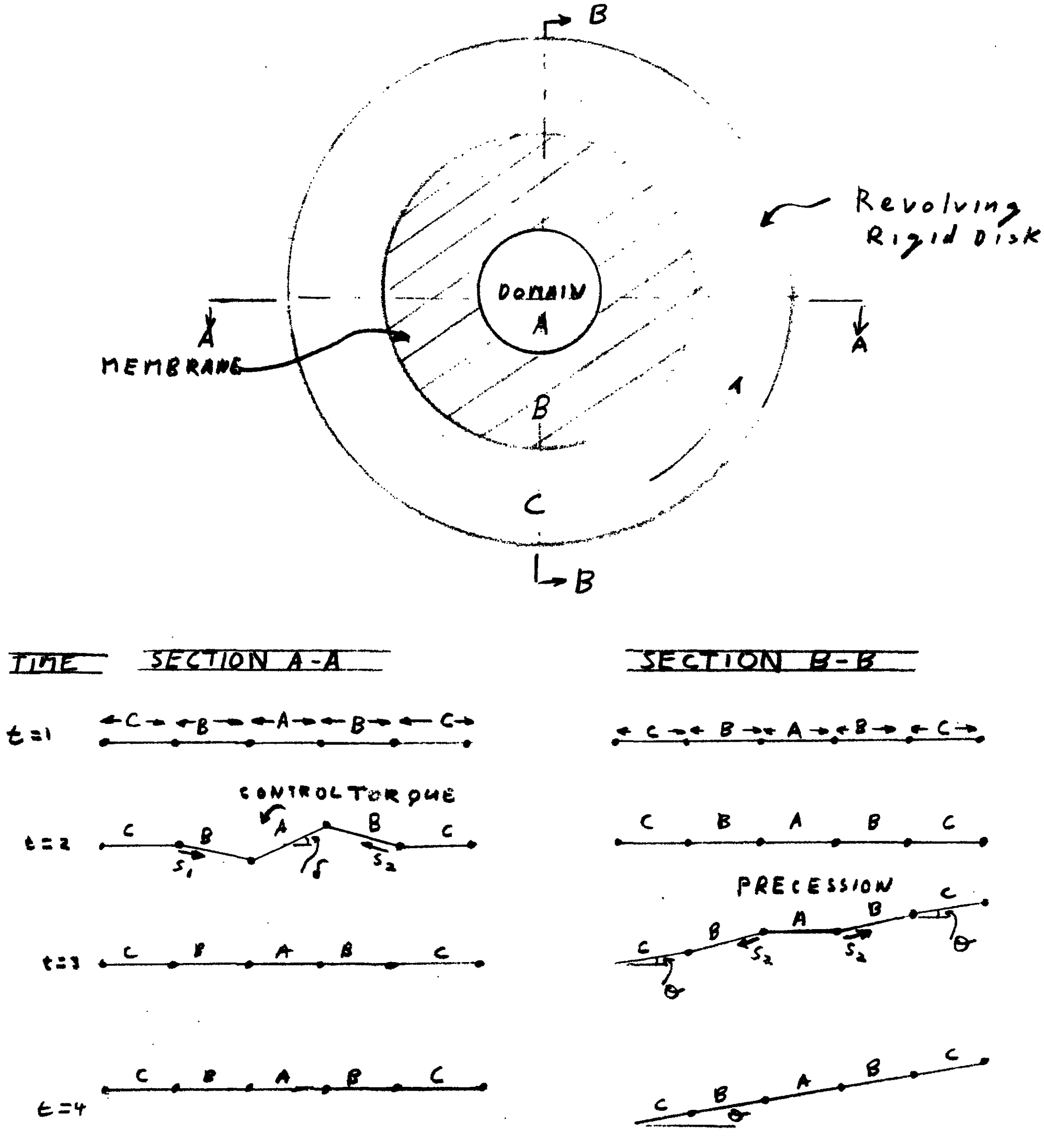
[1] Private communication, Mr. James G, Simmonds

[2] Mr. Simmonds is currently proparing a report titled "The Dynamics of spinning Membranes" which will explore the problem further.

FIGURE 24



AN ANALYTICAL MODEL FOR PRECESSION OF A SPINNING SAIL

7.23 Analytical - Simplified Model

As seen in fig. 24 in determining the method of initiating precession we can conveniently consider the sail to be composed of 3 domains.

Domain A - a rigid disk not rotating on which a torque T is impressed.

Domain B - a non-rotating membrane.

Domain C - a rotating rigid body (where centrifugal force gives it pseudo stiffness.)

These approximations then allow us to connect the domains by appropriate boundary conditions.

Namely: at A-B $M_b = 0$

at B-C $M_b = 0$

where M_b is the bending moment

Obviously the membrane can only carry tension. In the successive sections of figure 24 labeled $t = 1$, $t = 2$, $t = 3$ and $t = 4$ the precession mechanism, is described. At time $t = 2$ we see a control torque causing a δ displacement in the membrane and transmitting a stress S_1, to the outer disk. At time $t = 3$ we see the outer disk precessing and transmitting a counter torque S_2. S_2 causes the hub to precess. This simplified analysis by correctly identifying 3 areas where different forces predominate reduces our interaction problem to one of: torque in a rigid non-rotating body, force transmission in a membrane, and precession in a rigid rotor, each of which is well known.

7.23 Analytical Treatment

Ref. 133, chap XVIII by Prescott concerns itself with natural vibrations of rotating disks. The analysis of a rotating disk in the above reference considers stiffness and centrifugally induced tensions. The contribution of aerodynamics (or solar bounce) effects is of course not included.

We quote:

> "When a rigid disk in rotation vibrates it is controlled both by the tensions and the stiffness. Since the tensions in the disk are proportional to the square of the angular velocity it is clear that, at low speeds, the main controlling force is the rigidity, whereas, at very high speeds, the main controlling force may be mainly the tensions due to rotation. We shall first find the periods of vibration of a disk which has a negligible rigidity. Afterwards, when the periods of vibration due to rigidity alone have also been found, a method will be given for finding the period of the rotating disk in which both rigidity and tensions are taken into account."[1]

Of special interest to us is the case of a disk without stiffness.

For this case the differential equation for equilibrium reduces to (p568)

$$\frac{1}{r}\,\frac{\partial}{\partial r}\left(r\,P\,\frac{\partial w}{\partial r}\right)\quad \frac{Q}{r^2}\,\frac{\partial^2 w}{\partial \theta^2} = \rho\,\frac{\partial^2 w}{\partial t^2} \qquad (73)$$

where: W = displacement normal to the plane
P = radial tension
Q = circumferential tension
ρ = density

[1] Ref. 133, p. 565

Solutions of this equation are then derived. The equations satisfied by solutions consisting of one or more nodal circles and one or more nodal diameters are as per the following table (from pages 571-574).

N = (nodal diameter)

Number of Nodes									
N	0	2	1	2	4	15	56	n	n
S	1	1	0	0	1	5	20	0	1

Further, Prescott finds that the case for N = 0 can only be satisfied for a free disk since the center of the disk is not at rest.

Prescott solves for the rate of rotation of nodal diameters. He finds 3 possible angular velocities:

$$W - \frac{P_1}{n},\ W,\ W + \frac{P_1}{n} \qquad (74)$$

where P_1 is the frequency of vibration.

A case of special pertinence is that of the "uniform disk vibrating with one nodal diameter" (p. 610) for this is the case of a disk clamped at an inner radius, and in this respect corresponds to our sail. For this case (analysed by R. V. Southwell) "it is possible for the disk to vibrate with its edge free in a mode with one nodal diameter and no nodal circle."

We could easily consider this analysis in more detail, but the above is sufficient to lead us to conclude that:

1. The above analytical approach by not including the very complex aerodynamical effects fails to predict modes obtained experimentally (fig 22).

2. In any case according to Prescott the analytical

approximation which he uses to make the problem tractable do not apply to the relatively large amplitudes of interest. He says:

> "If the maximum deflexion in any vibration is less than one fifth of the thickness of the disk the theories of this chapter can be regarded as practically accurate; for a maximum deflexion equal to the thickness the frequency might be, according to the particular mode of vibration, 10 to 25 per cent greater than we have calculated. In fact, for large amplitudes, the motion cannot be resolved into normal modes of vibration; and probably there are no pure vibrations at all, but only an irregular wobbling in which amplitude and period both change considerably from one vibration to the next."[1]

3. Analytical solutions of Prescott's show a great many permissible modes and show the importance of the ratio of hub diameter to sail diameter, this can very probably be considered as generally valid and should be taken into account in experimental work.

4. Conspicuous by their absence in our experimental results are both alternate modes involving various diametrical nodes and any form of circular nodes.

We can accordingly conclude that, as for instance in the case of lift in fluid mechanics, purely analytical techniques for the problem of a spinning sail may require too many simplifying assumptions to allow useful predictions and that it may possibly prove more fruitful to resort to experiment coupled with dimensional analysis. If, however, the above conclusion should be disproven we are confident this will not be without much labor.

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[1] Ibid p. 619

7.24 Discussion of Results

Our experiments and those of NASA show the potential importance of flutter in a spinning sail. They also show the importance of aerodynamic effects, and how only a high vacuum will permit adequate simulation. The analytical work of Prescott's, which do not take into account aerodynamic effects, should better describe phenomena in a vacuum and (since they are only valid for a small amplitude) can serve as a useful starting point for the analytical evaluation of conditions of stability which can be done in conjunction with experimental work. We note that Solar Bounce, providing as it does a vector force of varying intensity normal to the local unit area, can act as a damping force or as an exciting force depending on assumed flutter geometry. Aerodynamic effects are several orders of magnitude smaller. As for self damping of the mylar no information is currently available.

Flutter in a spinning disk subject to dishing is inherent in the fact that a dished disk is a "non developable surface", i.e. cannot be developed from a two dimensional shape without distortion. Conceivably a solar sail whose equilibrium shape would be slightly dished could eliminate flutter since such a shape would not require distortion to take on a dished configuration. In this context we note with interest the experimental results at NASA with spinning of a "cone parabola combination which contrary to the disk behaved excellently".[1]

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[1] Private communication, Mr. James Simmonds

8.0 ORIENTATION CONTROL BY PAYLOAD SHIFT

As we have seen earlier a solar sail controls its velocity vector by shifting its orientation. $\cos^3\alpha$ and $\cos^2\alpha \sin\alpha$ are respectively radial, (a_r) and tangential, (a_θ) accelerations. In addition a component of velocity normal to the ecliptic can be obtained by orienting the sail at some angle β. To change orientation several methods are available as was shown in section 2.6. The method to be known as method of "Payload Shift" which will be developed here has the advantage of requiring only very slow displacements and infinitesimal forces. These 2 features are of special value in space vehicle reliability for relatively long periods of time. The method of Payload Shift relies on the fact that if the center of mass and center of pressure do not coincide a torque results. In our particular case for a payload whose mass equals that of the sail a payload displacement of a few feet from the center creates adequate torque. For our proposed design the torque equals 0.15 ft. lb. For a spinning sail this torque causes a precession normal to itself. This has been evaluated in appendix V. The result is an angular velocity of 0.25°/minute for $\alpha = 30^\circ$ and $B_o = 2\times10^{-7}$ psf. At the orbit of Mars this is reduced to about 0.18 $\cos^2\alpha$ degree/min. These values are considered quite adequate for the purpose and are typically 6 or more orders of magnitude greater than solar gravity gradient forces.[1] On the other hand, for a spiral escape from earth starting at an altitude of 1000n miles we have calculated a maximum initial gravity gradient torque of .025 ft. lbs. Compared to our maximum of 0.15 $\cos^2\alpha$ ft. lb of Payload Shift control torque it is clear that Payload Shift does not lend itself to orientation control during escape. However, this is no great loss since a more serious objection already exists. Since the payload shift attitude control depends on $\cos^2\alpha$ a condition would occur during the earth escape spiral where $\alpha = 90^\circ$, in which case our control would fail. This remark holds true for all escape spirals except a polar one in which α is never 90°.

