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Analysis of an Arctic Polesitter

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The concept and rationale are presented and the fundamental dynamical requirements set forth for a spacecraft that remains stationary in space above the North or South Pole of the Earth for an extended period of time. The mathematical basis and acceleration characteristics are shown. Performance capability using present-day ion drive technology is evaluated in terms of stay time at the pole and imaging resolution as a function of viewing distance. The analysis shows that a polesitter spacecraft can be maintained without difficulty for one or two years at several (≈5) lunar distances from the Earth, admitting low-resolution (3 km) visual imagery and yet lower resolution (12-70 km) measurements in the infrared regime. Microwave measurements are not practical using today's technology. Sensitivity calculations show that substantial improvement in performance capability must await major advances in available technology.

Introduction and Rationale

LANDSAT, Nimbus, Tiros, and other low-Earth-orbiting spacecraft have demonstrated the potential for gathering valuable Earth applications data from near-Earth space using remote sensing devices. This great potential for weather analysis, Earth resource surveys, and high-resolution imagery is somewhat diluted because of the limited spacecraft time over a region of particular interest and long times to repeat coverage. The desire for more continuous coverage can be satisfied somewhat by using multiple satellites or wide-swath instruments.

Measurements from a geosynchronous equatorial orbit provide an alternative for continuous coverage of a particular region at some cost in resolution due to the greater distance. Coverage is limited primarily to the temperate zone where viewing incidence angles are adequate.

A polesitter spacecraft presents the potential for continuous coverage in the polar regions—a new dimension in applications satellite utilization. Such a spacecraft would remain in a fixed position above the North or South Pole for a year or more, allowing long-term, multi-instrument measurements in the polar region as frequently as desired. At mission end the spacecraft could spiral down to low Earth orbit for possible retrieval, refurbishment or replenishment, and reuse.

This paper presents an analysis of the fundamental dynamical requirements for such a mission and shows the performance attainable using present-day technology. The analysis establishes the mathematical basis, determines the required acceleration characteristics, and shows the propulsion needed to do such a mission. It takes into account perturbations due to lunar and solar gravity with their variation through the year and assesses the impact of solar radiation pressure. A computer program was constructed and used to compute and plot stay time at the pole for a wide range of spacecraft distances and propulsion parameter assumptions.

This analysis contributes to the state-of-the-art in formulating the fundamental polesitter requirements, in showing what can be reasonably expected for polesitter performance today, and indicating technology advances needed for increased attractiveness. Any future analyses of the polesitter concept can well use as a starting point the foundational data presented here.

Mathematical Foundation

The fundamental requirements for a polesitter spacecraft can be expressed (in Earth-centered coordinates) as follows:

$$\bar{R}_{S/C} = \pm R_{S/C}\hat{k}q \qquad \bar{V}_{S/C} = 0 \qquad \bar{A}_{S/C} = 0 \tag{1}$$

where the direction, $\hat{k}q$, distinguishes the polesitter spacecraft from other geostationary spacecraft, and plus and minus signs refer to the North and South Poles, respectively. The spacecraft is positioned at distance $R_{S/C}$ from the Earth center along the Z-axis in geocentric equatorial coordinates. The velocity null establishes the required initial spacecraft conditions at $\bar{R}_{S/C}$. The acceleration null denotes the condition to be maintained throughout the mission. The same conditions in inertial coordinates require:

$$\bar{R}_s = \bar{R}_{s0} + \bar{R}_{S/C} \quad \bar{V}_E - \bar{V}_{S/C} = 0 \quad \bar{A}_E - \bar{A}_{S/C} = 0 \tag{2}$$

so that the spacecraft and the Earth experience identical inertial velocity and acceleration and the spacecraft remains fixed in Earth-centered coordinates.

The accleration condition is expressed geometrically in Fig. 1, assuming inertial and solar coincidence. The spacecraft must accelerate with respect to a virtual sun S' at the same direction and rate as the Earth accelerates with respect to the sun S. Thus the spacecraft moves in an identically shaped orbit as Earth but with respect to focal point S' instead of S.

The desired spacecraft acceleration is

$$\bar{A}_{S/C} = \bar{A}_E = \bar{A}_{s0} = A_{s0}\hat{r}_{s0} = -(\mu_s/r_{s0}^2)\hat{r}_{s0} \tag{3}$$

The actual acceleration acting on the (nonthrusting) spacecraft is

$$A_{S/C} = \bar{A}_s + \bar{A}_e + \bar{A}_m = (A_s + A_{sp})\hat{r}_s + A_e\hat{r}_e + A_m\hat{r}_m \tag{4}$$

where the accelerations are imposed on the spacecraft by the sun $(\ )_s$, Earth $(\ )_e$, and moon $(\ )_m$, respectively. The term A_{sp} refers to solar pressure. The remaining terms are gravitational effects and have the form,

$$A_i = -\mu_i/r_i^2$$

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where $i=s$, e, or m and where μ_i is the gravitational parameter of the attracting body. Distant planet effects are neglected. Also, certain indirect acceleration terms are neglected (e.g., the acceleration of the Earth by the moon) which in effect ignores the negligibly small barycenter displacements.

A portion of the solar acceleration but none of the Earth and lunar accelerations are desired. The perturbing solar acceleration is given by

$$\overline{DA}_s = \bar{A}_s - \bar{A}_{s0} \tag{5}$$

The total perturbing acceleration is

$$\overline{DA} = \overline{DA}_s + \bar{A}_e + \bar{A}_m \tag{6}$$

The perturbing acceleration must be counteracted by continuous thrusting to satisfy the polesitter fundamental acceleration requirement.

Acceleration Characteristics

Dependence on Spacecraft Distance

The perturbing acceleration of Eq. (6) varies with spacecraft distance from the Earth, ecliptic longitude of the Earth (i.e., time of year), and lunar true anomaly in descending order of importance. Figure 2 shows the dependence of perturbing acceleration DA on distance from the Earth center with ecliptic longitude of the Earth, θ, constant at 90 deg. The acceleration is quite high when the spacecraft is near the Earth and drops as the inverse square with increasing distance ($DA \approx A_e$).

The perturbing solar acceleration DA_s varies linearly from zero near the Earth to become predominant above 300 Earth radii distance. The combined effects of Earth and solar perturbing accelerations produce an acceleration minimum at 380 Earth radii. The acceleration varies from 2.74 mm/s^2 at lunar distance (60 Earth radii) to 0.163 mm/s^2 minimum at 380 Earth radii (6.3 lunar distance). The thrust level required to neutralize perturbing acceleration at lunar distance is therefore 16.8 times that at the minimum. Clearly, the operating distance has substantial impact on polesitter propulsion requirements and the requirement is less for greater operating distance.

Dependence on Earth Longitude

Dependence on Earth longitude would be barely discernible on the scale of Fig. 2 and is shown in Fig. 3 for values of distance near the acceleration minimum, i.e., at a spacecraft distance of 380 Earth radii. At any given spacecraft distance from the Earth $R_{S/C}$ the acceleration varies throughout the year as the ecliptic longitude of the Earth, θ, changes. The total amplitude swing of this cyclic acceleration with θ is about 0.021 mm/s^2 for $R_{S/C}=225$ Earth radii and drops near parabolically to lower values at higher or lower $R_{S/C}$ (to 0.010 mm/s^2 at $R_{S/C}=70$ Earth radii, and again around $R_{S/C}=380$ Earth radii at the acceleration minimum). At $R_{S/C}$ less than 380 Earth radii acceleration minima occur at $\theta=90$ and 270 deg. Above ~ Earth radii, minima occur only at $\theta=270$ deg. Fourier analysis can be used to obtain the following approximate analytic expressions for the envelope of acceleration minima [Fig. 4 and Eq. (7)] and the distances at which they occur [Fig. 5 and Eq. (8)].

$$\begin{aligned}\mathrm{ACMIN} &= 0.1595 + 0.0034 \sin(\theta/2) + 0.0039 \sin\theta \\ &\quad + 0.002 \cos 2\theta + f(R_m) \qquad (\mathrm{mm/s^2})\end{aligned} \tag{7}$$

$$\mathrm{RMIN} = 400. - 8 \sin\theta + 31.4 \cos 2\theta \quad \text{(Earth radii)} \tag{8}$$

These equations are useful for determining the minimum permissible thrust level at any time of year and the spacecraft operating distance where this thrust level is needed. An operating point above or below this minimum thrust distance

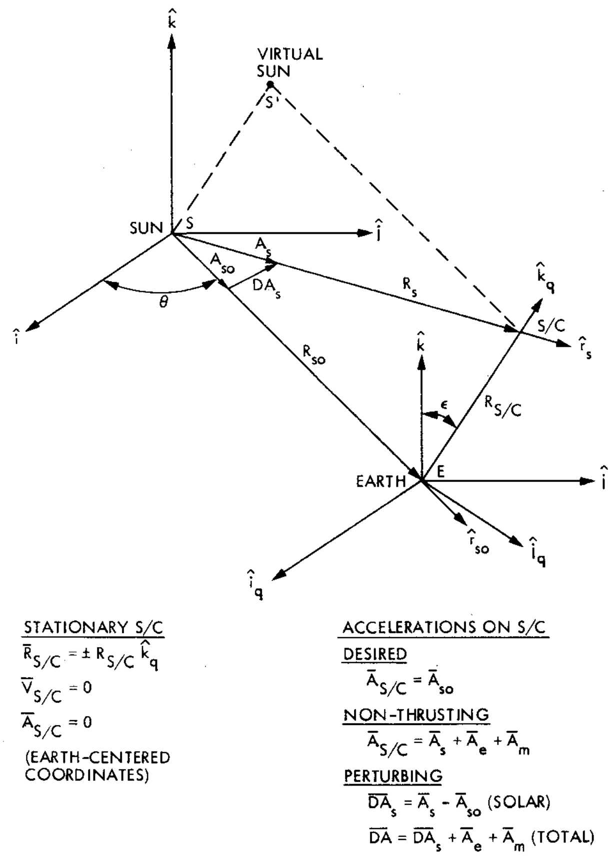

Fig. 1 Geometry for a polesitter.