Since our vehicle's Lightness number of .16 renders it unsuitable for polar orbit escape as demonstrated in section 7.1, we realize that if we were to consider earth escape by solar sailing some other attitude system such as Cotter's mutual precession method mentioned in section 2.6 would be required.

To summarize then, payload shift is an effective orientation control method except during manoeuvers requiring $\alpha = 90^\circ$, as is the case for almost all eartn escape spirals.

9.0 GUIDANCE, COMMUNICATIONS AND COMPUTER FUNCTIONS

These three topics are treated in a single section because of their close interrelationship. The mode of

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[1]Ref. 3, p. 15

guidance cannot be chosen until both communication and computer functions have been decided, which in turn are effected by guidance requirements.

Many interesting possibilities are discussed in the literature, and choice among these possibilities of a solution best suited to a solar sailing vehicle can, of course, be a major investigation in itself. The design to be offered should therefore be considered primarily as a means of placing the complete solar sailing vehicle design in concrete terms, rather than an optimization study on guidance, communication, and computer functions.

With these reservations there are nonetheless a number of logical considerations which can guide our choice of a system well matched to the special characteristics of a solar sail. The most important consideration is that of system complexity and reliability. As compared to an ion engine with its heat source, conversion system and heat dissipation systems, the solar sail is of extreme simplicity. To a lesser extent this is true as compared to the ballistic vehicle with rocket mid-course correction devices, and with its very stringent launch requirements, and the need for very precise predetermination of the influence of pertubating bodies. Further the solar sail is not vulnerable to micrometeorites. The large deviations in trajectory, which its relatively long transfer time and unlimited energy source allow it, make the guidance accuracy requirements, at least until the final phase, unusually relaxed. Since a chain is only as strong as its weakest link, a solar sail vehicle seems to suggest the use of as simple a guidance and control system as possible with sacrifice, when necessary, of speed and accuracy. Further calculations based on ref. 23, and carried out in appendix VII show that optical tracking of the 250 ft. dia. sail with a 100" telescope, will be possibly all the way to Mars. Radar ranging to the limit of available equipment is also helped by the 96-97% radar reflectivity of aluminium coated mylar.[1]

As for communication, if we are to receive data about Mars we must either have adequate communication facilities or we must recover the vehicle. A solar sail is poorly suited to recovery, first because we must then add a velocity matching requirement on the return trip, and second the reentry problem sacrifices the main asset of the vehicle, great simplicity and reliability, by adding a mission requirement of distinctly less reliable a nature.

Putting all the above elements together makes us conclude that:

1 A telemeter link to Mars will be the mode of data recovery, and therefore will be necessary for mission success.

2 Since telemetering failure will mean mission

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[1] Ref. 109

failure we are not reducing system reliability appreciably by operating the vehicle as much as possible by earth command. By eliminating a sophisticated computer we are therefore significantly raising the system reliability, lowering cost and complexity.

3 Our guidance system data must be suitable for telemetry in as compact form as possible and must rely as much as possible on ground optical observation to eliminate complexity, and unnecessary telemetering requirements. Accuracy, especially in early phases, can be relaxed as compared to other vehicles.

4 Control command functions must be in a form for direct application to the attitude control subsystem (payload shift system).

5 Advantage can be taken of known approximate location, velocity and acceleration. The first two based on optical data, the third on sail orientation. Orientation can be determined with adequate precision as will be seen later.

To implement the above we suggest the system described in the following two subsections.

9.1 Guidance

Dr. Tsu has suggested that the sail should be turned toward the earth at regular and fairly frequent intervals.

This will allow optical observation and in the event of telemetering system failure or structured failure, will allow direct knowledge as to the path of the vehicle.[1] For any position on the transfer orbit, the correct angle for maximum reflection to an earth station can be easily calculated. Except during the final portion of the mission viewing will be from the dark side of the earth. However, for the proposed 250 day transfer time viewing from the day light side of earth would be required during the final Mars approach phase. Even under these circumstances as seen in appendix VII viewing is possible, at least on the basis of our simple calculations. However, this point requires further study.

We note that with the sail's slow rate of precession of about $\frac{1}{4}^{\circ}$/minute the problem of orienting the sail is simplified, as it needs merely to slowly traverse the angle of optimum reflection to be seen. Both knowledge as to approximate sail location and the use of photomultipliers can contribute to improved tracking ability from the telescope. Because of the low sail acceleration we further note that the effect of the earth facing manoeuver on the trajectory is a very small pertubation. The possibility of using angular measurements from earth has been examined in a recent paper by F. H. Kierstead, Jr. of the Goodyear Aircraft Corp.[2] His earth based approach is essentially

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[1] Private communication, Dr. T. C. Tsu
[2] Ref. 71

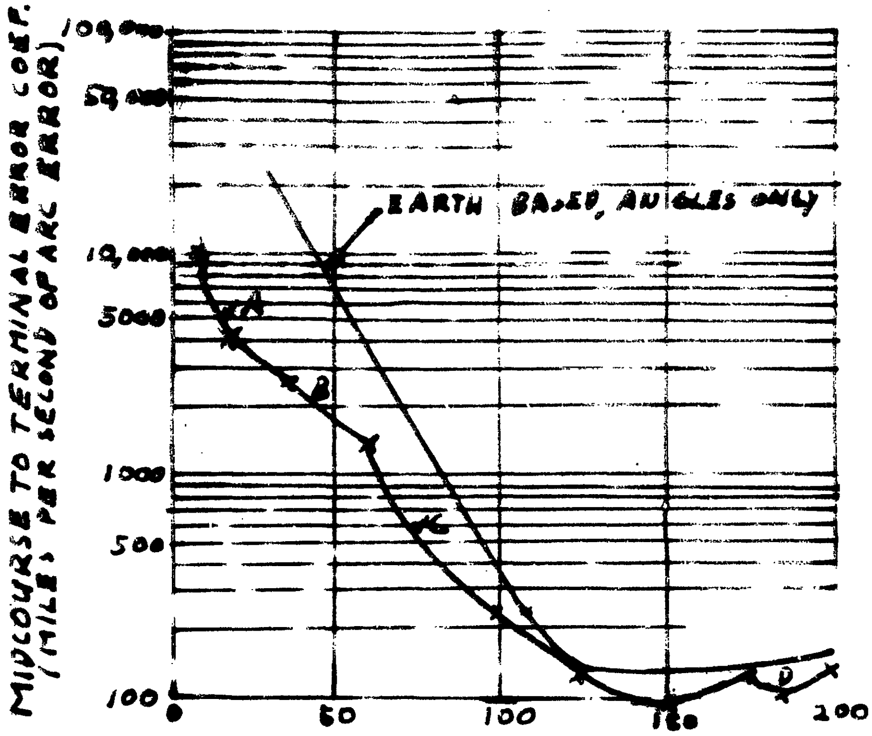
the same as that used for determining orbits of comets.

Kierstead used the technique of linear error coefficients to compare various navigation modes on the basis of their midcourse to terminal miss distance. This he plots in miles/second of arc error (in angle measurement). His Fig. 2 (not reproduced) then goes on to relate any given "miss distance" to actual distance of passage as a function of relative velocity. In this manner Mars focusing effects are taken into account.

In Fig. 25 we have combined results of Kierstead's figures 3 and 13 to show how Earth vs. "on board" navigation compare for a Mars vehicle. The figure shows the somewhat greater accuracy of "on board" measurements in the initial phase assuming equal angular accuracy. However, if earth measurements are made to 1 second of arc accuracy the resultant errors seem entirely acceptable. This is all the more true because of the solar sail's continuous propulsion which allows considerable orbital change even relatively late in the transfer orbit. 1 second of arc is readily obtainable using existing telescope photographic techniques.

Further refinements, if found necessary, can include use of radar ranging during the early part of the transfer with the large flat sail area acting as a passive target. Kierstead's fig 9 (not reproduced here) shows the gains made by this method as a linear error function of range and angle error. At a cost of increased complexity terminal

FIGURE 25



TIME FROM LAUNCH (DAYS)

LINEAR ERROR COEFFICIENTS

FOR 150 DEGREE MARS TRANSFER

(BASED ON FIGS 7 & 12 REF 71 by [illegible])

A - EARTH - SUN

B & D - MARS - SUN

C - EARTH - MARS

guidance could also be based on "on board", measurement of Sun-Earth angles.

We have not yet mentioned the effect of sail acceleration on the above analysis, or for that matter the effect of extrapolation of Kierstead's data based on a 150 degree 200 day ballistic transfer to a 250 day logarithmic transfer. Kierstead himself mentions the problem and concludes that values for similar trajectories will not be far different. This remark of course cannot apply to the effect of the Solar Bounce. This fortunately, is a small acceleration producing small ΔV between reasonable intervals. If we therefore know orientation to a moderate degree of accuracy we can know the acceleration vector, to a good approximation. Analysis of the initial phases of the trajectory when the sail is still relatively close to the earth should provide an empirical correction to establish a more precise value of the vehicle lightness L. There remains then to fix the sail orientation to moderate accuracies. We propose that this be accomplished by use of two sensors.

One sensor will track the Sun and Earth. The second will track the Sun and Mars. We can now consider the Sun as our reference axis and Earth-Sun or Mars-Sun as our reference plane. In this coordinate system we can instrument our device to measure two Eulerian angles representing sail orientation with respect to the two above reference plane.

We have now provided redundant information from which sail orientation is implicit. The redundancy provides us with a choice in which data we use, a desirable feature in that for any orientation we can use the plane whose data is more precise for the particular vehicle position. In addition, the provision of a redundant system contributes to vehicle reliability.

We note that, if desirable, the above sensors could by the addition of an angle measurement, (i.e.: Sun-Earth or Sun-Mars,) be used as an "on board" navigation system. This we do not suggest, however, in that we feel that emphasis must be on simplicity and reliability rather than performance. Acquisition and readout errors of the order of 1 degree are contemplated. The type of sun sensor might for instance be adapted from the highly reliable type developed for sounding rockets on which much field experience already exists, (ref. 47). Another and possibly more sophisticated design, based on trackers successfully used in connection with balloon astronomy, employs a beam splitter and nulls the output of two photo detectors, (ref. 134).

By using only planet trackers instead of star trackers we eliminate the need for photo multipliers. The low accuracy desired will eliminate the sophisticated procedure required to null the sensor on the center of a planetary disk and the requirement for correction to compensate for changing planetary phases, as is necessary when accuracy does not permit treating the planet in question as a point source. The above arrangement also obviates the need for use of both coarse, broad field and fine, narrow field sensors. Finally, the Mars tracking function can be used to orient the camera system and to trigger it when the planet passes at a preset angle with respect to the sun. Alternatively triggering can be based on the planet subtending a given angle at the tracker.