Table 1 Diffraction-limited resolution

Frequency regime	Radiation frequency f, Hz	Radiation wavelength λ, μ	Aperture diameter α, m	Angular resolution ϕ, μrad	Linear resolution d, km	Viewing distance $R_{S/C}$ 10^6 km	Lunar distances
Visual	5.3×10^{14}	0.57	0.15	10.5[a]	20	1.9	5.0
	5.3×10^{14}	0.57	0.40	1.7	3.3	1.9	5.0
	5.3×10^{14}	0.57	1.00	0.7	0.5	0.7	2.0
Infrared	1.5×10^{14}	2.0	0.40	6.1	12	1.9	5.0
	2.5×10^{13}	12.0	0.40	36.6	70	1.9	5.0
Microwave	1000×10^{9}	300	7.0	52.3	100	1.9	5.0
	6×10^{9}	50,000	1000	61.0	117	1.9	5.0

[a] Distortion-limited resolution.

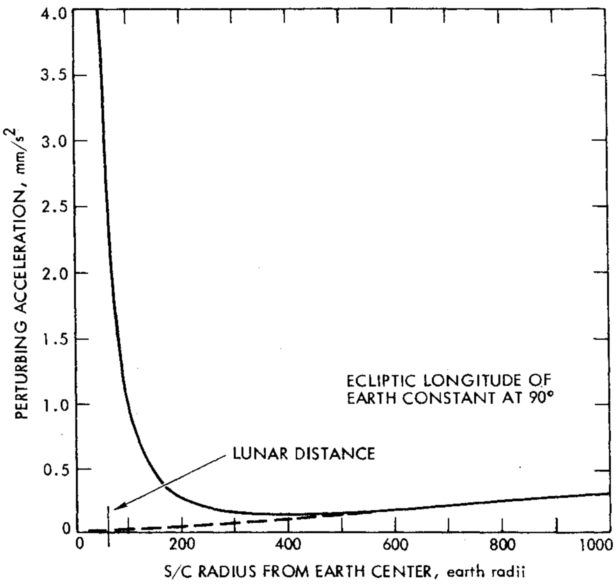

Fig. 2 Acceleration dependence on distance from Earth.

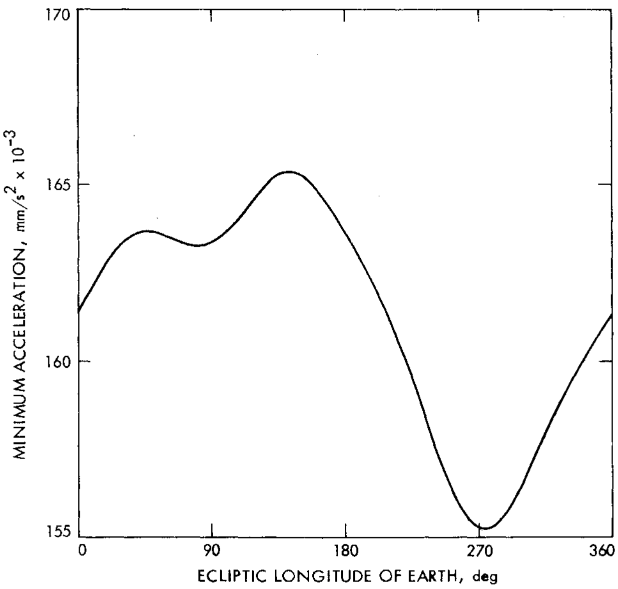

Fig. 4 Envelope of acceleration minima.

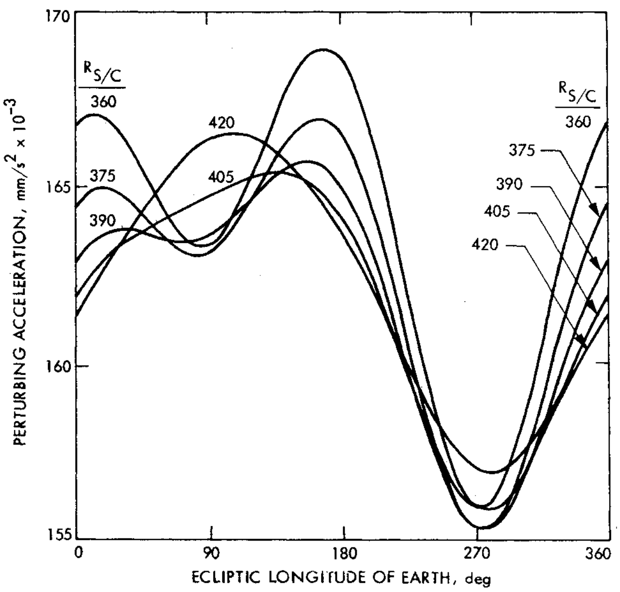

Fig. 3 Acceleration dependence on Earth longitude.

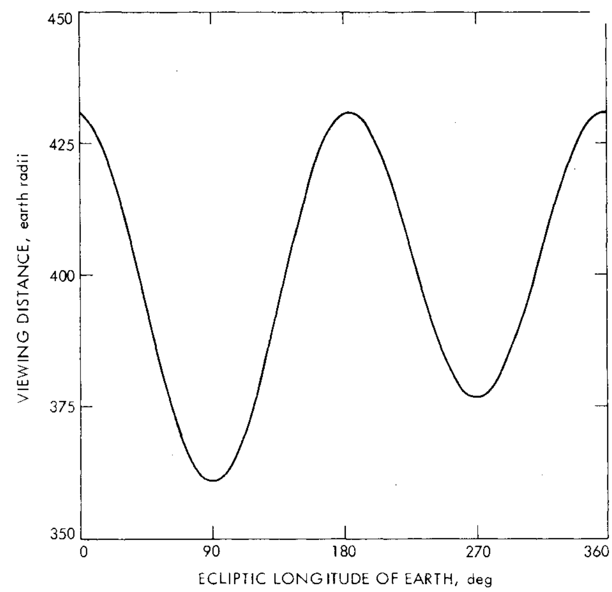

Fig. 5 Spacecraft distance at acceleration minimum.

may be chosen for other reasons at a cost of higher thrust.

In Eq. (7), $f(R_m)$ is the variation in the lunar perturbing acceleration, which can be ignored to first order.

Lunar Component Behavior

Lunar perturbing acceleration, shown in Fig. 6, varies roughly with the inverse square of $R_{S/C}$ but remains on the order of 0.5-1% of total perturbing acceleration for all values of $R_{S/C}$ below the acceleration minimum.

Figure 7 shows the variation in acceleration across lunar period for various values of $R_{S/C}$. This variation will appear as a periodic ripple on the perturbing acceleration. The ripple is pronounced at lower $R_{S/C}$ values ($\approx 40\%$ of A_m at $R_{S/C} = 150$ Earth radii) but becomes negligible for $R_{S/C}$ near the acceleration minimum.

Lunar acceleration effects were considered in this analysis but they exert minimal impact on polesitter performance.

Effect of Solar Pressure

A 5.0-N/km² solar pressure at 1.0 a.u. causes a perturbing force of 50×10^{-6} N/kW, assuming a specific installed power capability of 100 W/m² before degradation. At 100 kW of installed power, the perturbing solar pressure force would be 5×10^{-3}N, more than two orders of magnitude less than minimum nominal thrust, and is neglected in this analysis.

Thrust Pointing Direction

The thrust pointing direction will always be diametrically opposite the direction of the perturbing acceleration $\overline{DA}$. Figure 8 shows how thrust pointing direction varies with Earth longitude, for various values of spacecraft distance $R_{S/C}$. The radial axis shows thrust latitude on the polar plot and the angular axis shows thrust longitude. The angles are specified with respect to Earth equatorial coordinates centered at the spacecraft.

At 90 deg latitude, thrust is directly away from the pole, posing a potential measurement problem for instruments viewing the Earth through the thruster exhaust beam. Thrust pointing remains near the zenith for all Earth longitudes θ when $R_{S/C}$ is 100 Earth radii or below. At distances from Earth greater than 200 Earth radii, a potential viewing problem occurs only when Earth longitude is near 0 or 180 deg. When $R_{S/C}$ equals 300 Earth radii, thrust latitude is within 5 deg of zenith for only three weeks in March of each

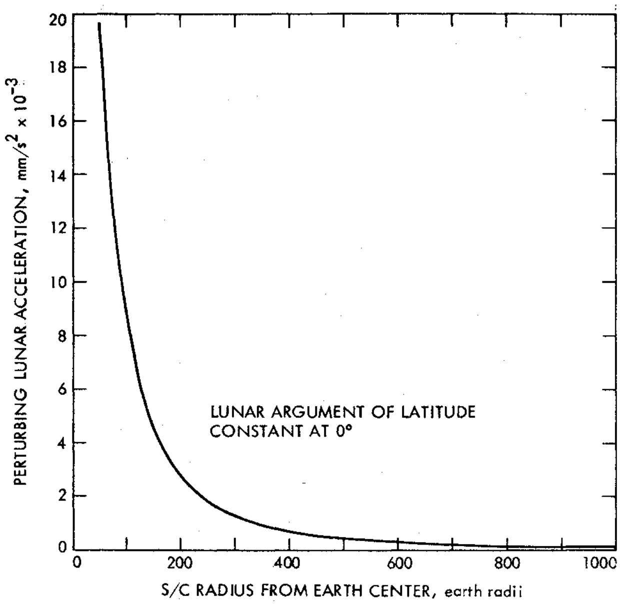

Fig. 6 Lunar perturbing acceleration.