As for alternate navigation schemes, a number have been proposed including: recording star pattern 180° from the sun coupled with radio ranging (ref. 67), sun and two of nine principal stars, (ref. 135). Two sun-star measurements plus angle subtended by sun, two sun-star, a planet-star and time, two sun-star and two planet-star measurements (ref. 72), and the optimization study previously discussed involving various combinations of earth, planetary and sun data as the mission progresses, (ref. 71).

9.2 Communications, Computer Functions, and Auxiliary Power[1]

As indicated earlier our proposed vehicle design

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[1] In this section and the following one on instrumentation we shall draw on the very excellent and thorough 4 volume study of a Mars probe by M. Trageser and associates at the MIT Instrumentation Lab. (ref. 135) for weight, power requirements, and where indicated specific hardware. The writer wishes to thank Messrs. Trageser, Dahlen, Magee, Bowditch, Toth, and Scholten for having made available this material and for some very interesting and stimulating discussions.

involves a minimum of on board computer type functions and substitutes the extensive use of telemetry. Since we propose to send back photographic data on Mars our telemetry requirements are primarily dictated by that purpose, and other requirements will then be easily met. Let us consider for the moment that navigation requirements will be met by computing a new fix every ten to twelve days, (for considerations dictating optimum time intervals see ref. 71, p. 14.)

Telemetering requirements will be as follows:

From probe:

1. Data on orientation of sail for preceeding period (two angles from Sun-Earth tracker, two from Sun-Mars,) measured to around 1 degree accuracy.
2. Temperature.
3. Other scientific data, (see section 11.4,).
4. Signal that photograph has been taken.
5. Telemetry of photograph.

To probe:

1. Two angles specifying orientation during next period.
2. Direction as to whether angles to be used are in Sun-Earth system or Sun-Mars system.
3. Command to initiate a precession for purpose of observation of sail from earth, variant on command #1, to be used in case of failure of automatic timing sequence.)

4. Command to start photographic mission, (in case of failure of automatic equipment.)
5. Command to control rate of data transmission, (in case of failure of timed sequence.)

For the above we shall require considerable electrical energy. We shall assume for the purpose of this study a ½ k.w. capacity. This very large requirement is based on the photograph telemetering requirements. To meet the requirement, will require approximately 75 ft. of solar cells based on 10% effeciency in the vicinity of Mars, (figures based on data of ref. 130). The above reference makes the worthwhile point that the low acceleration of a, in that case ion propelled vehicle, allows the use of an ultra light "shingled" mounting for the individual solar cells without use of casing. Estimates on weight of silicon photo voltaic cells ranges from about 75 to 175 lbs/k.w. varying principally as a function of assumed casing weight, and on random vs. fixed orientation. See ref. 21; 105; 24 table 5; 19; 20 p. 49; 135; 123 fig. 4; and 130. Ref. 20 in particular shows improvement in efficiency possible by use of an infrared filter (which allows cooler operation.) The same reference shows the opposite effects of increase in cell operating efficiency, (due to lower temperature,) and decrease in power flux between Earth and Mars orbit. The overall effect is an almost doubling of the lbs/k.w. ratio.

The optimistic 100 lb./k.w. and 150 ft.^{2}/k.w. ratio we have used as a basis is in keeping with the remarks of Messrs. Hebeler, and White (ref. 130) concerning the lightweight mounting possibilities inherent in a low acceleration vehicle ~~device~~. We also note in passing that a weight of 40 lb-80lb./k.w. is estimated as feasible with the use of thermionic devices. These, however, are not yet fully developed. A thermionic device using part of the sail as collector by making such a section parabolic offers interesting future possibilities.

We will, of course, require a sophisticated telemetering link and in addition one or preferably two receivers, (a very desirable redundency,) and a directional antenna system. Discussion of requirements placed on this link is differed to the next section, while motor types, power and weight estimates are evaluated in section 11.2.

As for computer equipment, we have reduced our requirements to very modest ones. We require a clock and it seems that a crystal oscillator at 100 KC might be appropriate, with an annual drift of less than 1 part per million (ref. 135, p. 326). We require an appropriate simple memory device to activate the previously enumerated commands. We require a closed loop servo that will change the sail orientation with respect to the trackers in accordance with the guidance command. One such closed loop system must activate the payload shift mechanism by first orienting it and then extending the payload. A two state "in-out" mode is proposed for design simplicity. We note that for the periodic orientation of the sail towards the Earth the desired orientation is very simply that whose normal bisects the angle formed by the Sun-Earth tracker. The camera and instruments each require their own controls quite analogous to those presently used on earth satellites.

Necessarily the above description is very sketchy, but the reader will note that through use of earth guidance and a ground based computer system (whose design is not included in this paper) complexity is moderate. As discussed earlier, the clock functions can be overridden in the case of malfunction, by command signal.

One also notices the absence of flywheels. This is because the inertia of the sail is such that any desired movement of the payload or instruments can be made with negligible reaction on the part of the sail. A point which requires further study, however, is whether the sail taken as a whole will be steady enough or whether the trackers antennas and possibly camera will require vibration isolation and, or inertial platforms. The question of vibration amplitudes as discussed earlier is one of considerable difficulty.

As for alternative modes of communication a very interesting proposal is made in ref. 130 by H. K. Hebeler and R. D. White at Boeing. They suggest bypassing the need for electrical power for the telemetering link in favor of an optical data link consisting of a mirror, a polarizer, and a Kerr cell modulator. The Kerr cell would modulate sunlight aimed at the earth and thus an amplitude modulated signal of white light could be received by a large earth telescope, passed through a photo multiplier and from then on electronically analysed in the same manner as a radio signal. Because of the undeveloped state of this idea and the writer's lack of knowledge about pertinent details that would bear on its feasibility (such as atmospheric "seeing" random fluctuations), it has not been seriously considered. It is mentioned, however, as a rather interesting potential communication link.

10.0 INSTRUMENTATION

The most important piece of instrumentation on our proposed vehicle is the camera and its related data conditioning and telemetering system. Present technology in this area has made great strides and the Russian Lunar probe's photograph of the back side of the moon is an excellent example of results which have already been achieved.

We will use the results and many of the design features of the MIT proposal, (ref. 135,) as the basis for our camera and telemetering description. The proposed camera will have a 19" focal length and a 1.5" aperture. Nominal distance from Mars will be 5000 miles with a ± 100 mile tolerance. For a film resolution of 150 lines/mm a ground resolution of 7 lines/mile on Mars should result. The image of Mars should roughly fill a 6" plate. The number of bits of information will then total approximately 1 billion. (Above resolution values are quoted from ref. 135, and are on the whole optimistic, in any case 75-100 lines/mm could definitely be met for high contrast features.)[1]

The proposed photograph further requires image motion conpensation of about 1 millirad/sec ± 10 %.[2] The information so obtained should answer a great many questions about Mars, in particular comparable photographs of Earth according to the MIT study would definitely reveal the works of man. They find that "the image of Mars obtained....will increase our knowledge of Mars by a factor exceeding the advantage gained over the unaided eye by the use of the most powerful telescopes in the observation of the moon."[3] The MIT proposal involved recovery of the vehicle and only touched on telemetry as an alternative. The telemetry requirements they find, however, would be as follows. A 6"x6" photograph at 100% contrast and 200 line/mm (sic) would require assuming, a 2 level photograph, 9.3×10^8 bits. At 50 million mile range the report assumes 1 joule/bit.

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[1] For a sophisticated information theory approach to the question of optical system performance capabilities as a function of the various applicable factors see ref. 48 by Dr. R. Scott at Perkin-Elmer.

[2] Ref 135, section 14.18

[3] Ibid, Vol. 1, p. 6

Adapting its data to 5 levels, but only 75 lines/mm we find that at an average power of 450 watts the data would require about 25 days to transmit. At 1×10^6 mile it would only require a half hour. The above values are based on the equation, (ref. 135, p. 442,) :

$$R\ max = \left[\frac{E_V\ G_V\ A_e}{4 \pi K\ T_e\ L\ S/N} \right]^{\frac{1}{2}}$$

where:

R max	=	Range in meters
E_V	=	Transmitted energy/bit in Joules.
G_V	=	Transmission gain of vehicle antenna.
A_e	=	Effective receiver aperture of Earth antenna in m^2.
K	=	Boltzmann constant 1.38×10^{-23} watts/0^o cps
T_e	=	Effective receiver temperature 0^o K.
s/n	=	s/n ratio required at output of a matched filter to provide reliable reception.
L	=	System losses.

Where many of the above factors are frequency sensitive:

G_V, the Vehicle Antenna gain for the dipole used $= 4 \pi AV/\lambda^2$
Where A = 4 sq. ft. for the proposed design.

With the above in mind we feel that a logical extension of the technique successfully used on the Vanguard "Cloud Cover" Satellite will be adequate for intermediate data storage. The process involves a magnetic tape recorder, on which the information is stored. (See ref. 121 for a detailed account and photograph of system hardware.) For our application an image orthicon tube might be the intermediate storage, replacing the conventional film, if sufficient resolution can be obtained. A number of other techniques are also possible and we will not attempt to indicate which would be optimum, a choice would definitely involve considerable study and cannot be the object of a casual selection.

We propose that part of the signal conditioning equipment include provision for controlling the rate of transmission information as the probe returns toward Earth. Here again redundency may prove a desirable tool for it would be desirable to start transmitting back as soon as possible at reduced bandwidth to minimize the risk of system failure after photograph and before telemetry. Further a presently somewhat unpredictable degradation will be suffered by the tape in storage during the return journey due to damaging influences discussed in considerable detail in section 1.3. After initial communication we propose that the transmission of our (single) photograph be repeated. With decreasing range, quality of reception should improve and statistical correlation techniques will allow production of a greatly improved composite, after several complete transmissions.

Beyond this minimum objective, a more ambititious goal which should be considered is increasing storage to allow for color transmission. Here one might consider use of only two suitable filters chosen so that, in accordance with the recent work of Land at Polaroid, a color reproductiom could be made using only two colors. One would, of course, need to compare the sensitivity to transmission noise of this approach to color reproduction versus that of the more orthodox three color approach. In any case the large increase in information content by use of color is well known. Some reduction in resolution would even be worthwhile.

Further desirable instrumentation might include, (as selected from refs. 24, 130):

1. A micrometeorite counter
2. A magnetometer
3. A radiation sensor
4. A cosmic ray counter
5. A spectrometer (if weight allows)
6. Temperature gauges.

These instruments would of course provide interesting information during the entire transfer and would be telemetering during this period. Weight and power estimates for all the above cited purposes will be found in section 11.(Table VI).

11.0 AN INTEGRATED VEHICLE DESIGN

In the previous sections of part III we have examined a number of the important features and characteristics of a 400 lb. Mars photographic reconnaissance vehicle using a class 3, Centrifugal Sail, of 250 ft. dia. and vehicle lightness number $L = 0.16$. In this section we shall incorporate these features into a vehicle design which will bring out possible design solutions to a solar sailing vehicle.

Once more it is well to mention that optimization has not been attempted. What is done, as a logical next step to previous work in the literature is to consider the general problems of solar sailing and translate them into a set of possible vehicle design decisions. Those hopefully will serve as a preliminary, but realistic starting point for further studies.

Vehicle design requirements can be conveniently grouped into overall configuration; motors, trackers, and antenna; electronic and electrical system; instrumentation; launch and development process.

11.1 Overall Configuration

As shown in fig. 26 the proposed overall configuration consists of:

Part 1. The sail proper a centrifugal sail with a 250 ft. o.d. and 22 ft. i.d. made of .00025 in mylar with a 3000 A° vapor deposited aluminum coating, possibly including some silver.

Part 2. A fiberglass filament ring with a diameter of D=22 ft. and another with a diameter of 6 ft.

Part 3. A "spider web" type section of fiberglass filaments between the inner ring and outer rings (part 1.), and between the inner ring and part 4.

Part 4. 3 compression members in the form of 3 ft. I beams of uniform stiffness, (tapered section.)

Part 5. A connecting constant speed motor of special design.

FIGURE 26A

CENTRIFUGAL SAIL MARS RECONNAISSANCE VEHICLE

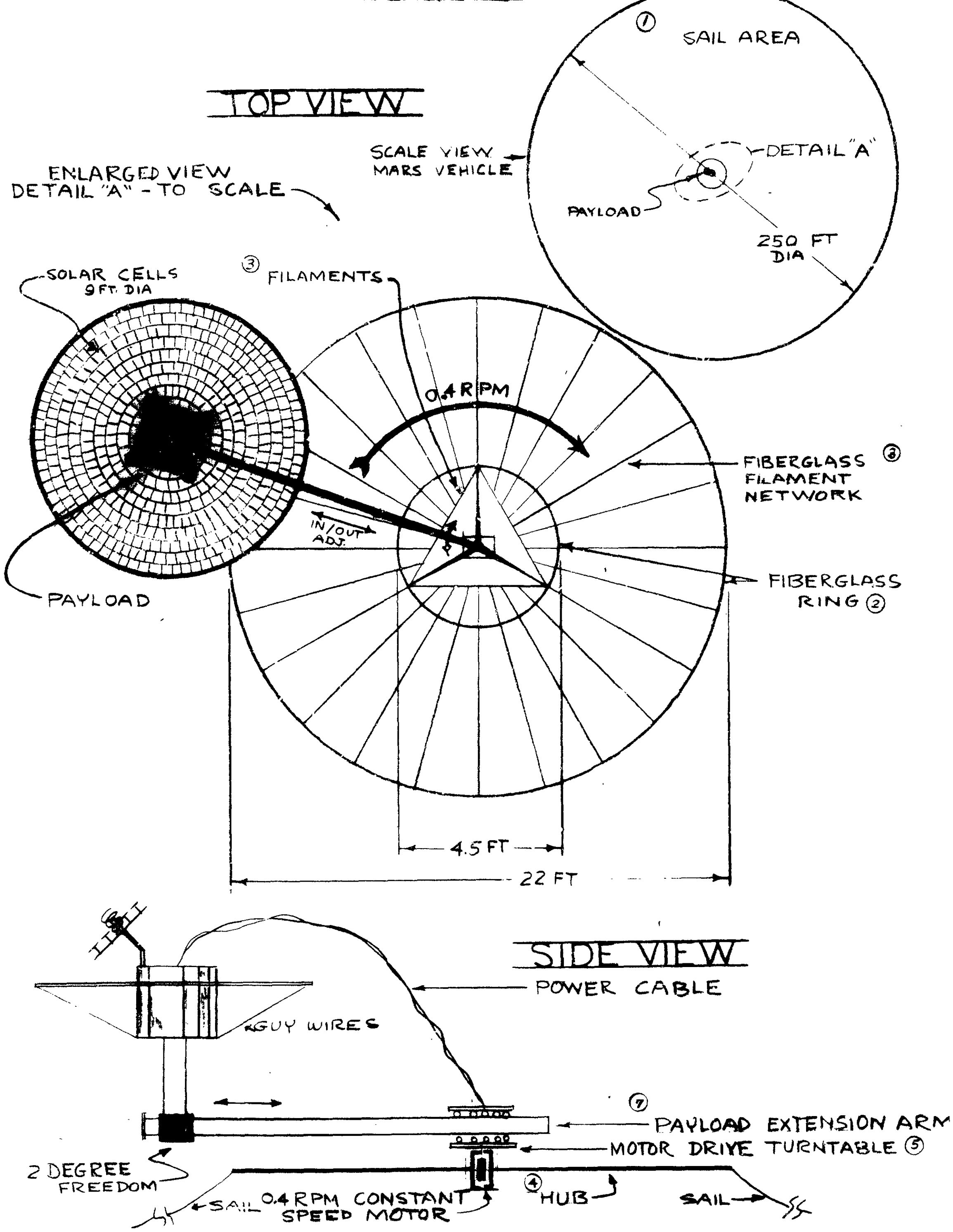
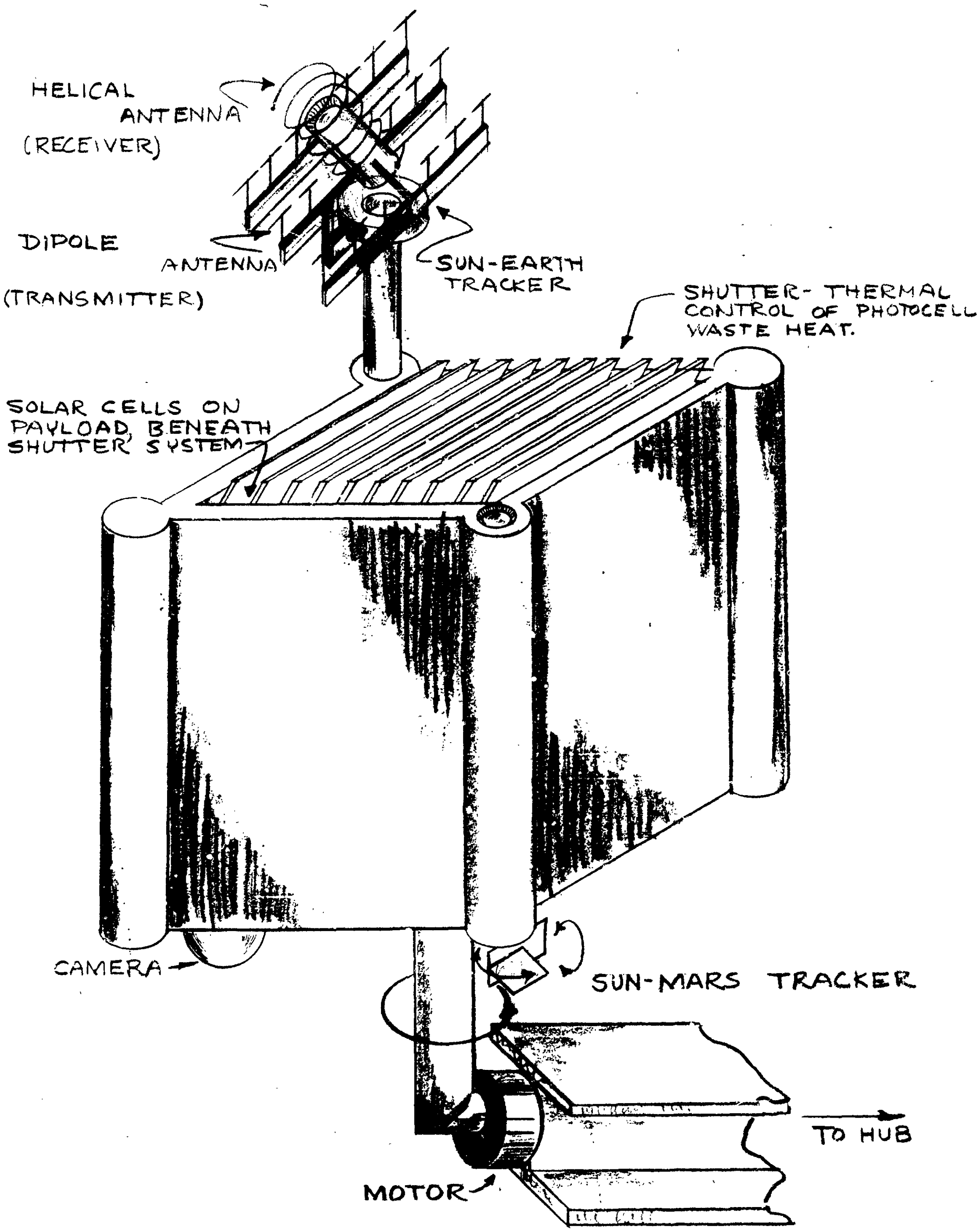


FIGURE 26B

PAYLOAD DESIGN



<u>Part 6.</u> A motor drive turn-table.

<u>Part 7.</u> A 10 ft. uniform stiffness I beam, (tapered) joining the sail and payload. This I beam is used for payload shift.

<u>Part 8.</u> The payload pod proper weighing ≈ 190 lbs. including a surrounding array of solar cells of 9 foot diameter.

The dimensions and values to be listed over satisfy static requirements, but are only suggestive in that the final values would, in the writer's opinion, be based on the dynamic analysis of the system, a problem of some difficulty.

Let us now discuss each of the listed features and their basis.

Part 1. The sail proper, has been discussed in detail earlier in section 7. At .001 lb./sq. ft. it is the lightest design potentially available in the immediate future.

Part 2 & 3. The fiberglass filaments were chosen on the basis of their strength/wt. ratio, and availibility in very small sizes. The inner section had to be transparent because of navigation requirements during payload shift. A clear mylar section was considered and rejected on the grounds of ultraviolet hazzard when not protected by aluminium. To reduce weight further requires eliminating the sail backing as discussed in section 2.4. This, as stated earlier, is not an immediate prospect and requires considerable R&D.

Part 4. The central "compression members" are required to carry loads in bending. We note that these loads will always be normal to the sail, (since acceleration and precession forces must act in this plane.) For this reason tapered I beams are chosen as the most efficent bending load carrier. On the basis of ref. 116, 118, 119 berrylium despite its serious manufacturing and handling problems, is the most desirable material. The equilateral triangle offers an apparently irreducible minimum of 3 members for providing plate stiffness. A continuous plate would be possible, but heavier. Other configurations considered included 4 or more members.

Part 5. We require a slow constant speed motor to compensate for bearing friction and thereby prevent angular momentum transfer between sail and payload. We note that bearing loads under 10^{-4} go acceleration at .4 rpm are most conducive to reliable operation. The motor design is discussed in section 11.3.

Part 6. This platform is mounded directly to the motor shaft. On it is located a friction drive to move part 7 in and out.

Part 7. The payload shift extension arm. This I beam has a capability of moving 10 ft. in or out. By this means a 10 ft. payload shift motion is accomplished.

Part 8. The payload package as shown is on a two degrees of freedom mount. By remaining pointed at the sun it optimizes solar cell energy conversion and allows the Sun tracker to be built in without any relative motion. It also simplifies the payload thermal equilibrium problem. Berrylium sheet is most suitable to reduce micrometeoritic penetration, (ref. 116) and so is proposed for payload walls. Payload configuration follows that of the MIT design, (ref. 135,) in using a modified rectangular solid. In space, aerodynamic considerations are of no interest and a rectangular solid roughly approaching a cube is attractive because of the greater packing efficiency which can be achieved as compared to a sphere.