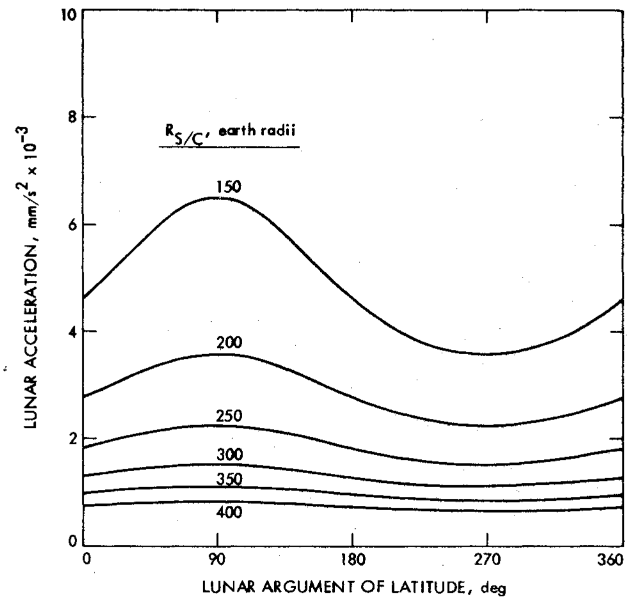

Fig. 7 Acceleration dependence on lunar position.

year and another three weeks in September. Still greater spacecraft distance from Earth offers a more attractive operating point for avoiding a viewing problem through the thruster exhaust.

Propulsion Requirements

Propulsion Mode Description

Continuous propulsion is needed to maintain a fixed spacecraft position above the pole. The thrust provided must give an acceleration that is equal and opposite the perturbing acceleration $\overline{DA}$. One or two years operating time dictates a high specific impulse to limit the amount of propellant required. The propulsion characteristics required suggest solar electric or nuclear electric propulsion. Solar Electric Propulsion (SEP) is assumed for this analysis. The power level needed depends on the perturbing acceleration magnitude at the desired operating distance. Equations modeling the propulsion system are

$$T = 2F_I P_B / c \tag{9}$$

where the available thrust T is proportional to the beam power P_B and inversely proportional to the exhaust beam velocity c. The dimensionless power reserve factor F_I provides a margin for unscheduled thrust deviations. The perturbing acceleration is

$$DA = T/M_0 \tag{10}$$

where

$$M_0 = M_n + M_{ps} + M_p + M_{\rm res} \tag{11}$$

In Eq. (11), M_n is the net spacecraft mass required to carry viewing instruments and all spacecraft subsystems except for propulsion. M_{ps} is the mass of the total propulsion system including solar array, power conditioners, thrusters, and all associated tankage and structure. M_{ps} is dictated for a given system by the power level required. M_p is the propellant needed to hold the spacecraft at the desired position for the operation time required. M_p is highly dependent on c. $M_{\rm res}$ is the propellant held in reserve to insure that the spacecraft can be returned to Earth in the manner desired at the end of the mission. If economically justified, sufficient propellant can be left to allow the spacecraft to spiral down to low Earth orbit over a several-month period where the spacecraft can be retrieved, refurbished/replenished, and reused. The summation M_0 is the initial spacecraft mass at the pole. The spacecraft is launched by the Shuttle/IUS into a highly elliptic orbit. The SEP system, initiated near apogee, completes the transfer to the pole.

Spacecraft mass at the end of the stay is

$$M_f = M_n + M_{ps} + M_{\rm res} \tag{12}$$

where all propellant except the reserve has been used.

Operating Mode Selection

Two operation modes are possible. A constant acceleration mode provides analysis simplicity but results in a continuously varying spacecraft operating distance which may be undesirable for image analysis. Alternatively, a constant spacecraft operating distance would provide relatively fixed imaging distance but is less amenable to simple analysis. Neither operating mode would impact greatly the amount of propulsion control required nor the propulsion performance capability since continuous small thrust vector variations are required in either case. The constant acceleration mode is assumed here for analysis simplicity.

Operating Point Selection

A tradeoff is needed to choose the best distance above Earth for an operating point. The minimum acceleration distance (6-7 lunar distances) minimizes the propulsion requirement. A lower operating distance provides better imaging resolution. A higher operating distance provides a safer region for recovering from unexpected altitude excursions or power level degradation. The choice for further analyses favors the lower altitude region which provides improved imaging resolution at a somewhat larger cost in propulsion requirement. A margin of safety from catastrophic altitude or power level excursions is assured by normal operation at less than the available power level. With a 2% power reserve, all thrusters could be deliberately shut off for up to eight days and then restarted without the spacecraft falling beyond the safe recovery region.