Part 9. The variable opening shutter is provided to control the input of Flux to the solar cells on the payload, By indirect control (i.e. controlling silicon cell waste heat) we can maintain desired payload temperature.[1] In fact spherical configuration for small sizes tend to waste the space between the sphere and the inscribed cube. A cube comes close (after the sphere and cylinder) in optimising volume/area ratio. It is much easier to fabricate and lends itself well to such components as camera and Sun tracker. The use of corner tubes, (following the MIT design,) is a convenient way of eliminating stress concentration, improving rigidity and housing the Sun trackers. As for the

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[1]This approach was previously proposed, see ref 135.

silicon cells, they require compression members to hold them out, but as shown in fig. 25 they do not require support in bending as this is done with guy wires. Guy wires could also have been used to reduce the weight of the other compression members, (parts 4,7,) but the vehicle configuration is such as to prevent such an approach unless the payload is separated axially several feet from the sail. We note that according to a static analysis, tension wires on the sail hub and on the payload shift arm would be required in opposite directions. However, considering dynamic characteristics, tension members in both direction would be desirable on hub and arm.

11.2 Trackers and Antennas

Our proposed design calls for a solar cell collector area always normal to the sun. This allows the sun seeker portion of our astrotrackers to be built into the payload. Nulling of the sun trackers is done by controlling the 2 motors which give the payload its two degrees of freedom with respect to the payload shift arm. The third degree of freedom provided at the juncture between sail and payload is required for angular displacement of the payload during payload shift attitude control manoeuvers. With the 2 restrictions imposed by payload shift requirements and constant sun orientation we require 2 additional degrees of freedom for Earth and Mars trackers respectively. For the Mars tracker a rotatable, oscillating mirror provides the required image in the tracker. For the Earth tracker a separate mount is used. This mount connects the Earth tracker to the Sun tracker. Together they define the Sun-Earth plane. The mount then serves the additional purpose of mounting the helical receiver and dipole transmitter antennas. (Alternate antenna types such as spirals have not been considered in detail.) In the above manner our antennas are kept oriented towards the earth which is required for the long periods of transmission of the telemetered data. Use of a constantly pointing directional antenna allows continuous transmission. Consequently no batteries for peak loads are provided in our design, and continuous transmission is used instead.

11.3 Motors and Drives

In interplanetary vehicles, motor selection must take into account low loads, extremely high reliability requirements, friction problems at bearings and brushes, as well as the importance of low weight and power requirements. We propose to use D.C. power. In addition, D.C. power is suitable for low voltage application and does not require D.C. to A.C. conversion from the silicon cells. The major obstacle is of course that of commutation in a vacuum. Haussermann in ref. 6 proposes a solution to this problem. His "impedence commutator motor" uses permanent magnets for field excitation, and provides a variable induction as a switching signal to a transister switching device. Our most severe reliability problem may well be that of part 5, a continuously running 0.4 rpm constant speed motor.

We require very close speed control in order to maintain the payload stationery and just compensate for bearing friction. This undoubtedly requires a closed loop speed control using either planetary tracker or possibly an accelerometer located on the edge of the hub as sensor.

For orientation change it is a simple matter to vary the speed control and thereby rotate the payload to a predetermined angular setting as provided by ground command. In addition, analysis may show the need for a closed loop system for maintaining orientation, although spin stabilization tends to minimize the need.

Other existing drive functions include a device on part 6 to drive the payload shift extension arm back and forth from a centered payload to the payload 10 ft. off center ~~moment arm~~ position. A simple possibility would be to take advantage of the constantly running motor (part 5) with a power take off which would power a friction drive. This requires a 90° transmission link and a clutch, (possible magnetic.) The advantage of this approach is avoidance of another off-on motor.

Additional motor requirements are also of the low capacity type. They include a drive motor for the two degrees of freedom required by the payload to remain oriented toward the Sun. They also include drives for the two planetary trackers. The antenna system as shown is mounted with the earth tracker, and the camera drive can be run off the Mars tracker. The payload we recall is to be continuously pointed toward the Sun.

As mentioned earlier, it has been assumed that because of the sail's very large inertia we can consider it as an infinite angular momentum sink for the payload. This assumption, while undoubtedly valid for a rigid system, bears further study for a membrane type structure with its centrifugally induced pseudo-stiffness. If those fears do turn out to be justified and oscillations of appreciable amplitude exist the best solution might be to place the entire payload on an inertial platform and thereby obtain decoupling of sail oscillations and payload. In no case does it seem reasonable to suppose that a flywheel system will be required, in as much as at worst it would appear that the sail is a very large angular momentum dump which interacts with the payload to produce transient and or oscillatory modes.

11.4 Electronic and Electrical Systems

To reduce radiation and penetration hazards, (except for secondary emissions), it is desirable to place the electronics as far from the surface as possible. This consideration, however, has to be balanced against heat dissipation requirements. We have assumed that the latter problem can be met and therefore have located electronics well inside the payload. Our electronics includes primarily a transmitter, receiver, data conditioning system, tape recorder, image orthicon tube and associated amplification equipment, as well as the oscillator "clock" and a "memory" storage to activate subsystems which must be turned on either by ground command, time sequence, or both. As compared to other vehicles, the size and sophistication of the latter is very modest. Additional electronic equipment, in preference to more Vulnerable pneumatic or hydraulic systems, are required in connection with the tracker, nulling drive, and the closed loop drives mentioned in section 11.2.

A special problem present in our design is that of power transmission between the payload source and the sail hub while in the "shifted position." A sliding contact does not seem reliable enough and we tentatively suggest a flexible cable whose inertia would insure its staying clear of the vehicle. An alternative is a take-up reel, which again involves a sliding contact. Our proposed approach may appear unreasonable in the light of earth experience with "loose" objects, but there being no stray gusts of wind in space, it may well be that the inertial considerations permit a confident conclusion as to behavior of a wire supported in tension at its ends and "trailing behind."

11.5 Instrumentation

The problems of instrumenting our proposed vehicle seems to be confined largely to the photographic system. By that we mean that secondary experiments as listed earlier can be treated as another application of existing satellite hardware. These remarks apply equally well to the problem of micrometeorite gages, temperature sensors, radiation detectors, etc. The spectrograph mentioned earlier would involve an instrument that to the writer's knowledge has not been orbited to date, but proposals for such are known to have been made. Incorporation of a spectrograph is largely a matter of decision based on weight. To save weight we have not provided storage batteries, as is required for lower powered vehicles, and for this reason there should be power to spare on subsidiary experiments and their telemetry during the outbound transfer.

11.6 Launch and Deployment Considerations

The problem of launch and deployment of our proposed vehicle is of appreciable complexity and deserves attention. Assembly in space which might be convenient is of course of no interest in the next few years.

A 400 lb. payload on a parabolic escape as is shown in section 12 is well within the projected U.S. space capabilities of the period beginning in late 1960, early 1961. Even assuming delays, late 1961, early 1962 is a realistic date. The problem is then to show that our vehicle can be packed into a reasonable configuration, and then deployed at some point where aerodynamic effects are negligible. In fact the best location for deployment would be at 50-90 Earth radii where the sail can begin functioning in its normal mode.

The sail proper could be launched by spinning the payload and then letting the sail open centrifugally, however, for reasonable payload diameters the required angular momentum is excessive. (a payload spin of 3600 rpm was calculated for one assumed mass distribution.) A more attractive approach is that of a modest spin which would only cause the sail to spin. But this can then be supplemented by small tangential rockets at the edges of a thin circumferential ring which would be located at the outside of the sail. Calculations in appendix V show that four $\frac{1}{4}$ lb thrust rockets will provide the entire required spin in 32 seconds. The initial spin therefore merely eliminates the need for some other deployment medhanism and microthrust rockets are proposed for bringing the sail up to speed. As for folding of the sail, it will undoubtedly require both ingenuity and slip powder, but is of comparable complexity to the problem of folding the 100 ft. dia. NASA balloon into its 30" spherical container, (ref. 139, p. 143.)

The 10 ft payload shift arm may require a spring released hinged section, which locks in place after opening. The 3 ft. sections of sail hub can fold together to a common apex and be opened by a spring release. The payload itself is potentially suitable for fairing in as part of the last stage. Provisions are required to provide enough rigidity during the high acceleration and vibration levels encountered solely during launch. This problem is not a new one, however, and considerable experience in this has been built up with satellite launchings. In general, temporary pressure stabilization is a very useful approach for the payload. Solar cells deployment would be an adaption to a more difficult configuration of the technique successfully used on the "paddlewheel" Explorer satellite.

Table VI Wt. and Power Estimates

	quantity	weight in lbs.	power
1 Payload			
Camera	1	20	
Motors	5	5	
Data conditioning	--	10	
Tape recorder	1	3	
Transmitter	1	10	450 watts
All other	--	--	50 watts
Receiver	2	9	
Clock	--	1	
Trackers	2	10	
Spectroscope	--	10	
Radiation flux counter	1	2	
Scintillation counter	1	3	
Micrometeorite gages	--	5	
Magnetometer	--	5	
Solar cells	75 sq.ft.	50	
Structure	--	32	
Electronics	--	15	
Miscellaneous	--	5	
Payload total		190	500 watts
2 Payload shift extension		2	
3 Platform		4	
4 Payload shift, and sail to load connection motor and drive	1	4	
5 Sail hub (Fiberglass mesh)		1	
6 Sail		199	

PART IV ON THE MERITS OF DEVELOPING A SOLAR SAIL AT THE PRESENT TIME

There is a growing recognition that the prestige and hence the effectiveness of the United States as a world leader is very strongly influenced by the degree to which our technology, specifically our space technology, leads that of the world. It is therefore serious that, lead times being what they are, we are for the next few years definitely relegated to second place in terms of booster capabilities. As of a July 1959 report the Thor-Able booster with Mars ballistic payload capability of 50 lbs. and the Atlas-Able with Mars payload capability of 200 lbs. were expected to be the largest boosters during 1960.[1] Since the publication of the above report further postponement and cancellations have been announced. It is in this context that we should view the need for continuing our present lead in payload weight reduction through miniaturization and microminiaturization. It also is important to extend our capabilities by other means, such as decreasing booster requirements by providing higher or "infinite" specific impulse propulsion after booster cutoff, without excessive weight penalties and with high reliability. The second major problem as emphasized by difficulties faced to date is to improve system reliability.

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[1] Ref. 96, p. 125

When research for this thesis was begun it was the writer's intention to try and arrive at some substantiated answers on whether solar sailing either in the long or short run could, with reasonable effort, make a worth while contribution to these goals. The study reported here has lead the writer to the conclusion that in two very specific areas, namely an attitude stabilizer (Part II), and a moderate payload planetary reconnaissance mission (Part III), the capabilities, probabilities of success, and probable cost and time factor are such as to justify a substantial research and development effort directed towards a solar sail planetary reconnaissance probe. As to attitude stabilization it is concluded that no new technology is required and that applications can and should be made of solar sail attitude stabilization in the immediate future. The following sections will attempt to substantiate these conclusions, and section 15 will suggest a suitable development program.

12.0 COMPARISON OF CAPABILITIES

In the immediate future the only alternative to a solar sail for interplanetary transfer is a ballistic trajectory with chemical correction capabilities. The MIT Instrumentation Lab Proposal frequently referred to in this study (ref. 135) is a very competent example of this type of vehicle.

Assuming a very similar payload, the consequences of eliminating chemical correction and substituting a solar sail include the following:

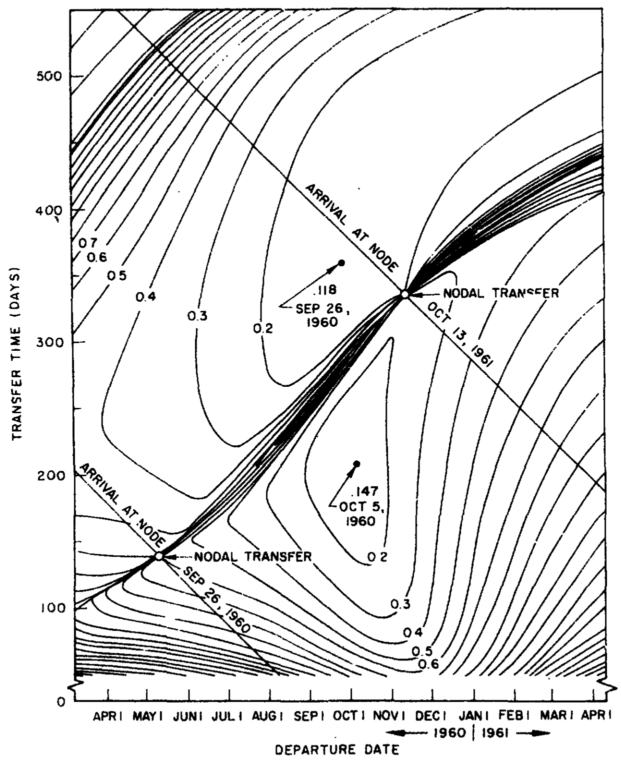
1. Launch requirements -- Substantial relaxation of launch requirements with regard to accuracy of cutoff. Relaxation of restriction on launch times.

2. Trip time and Mass Ratio -- For equal trip time, (one way transfer orbits of around 250 days), our computations based on Sutton's data, (ref. 105), and that of Breakwell, Gillespie and Ross, (ref. 64), show that our proposed solar sail can launch a vehicle of 1.3 times the weight of a chemical vehicle on the basis of same take off weight, (see appendix VIII). Chemical vehicles theoretically can make a faster transfer but the weight penalty and error sensitivities become very severe. Further the $\Delta V = 1.147V$ earth orbital transfer velocity used in our calculations is a rare occurrence for non coplanar, non circular orbit analysis and the penalty for a ballistic vehicle at any other time is considerably higher. This is seen in Fig. 27, a topological transfer velocity versus time plot.

3. Reduced Navigational Instrumentation -- Relaxed on board instrumentation requirements is an important advantage. As has been shown in appendix VII all navigation fixes and trajectory computation can be made on earth because of the sail visibility.

4. Elimination of Payload Attitude Control Subsystem -- The elimination of separate attitude stabilization and

FIGURE 27



TOPOLOGICAL REPRESENTATION OF HYPERBOLIC EXCESS VELOCITY FOR VARIOUS EARTH-MARS TRANSFERS (ELLIPTICAL-NON COPLANAR)
(From Figure 3, ref. 64 by Breakwell, Gillespie and Ross)

Hyperbolic excess departure speeds in units of mean earth orbital speed.

propulsion systems contributes to simplification. For a solar sail a simple attitude control system for the sail is required, and (on the basis of our preliminary study), no separate attitude controls are needed on the payload. Silicon cells can, therefore, be constantly oriented normal to the sun and the directional antenna at the earth with no weight penalty.

5. System Simplification -- The elimination of a number of subsystems including: computer, gyros, accelerometers, flywheels, and chemical microthrust correction system result in a simpler and, therefore, potentially more reliable system. It is felt that the addition of a radio receiver, and a sail orientation and angular velocity control do not represent a corresponding increase in complexity.

6. System Flexibility -- The problem of energy as a function of mission characteristic greatly effects chemical systems. With a solar sail and its unlimited energy, supply, unfavorable launch time, or an incorrect cutoff no longer need mean failure, Instead it may mean a longer trip. After a successful mission sail flexibility is such as to permit the vehicle to commence a second mission. With gradually increasing reliability the reuse and redirection capability may become of great interest.

7. Decreased Mars Pertubation Sensitivity -- The Mars pertubation effect has been shown by Battin to be very useful in providing a non stop round trip, (ref. 62). However, for the ballistic vehicle with a limited energy storage available

for corrections any substantial error in the pertubation, (such as took place with the Russian Lunar photographic reconnaissance vehicle) may well spell mission failure. For a solar sail the unlimited source of energy would prevent mission failure and instead result in a longer mission.

It is felt that the above listed reasons justify the desirability of a solar sailing vehicle, however, because of practical limits on size, manoeuverability in high gravity potential environments, and present lack of development it cannot be said that solar sails will supplant modified ballistic trajectory vehicles. The reasonable conclusion seemsto be that, as has been pointed out in the literature many times, a variety of vehicles are needed for an adequate space program; the above reasons lead us to believe that a place for solar sails exist in such a program.

13.0 RELIABILITY CONSIDERATIONS

As pointed out in the previous section a solar sail vehicle is simpler, and has a substantial reduction in the number of subsystems. In addition the elimination of a substantial computer, gyros, flywheels, etc. represent a very substantial decrease in what one might term the "vulnerable cross section". This cross section is the total cross section on which micrometeorite damage would cause system failure. This in turn reduces structural requirements for meteorite barriers.

A further result of system simplification and the improved mass ratio is to permit the use in many instances of redundant components without placing our vehicle weight outside of the limits feasible in the next two years. In this context it is well to note that estimates of computer reliability are consistently lower than that for other sybsystems. An extreme example is ref. 97, fig. 4, by Xenakes at WADC whose 1958 report gave mean time to failure for a lunar mission computer as 25 hrs. versus 1,060 hrs. for the next most vulnerable element, (astrotrackers). Since then computer development and therefore, computer reliability has made tremendous strides, but it remains by sheer mass of components the weakest reliability link in most system proposals.[1]

We have said very little here about comparing a solar sail vehicle with ion drives, or nuclear or solar heating rockets. This is because the two are not competitive. The above named will not be available for interplanetary vehicles in the immediate future, nor are their system

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[1]A more recent and more optimistic evaluation of computer reliability projected to 1960 optimum potential was made at Perkin-Elmer Corp. They estimate a MTF pf $8x10^6$ hrs. per computer component based on a ratio of 1 transistor per 7 passive components including connections. The same study uses 200,000-300,000 hrs. as tube, MTF, 10,000,000 hrs. as silicon diode MTF, and 50,000 hrs. as motor MTF. These figures are based of course on conservative design in a space environment free from high vibration and acceleration effects. The above data is from a private communication by Dr. Robert Hufnagel whose help is gratefully acknowledged.

weights reasonable for boosters available in the immediate future. Radiator requirements, system complexity, and vulnerable meteorite cross sections will be much greater for those vehicles. Their mission capabilities, however, should be ~~much~~ correspondingly greater and vast increase in payloads are comtemplated.

As earlier stated, no profitable application of solar sailing to the above high payload missions or to manned vehicles is foreseen. Use of ion engines for comparable low payloads, (200-1000 lbs.), suffers from increased complexity even more than the chemical modified ballistic vehicle to which we have compared a solar sail.

14.0 EVALUATION OF COST AND TIME SCALE

A solar sail as compared to other forms of space propulsion is an inherently inexpensive item. Material cost at $10./lb. would be only $2,000, at $100./lb only $20,000. As for development costs, contrary to other propulsion systems, a solar sail manufacturing capability is not different for different vehicles. If one has the know how and capability to manufacture a solar sail, the development cost of a different size sail is small indeed. This means that investment in a solar sail capability will develop a vehicle propulsion system of extremely low unit cost and one which need only be built in a few incremental sizes to be adaptable to a whole range of missions. The solar sail is unique in that payload and propulsion are structurally independent and can be designed independently, in the same manner as outboard motors and motor boats where one matches any appropriate size propulsion unit to the vehicle.

As for the time scale involved in development the next section discusses required R&D. From it, the reader will probably agree that an 18 month to 2 yr. program from initiation to completion of the sail unit is entirely reasonable. The payload development is not substantially different from other vehicle payload programs, except that it is simpler than most interplanetary systems where payload and propulsion configurations are more intimately tied. The development cost of a second generation solar sail involving a sail lightness number of $L_o = 1$ or greater has been estimated by Mr. Carey of the MIT Civil & Sanitary Engineering Dept. as stated earlier at $500,000.

The material development cost of a first generation sail, on the other hand, would probably be mainly that of putting into production 1/10 mil mylar film, (already made on trial basis as reported earlier) instead of the present $\frac{1}{4}$ mil. No cost estimate is available, but one should point out out that such a thin membrane is also of interest for other space programs such as a solar heating rockets, light weight parabolic space antennas and various condenser and radiater proposals.

15.0 A SUGGESTED PROGRAM OF DEVELOPMENT

In the previous 3 sections we have seen some reasons that make a solar sail capability desirable. In the body of the thesis we have seen the problems requiring answers. Here we shall indicate what appears to be a logical development program.

1. Materials Investigation -- Alternate materials to mylar, in section 2.4, should be included together with mylar in a systematic study of the relevant space effects discussed earlier. The NASA passive communication satellite should help greatly in this, especially if prior thought is given to the matter. In addition an experimental program should be undertaken to refine present knowledge on ultraviolet and x-ray effects, and on self damping and stiffness properties of these thin films. At the same time the sail fabrication problems and development costs should be evaluated.

2. Dynamic Analysis -- Further experimental and analytical work should be done to insure an adequate understanding of dynamic characteristics on which only tentative information, (as discussed in section 7.2) now exists. Properly scaled models would probably give useful answers if run in a high vacuum chamber, but as seen in section 7.2, moderate vacuums are unsatisfactory.

3. Optical Properties -- An analytical and possibly experimental study taking into account predicted dynamic performance and its effects on configuration are required

to determine the limits of optical tracking.

4 Packaging Design -- Although packaging and deployment problems for the sail are an entirely logical extension of the present state of the art, the size and design of the sail is sufficiently different to make a preliminary study of these aspects desirable.

5 Overall Vehicle Development -- It is felt that the results of the 4 above named studies should be sufficient to definitively answer the question of general feasibility of the "Centrifugal Sail" design approach, and to a large extent other designs.Either in parallel, to save time, or after the first 3 steps, a design study will of course be required to refine and improve or possible radically alter design ideas found in the literature. This phase is very similar to that of payload development for other vehicle types and would both apply to and derive benefit from studies on ballistic vehicles with error correction capabilities.

The above program could lead to a definitive answer as to whether what appear to be a promising vehicle concept should lead to an actual vehicle. In the event of affirmative findings it will be highly desirable for initial production phases to overlap with some of the above stages. It is felt that a tightly run project could very probably lead to a vehicle within 2 years.