Propulsion Performance Criteria

When acceleration a is held constant for the duration of the stay at the pole, the stay time t is given by

$$t = (c/a)\ln(M_0/M_f) \tag{13}$$

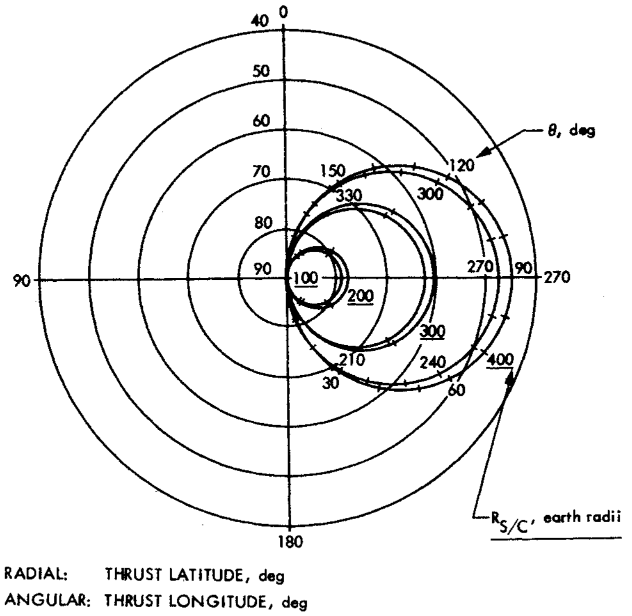

Fig. 8 Thrust pointing direction.

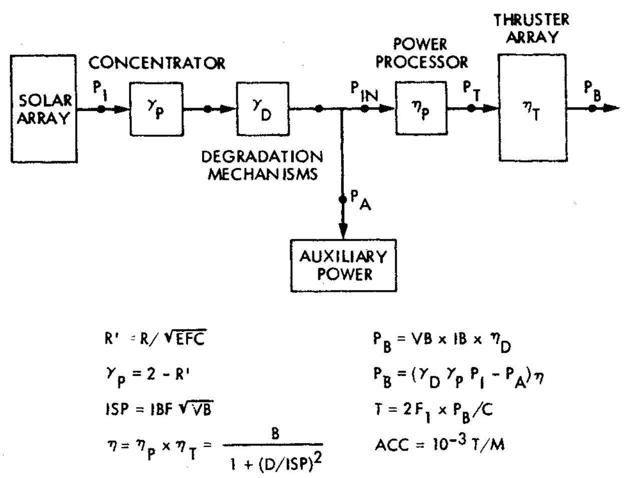

Fig. 9 Polesitter propulsion system model.

Alternatively, if the desired stay time is specified and acceleration is constant, the reserve propellant at mission end is

$$M_{\rm res} = M_0 \exp(-at/c) - M_n - M_{ps} \tag{14}$$

In either case, the propellant used is

$$M_p = M_0 - M_0 \exp(-at/c) \tag{15}$$

The required power level decreases continually throughout the mission. This behavior results from maintaining the relatively constant acceleration appropriate for a fixed polar position while spacecraft mass is being continually reduced by the amount of propellant used. The power level behavior imposes a requirement for temporary beam voltage or beam current reduction over a limited range, then reduction in the number of operating thrusters.

The decreasing power requirement will provide increasing reserve thrusting capacity without the need for spare power processors and thrusters.

Propulsion System Modeling Assumptions

The propulsion system modeling for this analysis assumes 1977 year-end propulsion technology.[1,2] Further studies of this concept should use revised assumptions appropriate to technology advancements.

Thrust Beam Power

The propulsion system model assumed for this analysis is shown in Fig. 9. The power P_I output from the basic solar array is modified by the power ratio γ_p as shown in Eq. (16) where R is solar distance, $R_0 = 1.0$ a.u., and EFC is the effective concentration. (EFC is greater than unity when solar reflectors (concentrators) are used to increase the effective solar energy collection area.) Power level is further modified by the degradation factor $\gamma_D = 0.84$ to account for cell breakage, radiation damage and other mechanisms which reduce available power. Subtracting auxiliary power P_A results in thruster input power $P_{\rm in}$.

$$P_{\rm in} = \gamma_p \gamma_D P_I - P_A \tag{16}$$

where

$$\gamma_p = 2 - R/(R_0\sqrt{\rm EFC})$$

The power conditioner efficiency η_p and the thruster efficiency η_T give a combined efficiency η, assuming appropriate dimensions for the constants with V_B in volts:

$$\eta = \eta_p \eta_T = \frac{0.778}{1 + (1549/87.45\sqrt{V_B})^2} \tag{17}$$

The resulting beam power is given by

$$P_B = \eta P_{\rm in} \tag{18}$$

Operating the propulsion system at a factor $F_I = 1.02$ above the nominal beam power required [Eq. (9)] provides a safety margin for unexpected altitude or power level excursions.

Beam power is related to beam voltage V_B and beam current I_B through

$$P_B = \eta_D V_B{}^* I_B \tag{19}$$

where $\eta_D = 0.83$ results from beam divergence and multiple ion production.

Propulsion System Mass

Propulsion system mass M_{ps} has two principal components: the solar array subsystem mass M_{sa} and the thruster subsystem mass M_{ts}. In each case, subsystem mass depends on the power level used. It is instructive to express the dependence as follows, where the numbers are empirically derived:

$$M_{sa} = -210 + 175\ {\rm EFC} + (12.0 + 5.17\ {\rm EFC})P_I \tag{20a}$$

$$M_{ts} = 383 + 62.5N_T + (3.935 + 0.1225N_T) \cdot [(V_B/500)N_T + 0.5N_T - 16] \tag{20b}$$

$$M_{ps} = M_{sa} + M_{ts} \tag{20c}$$

where N_T is the number of thrusters installed. EFC = 1.0 (no concentrators) was used for final performance analysis since higher values resulted in performance loss for the polesitter.

Propulsion Parameter Selection Constraints

The payload delivered to ballistic orbit apogee is constrained to 5700 kg maximum in this analysis to satisfy an assumed interim upper stage (IUS) structural design constraint. This limiting mass is ample for the cases considered and no launch energy constraint is imposed on payload capability.

Lifetime and performance characteristics have been demonstrated for 30 cm mercury ion bombardment thrusters[3] and their use is assumed in this analysis. Lifetime is about 15,000 h (1.7 yr). Without redesign, beam current up to 2.0 A and beam voltage between 1100 and 3200 V can be used. The

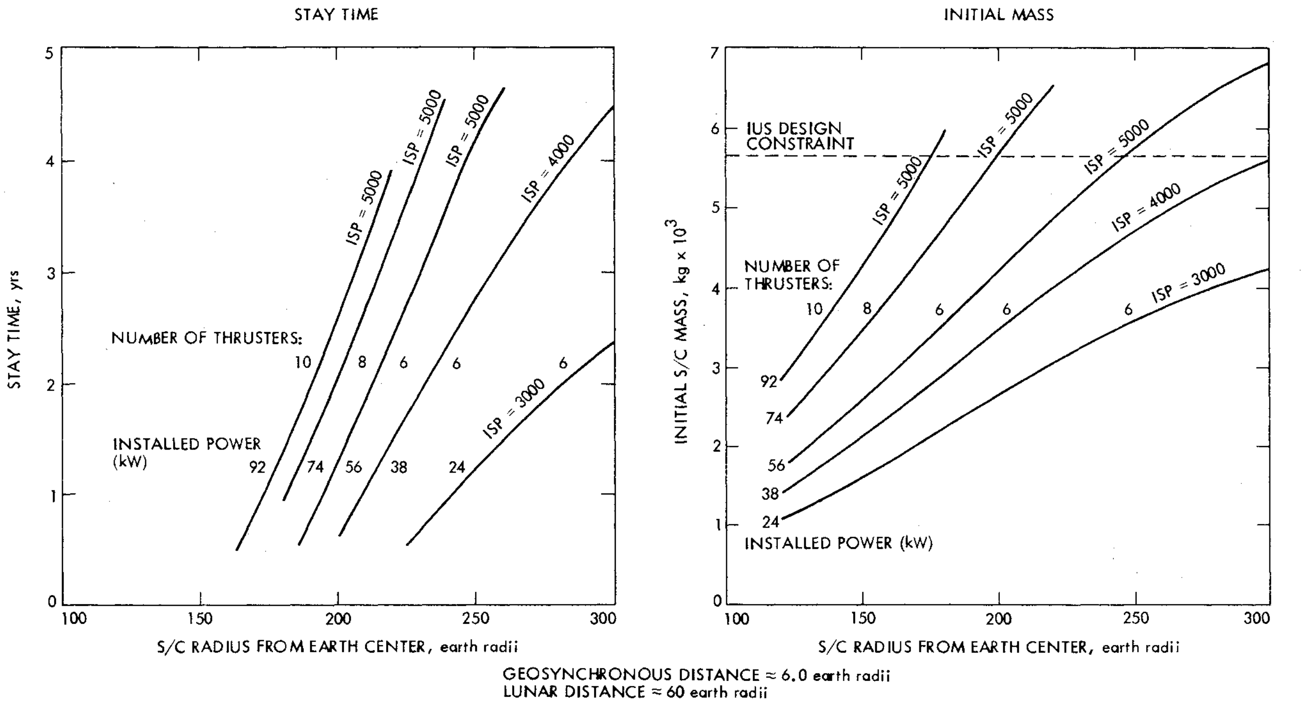

Fig. 10 Polesitter performance.

corresponding specific impulse ranges from about 3000 to 5000 s. Array configurations containing up to ten thrusters have been sized for the Space Shuttle without exceeding cargo bay dimensions. This analysis allows a maximum of ten thrusters in the propulsion system.

Installed power is not specifically constrained in this analysis but levels above 100 kW are considered excessive. The maximum power level considered (92 kW) results from using ten thrusters and 5000 s specific impulse.

A reserve propellant of 500 kg is used throughout the analysis. This propellant quantity should permit the spacecraft to spiral down to low Earth orbit at mission end for recovery, using the Space Shuttle. The craft can then be replenished with fuel or refurbished as appropriate and returned to service.

Polesitter Performance

Stay Time in Orbit

Performance is expressed here in terms of the length of time a given net mass can be maintained at various operating distances when using a specified propulsion system. Figure 10 summarizes this performance for a net spacecraft mass M_N of 1000 kg and also shows the corresponding initial mass the IUS must deliver to apogee, assuming 200 kg for SEP transfer to the pole.