The above development program is then what is needed to lead to a vehicle now. It appears as a desirable program at an acceptable cost. Its full execution requires substantial funding. A second question on which the writer proposes to close is what further work is suitable for thesis activity at MIT in the event that the full program previously mentioned is not initiated. The following areas offer good possibility:

1. More complete analytical and or (with funding) experimental study of vibration behavior of a solar sail. Of special interest, are the centrifugally stabilized type extensively discussed here. Any experimental work in a high vacuum will be faced with problems involving the motor and motor outgassing, one might take advantage (as suggested by Prof. Paynter) of the possibility of using a rotor only and providing a rotating field in the vacuum chamber to cause the device to spin.

2. A detailed feasibility study of a ground based navigation system for a solar sail such as the system for which a tentative evaluation has been presented here.

3. A general study of attitude control of a solar sail propulsion unit including dynamic response and stability parameters.

4. A solar sail orientation control design study for an earth satellite. A suitable application such as a satellite telescope could be studied, and the ideas developed or reported here might serve as a starting point. The

study would include a determination of suitable damping, required dimensions and optimization considerations.

5. A more detailed study of a solar sail vehicle design for a Mars reconnaissance mission. A number of design alternatives could be compared as a step towards design optimization.

6. A feasibility study for an ultra light sail of sail lightness number L = 1 or better. This would require investigating methods of sublimating all but the reflecting coating of the sail after its deployment in space.

APPENDIX I

A Derivation of Earth Bounce on a Satellite Vehicle

As defined earlier Solar Bounce or the rate of momentum transfer/unit area is equal at normal incidence to 1.95×10^{-7} psf. With the earth's albedo of approximately .36 (Ref. Van Allen - p. 80) and given the fact that the earth's mean temperature is roughly a constant we can calculate Bounce on a satellite due to the earth (note according to Ref. 109, p. 7, the earth's albedo varies seasonally from 0.32 to 0.52, and therefore our calculations are only approximate). For these calculations we shall continue to assume $\overline{R}$ (reflectivity) = 1 although in the infrared this may be somewhat inaccurate.

The earth absorbs:

$$S\,(\pi R^2)(1 - A_e) \qquad (101)$$

where:

S = solar constant at 1 A.U.
R = earth's radius
A_e = albedo of earth

If we assume as a rough approximation isotropic reradiation (in the infrared) then by conservation of energy,

$$S\pi R^2\,(1 - A_e) = S_r\,(4\pi R^2) \qquad (102)$$

where S_r = earth reradiation constant

$$S_r = S\left(\frac{1 - Ae}{4}\right) = \frac{1 - .36}{4} S = .16\,S \qquad (103)$$

For the sunlight side of earth we add to this the reflected light to arrive at a total light flux. For rough calculations we can assume that, for satellites near the earth in comparison with the earth's diameter, the reflection will be uniformly distributed. This assumption will be exactly valid only for an area normal to the sun and will constitute an upper bound for all other points on the sunlit side.

Using an albedo of 0.36,

$$S_r \leq S_e \leq S_r + A_e\,(S) \qquad (104)$$

$$0.16\ S \leq S_e \leq 0.16\ S + 0.36\ S \qquad (105)$$

$$0.16\ S \leq S_e \leq 0.52\ S \qquad (106)$$

at distance D above the surface of the earth,

$$4\pi R^2\, S'_e = 4\pi (R + D)^2\, S_e \qquad (107)$$

where S'_e is total earth radiation and reradiation per unit area at an altitude D.

$$S'_e = \left(\frac{R}{R + D} \right)^2 S_e \qquad (108)$$

typically for D = 1000 N miles

$$S'_e = \left(\frac{3440}{4440}\right)^2 S_e = .6\ S_e \qquad (109)$$

Since from Eq. 10 Section 1.1, $B_o = \frac{2\,S'}{C}$ where from Eq. 1 Section 1.1 $\frac{2\,S'}{C} = \frac{2\,S'}{C}$ at 1 A.U. then total Earth Bounce at ground level is:

$$0.16\ B_{o_{solar}} \leqq B_{o_{terrestrial}} \leqq 0.52\ B_{o_{solar}} \qquad (110)$$

for D = 1000 N miles above the earth using (108),

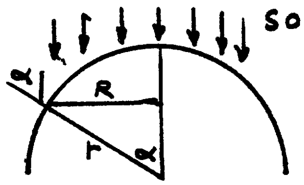
$$0.1\ B_{o_{solar}} \leqq B_{o_{terrestrial}} \leqq 0.31\ B_{o_{solar}} \qquad (111)$$

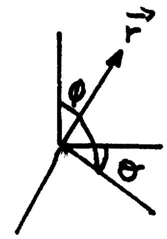
for any distance D from the earth,

$$0.16\ B_{o_{solar}} \left(\frac{R}{R+D}\right)^2 \leq B_{o_{terrestrial}} \leq 0.52 \left(\frac{R}{R+D}\right)^2 B_{o_{solar}} \qquad (112)$$

APPENDIX II

A Derivation of Solar Bounce Forces on a Spherical Solar Sail





Let F_s be the force due to Solar Bounce.

Consider a hoop $2\pi r\,(r d\alpha)$

$$R = r \sin \alpha \tag{113}$$

area of hoop,

$$2\pi r^2 \sin \alpha d\alpha \tag{114}$$

recalling that $\vec{B} = B_o \cos^2 \alpha$

where $\vec{B} = \vec{B}_\theta + \vec{B}_\phi = B_o \cos^2 \alpha \sin \alpha + B_o \cos^3 \alpha$ (115)

$$\int B_\theta \, dA = 0 \quad \text{by symmetry} \tag{116}$$

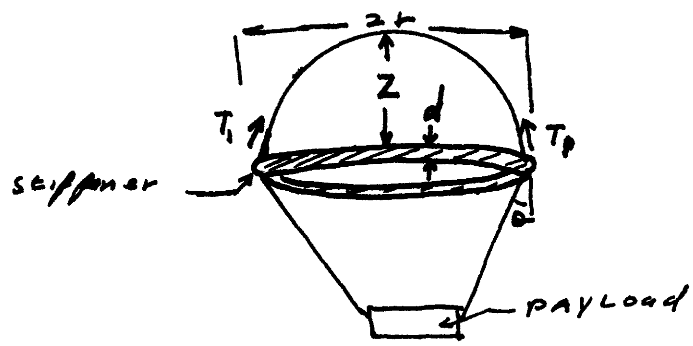
$$F_S = \int B_\phi \, dA = \int_0^{\pi/2} (B_o \cos^3 \alpha)\, 2\pi r^2 \sin \alpha d\alpha \tag{117}$$

$$F_S = [-2\pi r^2 \frac{\cos^4 \alpha}{4}]_o^{\pi/2} \tag{118}$$

$$F_S = \frac{\pi r^2}{2} B_o \tag{119}$$

APPENDIX III

A Derivation of Equilibrium Conditions for a Peripherally Stiffened Sail



$$T_1 \frac{dz}{dr} 2\pi r = \frac{1}{2} B_o \pi r^2 \qquad (120)$$

For $B \approx B_o$

$$T_1 = \frac{B_o r}{\left.\frac{dz}{dr}\right._{max}} \qquad (121)$$

Assume paraboidal shape then:

$$\left.\frac{dz}{dr}\right._{max} = 2 \left.\frac{dz}{dr}\right._{average} \qquad (122)$$

For buckling from Den Hartog Ref. 59a, p. 278.

$$T_{crit} = \text{critical tension} = \frac{3EI}{r^3} \qquad (123)$$

Therefore,

$$\frac{T_{crit}\, r^3}{3E} = I \qquad (124)$$

where:

E = Young's modulus

I = moment of inertia of peripheral stiffner required

r = radius of sail

T = tension in lbs/in.

Combining with Equation (121) and assuming that $\theta \approx 0$, and therefore that payload contribution to radial tension on stiffener is negligible, further assume that: $T_r \approx T_1$

where: T_r = radial tension on stiffener

$$EI = \frac{B_o r^4}{3 \frac{dz}{dr}_{max}} \quad (125)$$

assuming:

r = 125 ft.

z = 2.5 ft.

$B_o = 2 \times 10^{-7}$ psf

$$EI = \frac{\left(\frac{2 \times 10^{-7}}{144}\right) \text{psi} \quad (125 \times 12)^4 \ \text{in}^4}{3(.04)} = 58,000 \text{ lb. in.}^2 \quad (126)$$

for steel $E = 30 \times 10^6$

$$I = \frac{\pi d^4}{64} = .002 \quad (127)$$

therefore,

$$d = 0.8'' \approx 1 \text{ lb/foot length} \quad (128)$$

For a foam filled Mylar ring we calculate the following size:

assume E Mylar = 500 ,000 psi

assume foam = 2 lb/cu ft.

(where foam just prevents Mylar from collapsing)

assume t Mylar = .001 in.

Where t = thickness of Mylar ring of circular cross-section then approximately:

$$I = \pi r^3 t: \qquad (129)$$

using (126)

$$r^3 = \frac{58,000}{E\pi t} \qquad (130)$$

$$r^3 = \frac{58,000}{\pi \times 500,000 \times .001} \qquad (131)$$

$$r^3 = 37^{in.^3} \qquad (132)$$

$$r = 3.3^{in.} \qquad (133)$$

Weight of foam per foot:

$$\frac{\pi(3.3)^2 \ ft^2 \ (2)^{lb/ft^3} \ (1)^{ft}}{144^{in^2}/ft^2} \quad .48^{lb/foot\ length} \qquad (134)$$

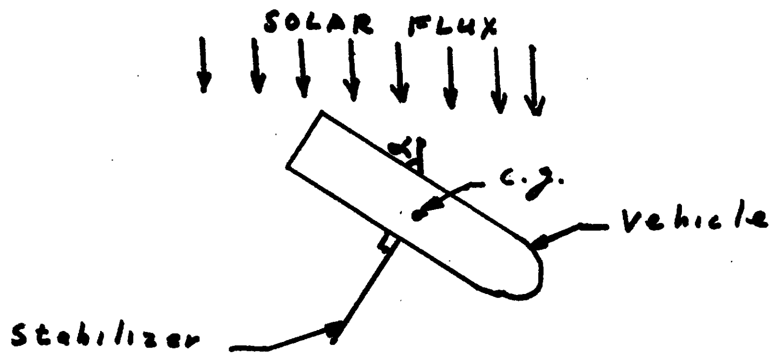
For .001 Mylar we can neglect the weight of the Mylar as compared to the foam.

Therefore weight of stiffener $\cong$ 0.5 lb/ft length

From this example we see that peripherally stiffened sails quickly become excessive in weight if large sizes are used even using low density stiffeners. For a small sail such as is used in the Self-Stabilized Stabilizer since $EI \sim r^4_{sail}$, we see that very reasonable values of stiffener diameter can be obtained.

APPENDIX I.V

A Derivation of Equations Governing the Restoring Force of a Solar Sail Attitude Control



The unbalance torque on a vehicle due to solar bounce is:

$$T_D = B x_1 A = B_o \cos^2 x_1 A_1 \qquad (135)$$

where:

B = Solar Bounce

B_o = Bounce at normal incidence

x_1 = distance between c.g. and center of bounce (pressure)

A_1 = cross-section area of vehicle (assumed in this expression to be coplanar) since force = $\int_0^A B \, dA$

T_D = disturbing torque

$$T_r = S x_2 A_2 B_o \sin^2 a \qquad (136)$$

where:

x_2 = moment arm of stabilizer around the vehicle c.g.

A_2 = stabilizer surface

T_r = stabilizer restoring force

Therefore, equilibrium angle (about which oscillation can take place) is:

$$B_o \cos^2 \alpha \;\; x_1 A_1 = x_2 A_2 B_o \sin^2 \alpha \qquad (137)$$

$$\tan^2 \alpha = \frac{x_1 A_1}{x_2 A_2} \qquad (138)$$

$$\tan \alpha = \left(\frac{x_1 A_1}{x_2 A_2}\right)^{\frac{1}{2}} \qquad (139)$$

APPENDIX V

Calculated Values of Angular Velocity, Angular Momentum, Deflection, and Stress in the Proposed Solar Sail Design

1. Assume design as follows:

   R = 125 ft. (250 ft. diameter)

   at $r = R_o$, let the centrifugal force be 40 times the Solar Bounce.

   This guarantees a negligible amount of "dishing" (z deflection)

2. Calculated required angular velocity ω

$$\frac{\gamma R_o \omega^2}{g} = 40\, B_o \qquad (140)$$

where:

γ = .001 psf

g = 32. ft/sec^2

ω = angular velocity

R_o = 125 ft

$B_o = 2 \times 10^{-7}$ psf

$$\omega^2 = \frac{40\,(2 \times 10^{-7})\,(32)}{.001 \times 125} \qquad (141)$$

$$\omega = .044 \text{ rad/sec} \qquad (142)$$

$$= 0.4 \text{ rpm} \qquad (143)$$

3. Calculated Maximum Stress (using a flat disk approximation)

   from Ref. 59A, p. 57

$$S_{max} = \text{at hub} = S_t = \left(\frac{3+\mu}{8}\right)\frac{\rho\,\omega^2}{g}\left[R_o^2 + R_1^2 - R_o^2 - \frac{1+3\mu}{3+\mu} R_1^2\right] \qquad (144)$$

where:

S_t = tangential stress

μ = poisson's ratio (not known for Mylar, shall omit in calculations)

ρ = density in lb/ft^3

$$S_t \approx 3/8 \times \frac{85\ lb/ft^3}{32\ ft/sec^2} \times .002\ (rad/sec)^2\ (125)^2\ ft^2\ (2) \qquad (145)$$

$$\approx 60\ psf \qquad (146)$$

$$\approx 0.4\ psi \qquad (147)$$

4. Energy Required to Deploy

$$T = Ip\ \dot{\omega}$$

where:

T = torque

Ip = polar inertia

t = time

$$Txt = Ip\ \omega \qquad (148)$$

$$Ip\ \omega = 2\ (\frac{mr^2}{4}):Ip\ \omega = (\frac{200\ lbs.}{32.2})\frac{(125)^2 ft^2}{2}(.044)\ rad/sec \qquad (149)$$

$$Ip\ \omega = 2 \times 10^3\ ft\ lb\ seconds \qquad (150)$$

5. Spin using four-1/4 lb. Thrust Rockets at Edge

We shall assume only a sufficient original spin to deploy sail (instead of the 3600 rpm required to bring it up to speed). We shall determine time required to reach 1 rpm from a negligibly low speed using rockets at R_o = 125 ft. Assume four-1/4 lb. thrust rockets.

$$t = \frac{I_P\,\omega}{T} = \frac{I_p\,\omega}{1 \times 125\ \text{ft}} \tag{151}$$

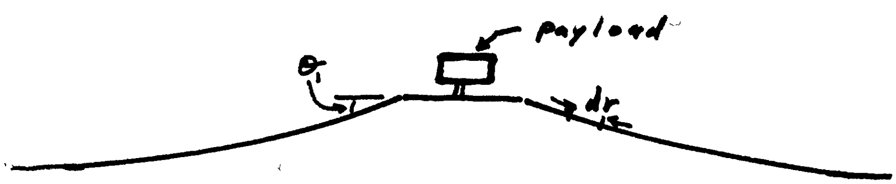
$$t = 32 \text{ seconds} \tag{152}$$

6. Angular Momentum, of Centrifugal Sail

$$H = I_p\,\omega \tag{153}$$

$$\text{from (5)} = 2 \times 10^3 \text{ ft lb seconds} \tag{154}$$

7. Deflection of Sail



Consider equilibrium forces parallel to the sail on a thin ring of width dr.

for $\cos\theta \approx 1$

$$dF = \left(\frac{2\gamma}{g_o}\,\omega\, r^2\right)(-2\pi r\ dr) \tag{155}$$

where:

γ = weight/ft^2

dF = change in tensile stress in membrane between r and r + dr.

$$F = \int_R^{r_1} dF = 2\pi\frac{\gamma\,\omega^2}{3}\,(R^3 - r^3) \tag{156}$$

where:

r_1 = inner radius = 2.2 ft.

R = outer radius = 125 ft.

Consider hub equilibrium:

$$F \sin\theta = Ma \tag{157}$$

where:

$$M = \text{mass of payload} = \frac{200}{g_o} \text{ slugs}$$

$$a = \text{acceleration of vehicle} = 1 \times 10^{-4}\, g_o$$

$$\sin \theta = \frac{Ma}{\frac{2}{3}\pi\, \omega^2\, (R^3 - r^3)} \qquad (158)$$

$$\sin\theta = \frac{\frac{200}{g_o} \text{ slugs} \times 1\text{x}10^{-4} \times g_o}{\frac{2}{3}\pi\left(\frac{.001}{32.2}\right) \text{slugs/ft}^2 (0. \quad \times 10^{-4})(\text{rad/sec})^2 [125^3 - (2.2)^3] \text{ft}^3} \qquad (159)$$

$$\sin\theta = .0795 \qquad (160)$$

$$\theta_1 = \theta_{max} = 4^{o}\ 30' \qquad (161)$$

since,

$$\theta_{max} = 4^{o}\ 30''$$

for any reasonable geometry,

$$\theta_{average} < \frac{4^{o}\ 30'}{3} \qquad (162)$$

Without calculating the equilibrium shape we can say that maximum dishing for r = 125 ft.

$$Z_{max} < 125 \times \frac{.08}{3} = 3.3 \text{ ft} \qquad (163)$$

We can therefore consider the sail as a flat disk (except for dynamic characteristic and flutter considerations discussed in Section 7.2).

APPENDIX VI

Calculations of Available Control Torques and Rate of Precession

The method of payload shift depends on precession for change of attitude. Since the payload inertia is negligible as compared to the sail's we shall ignore the non-rotating payload in this analysis.

For a gyroscope:

$$G = Ip\,\omega\,p \qquad (164)$$

G = couple

ω = spin in rad/sec (0.044 rad/sec)

p = precession in rad/sec

Ip = moment of inertia about spin axis

given:

payload = $\frac{200}{g_o}$ slugs

vehicle acceleration = $1.0 \times 10^{-4}\, g_o \cos^2 \alpha$

moment arm for payload shift of d = 10 ft., then

$$G = (m\,a)d = \left(\frac{200}{g_o}\right) \text{lbs}\ (1 \times 10^{-4}\, g_o)(10^{\text{ft}}) \cos^2 \alpha \qquad (165)$$

$$= 0.2 \cos^2 \alpha \text{ ft lbs.} \qquad (166)$$

for $\alpha = 30^o$ (typical)

$$G = 0.15 \text{ ft lbs} \qquad (167)$$

from (164)

$$P = \frac{G}{Ip\,\omega} = \frac{0.15}{2 \times 10^3} = 0.75 \times 10^{-4} \qquad (168)$$

$$= 0.7 \times 10^{-3} \text{ rpm} \qquad (169)$$

$$0.25^o/\text{min} \qquad (170)$$

APPENDIX VII

Calculations on the Visibility from Earth of the Proposed Solar Sail Design

We shall use the results derived for specular reflection in Ref. 28, by S. H. Dole.

From p. 23, we write:

$$d_{B_e} = \frac{DB}{2} \left(\frac{\pi E_{SB} T_e r_B}{E_r \alpha_{SB}} \right)^{\frac{1}{2}} \qquad (171)$$

where:

d_{B_e} = distance from vehicle to earth (in centimeters)

α_{SB} = angle subtended by sun at vehicle (steradians)

r_B = reflectivity

E_r = minimum illumination required to see vehicle against background

E_{SB} = illumination of vehicle

D_B = diameter of sail (5250 cm)

T_e = earth atmospheric effect

Using Fig. 5 and 7 of the above reference, let:

$E_r = 10^{-17}$ lumens/cm^2 for 100" telescope (from dark side of earth)

$\alpha_{sb} = 2.9 \times 10^{-5}$ steradians at Mars

$E_{sb} = 6$ lumens/cm^2 (at Mars)

$r_b = 1$

$T_e = 0.7$

$$d_{b_e} = \frac{7600}{2} \left[\frac{\pi\ 6\ (0.7)(1)}{10^{-17}\ (2.9 \times 10^{-5})} \right]^{\frac{1}{2}} = [4.5 \times 10^{22}]^{\frac{1}{2}} = 7.6 \times 10^{14} \text{cm} \qquad (172)$$

$$d_{be} = 4 \times 10^9 \text{ N miles} \qquad (173)$$

for viewing from daylight side, let:

$$E_r = 10^{-14}$$

then:

$$d_{be} = 1 \times 10^8 \text{ N miles} \qquad (174)$$

The above values are only order of magnitude, they include no "non-ideal effects". We should also note, however, that with the use of photomultipliers, given knowledge as to the approximate vehicle location, we should be able to substantially improve on the above values.

APPENDIX VIII

Comparison of Mars Transfer Mass Ratio for a Solar Sail vs. a Ballistic Vehicle

	Case 1 - Solar Sail launched at Earth Escape Velocity	Case 2 - Ballistic Vehicle at Minimum feasible Transfer Velocity (from elliptical non-coplanar analysis)	
V_i	36,700 fps 4100 fps losses 40,700 fps	$\sqrt{(40,000)^2 + (14,000)^2}$ 43,200 fps	(175)
$V_i/V_{exhaust}$	4.07	4.32	(176)
mR = mass ratio overall[1] $= e^{V_i/Ig_o}$	60	75	(177)
MR (Mass Ratio)	Stage: I II III 3.75 4 4	I II III 4 4 4.7	(178)
ϵ = structural factor	.05 .05 .05	.05 .05 .05	(179)
$\frac{1}{mR} = \epsilon + \lambda - \epsilon\lambda$	.265 .25 .25	.25 .25 .213	(180)
$\lambda = \frac{\frac{1}{mR} - \epsilon}{1-\epsilon} = \frac{W_{payload}}{W_{initial}}$	$\frac{.21}{.95}$ $\frac{.20}{.95}$ $\frac{.20}{.95}$	$\frac{.20}{.95}$ $\frac{.20}{.95}$ $\frac{.163}{.95}$	(181)
λ overall $= \lambda_1(\lambda_2)(\lambda_3) =$	.0098	.0075	(182)
$\frac{\text{payload solar sail}}{\text{payload ballistic}}$ (2)	1.3	-----	(183)

1. Assuming specific impulse I = 3000 seconds
2. Assuming same take off weight, and considering sail and midcourse correction rockets respectively as "payload".

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