A ten-thruster system operating at 5000 s specific impulse can accommodate 92 kW of installed power. (This is the most powerful propulsion system available under the specified constraints.) This system can maintain 1000 kg net mass for one year at a 173 Earth radii distance—about 3 lunar distances. This is the minimum operating distance using current technology. The acceleration magnitude at this distance is 0.35 mm/s^2.

At 2 lunar distances, acceleration magnitude is about 0.70 mm/s^2, which would require a power level that is just about double the maximum attainable power level under the given constraints. Note also that staying longer than one year at the 173 Earth radii distance would require a spacecraft initial mass greater than the IUS structural design constraint.

Operating at or closer than 3 lunar distances pushes propulsion system requirements to the limit. Operating at higher distances very rapidly relieves propulsion system requirements. Eight thrusters will provide the same 1-yr stay time at 182 Earth radii or 2 yr at 200 Earth radii.

Using 24 kW of installed power allows a 1-yr stay at 4 lunar distances and 2.3 yr at 5 lunar distances. Six thrusters are sufficient, and ISP can be reduced to 3000 s. Initial mass is well within IUS design capability. This is an attractive propulsion system size and can be readily mechanized using current technology. Noting from Fig. 8 that thruster pointing is also more acceptable at higher distances, the 5 lunar radii (300 Earth radii) distance looks like an attractive operating point, and is the selected point for assessment of nominal performance.

Measurement Capability

The limits of usefulness for the polesitter spacecraft will doubtless be determined by requirements on measurement accuracy for the terrestrial data of interest. It appears certain that visual and infrared measurements will be practical in the kilometer resolution range. Hence, measurement of cloud cover, temperature profiles, surface albedo, ice extent, and thermal balance will be feasible with some limitation depending on weather conditions. However, at the desired operating distance, aperture size requirements imposed by the optical diffraction limit seem to prevent practical measurement on Earth in the microwave frequency range and prevent visual and infrared measurements to less than about 1 km resolution.

The resolution limit[4] imposed by optical diffraction on a telescope type instrument is given by

$$\phi = 1.22\lambda/\alpha \tag{21}$$

where λ is the wavelength of incident radiation, α is the instrument aperture diameter, and ϕ is the minimum possible angular resolution—the minimum distinctly measurable angle subtended at the aperture by the object to be measured. At the desired spacecraft operating distance $R_{S/C}$ the limiting linear resolution is

$$d = (1.22\lambda \cdot R_{S/C})/\alpha \tag{22}$$

Table 1 lists the attainable angular and linear resolution at distance $R_{S/C}$ for various aperture sizes in the visual, infrared, and microwave frequency ranges.

In the visual range ($\lambda \approx 0.57\mu$), a resolution of less than 20 km is readily obtained at $R_{S/C} = 5.0$ lunar distances using off-the-shelf hardware. An improved visual instrument with 40-cm-diam aperture and properly corrected for lens aberrations can provide a 3-km resolution at the same operating distance. This seems the nominal resolution to expect for the polesitter spacecraft.

If aperture diameter is increased to 1.0 m and operating distance is lowered to 2 lunar distances, resolution will be 0.5 km. Each of these improvements will require significant technology extension. Hence, resolutions much less than 1 km are not expected in the visible range. As an aside, at ultraviolet frequencies, finer resolution is possible.

In the infrared frequency range, the longer wavelengths will degrade the available resolution from visible capability. Typical infrared measurements ($\lambda = 2\text{-}12\mu$) should yield resolutions of 12-70 km without major technology extension.

In the microwave regime at the highest possible frequency ($f = 1000$ Ghz, $\lambda = 300\mu$), a 100-km resolution requires a 7-m-diam aperture. While this size aperture would be unrealistic for an optical lens system on a spacecraft, it is not beyond reason for a microwave system. However, typical microwave measurements are at much lower frequencies. Wavelengths as long as 5 cm ($50{,}000\mu$) may be necessary to ensure penetration of the atmosphere to measure surface phenomena. Measurement at this wavelength will require a 1000-m-diam aperture for 117-km resolution. A resolution more gross than 100 km is not expected to elicit much interest. Also a spaceborne antenna this size will clearly require technology advancements and requires further investigation to assess feasibility.

Summary

It has been shown that using present-day ion drive technology, a polesitter spacecraft can be maintained for one or two years at several lunar distances from Earth, taking measurements of the Earth polar region in the visible and infrared frequency spectra at a surface resolution of 3-100 km. Measurement in the microwave spectrum is not practical. The major drawbacks of the polesitter concept are the continuous propulsion requirement and, more seriously, the diffraction limit encountered when attempting fine-resolution measurements.

The polesitter concept was found to be very attractive for continuous measurements in the Earth polar region and capable of providing valuable infrared and visual data when kilometer-scale resolution is acceptable and when weather conditions are favorable. Some other concept is needed, however, when fine resolution is desired.

Imaginative future concepts may well be advanced to get around the diffraction limit problem and future technology may permit microwave measurements as well. At present, however, measurement at finer resolution seems more promising using alternative concepts at some sacrifice in the continuous coverage desired.

Acknowledgments

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