Considerations of the Solar Probe

By

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(With 7 Figures)

Abstract — Zusammenfassung — Résumé

Considerations of the Solar Probe. A solar probe to investigate the solar atmosphere is considered with respect to propulsion, thermal design, component temperature limits, attitude stabilization and telemetry. A selected set of probe experiments are reviewed, and a suitable probe shown schematically. It is concluded that the present state of the rocket art limits the distance of closest approach to the Sun to the orbit of Mercury, but that available probe components would permit design of a probe capable of reaching within 5 million miles of the solar surface.

Betrachtungen über ein Sondenfahrzeug zur Sonne. In der vorliegenden Arbeit wird ein Sondenfahrzeug zur Erforschung der Sonnenatmosphäre erörtert, und zwar hinsichtlich des Antriebs, der thermischen Konstruktion, der Temperaturgrenzen, der Lagestabilisierung und des Fernmeldesystems. Eine Auswahl von Sondenexperimenten wird geprüft und ein geeignetes Sondenfahrzeug schematisch dargestellt. Der Verfasser schließt, daß der gegenwärtige Stand der Raketentechnik die Entfernung einer größten Annäherung an die Sonne bis zur Bahn des Planeten Merkur begrenzt, daß aber gewisse Komponenten des Sondenfahrzeuges, die schon jetzt verfügbar sind, die Konstruktion eines Sondenfahrzeuges gestatten würden, das sich der Sonnenoberfläche bis auf etwa 8 Millionen Kilometer nähern könnte.

Considérations relatives à une sonde solaire. La conception d'une sonde exploratrice de l'atmosphère solaire est examinée des points de vue: propulsion, limitations thermiques, stabilisation d'attitude et télémétrie. Un choix d'expériences est décrit et un schéma de sonde adéquate présenté. L'auteur conclut que dans l'état actuel de la fuséonautique l'orbite de Mercure marque la limite d'approche du soleil. Cependant l'équipement dont on dispose actuellement permettrait la conception d'une sonde pouvant approcher la surface du soleil à une distance de 5 millions de miles.

Introduction

It has recently become apparent that the Earth is immersed in the solar atmosphere. In the vicinity of the Earth this atmosphere consists largely of ionized hydrogen of a density of about 400 particles/cc, at a temperature of about 500,000° K. During an eclipse the corona or the excited atmosphere close to the Sun may be observed. The excitation creates the appearance of a gigantic dahlia or aureola whose petals have been observed to extend a distance of 24 radii from the Sun [1]. The temperature in this area is about 1,500,000° K.

The solar probe offers an opportunity for investigating the solar atmosphere directly and for conducting a number of other investigations of the Sun and the various environmental factors associated with it. The purpose of this paper

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is to review the major problems of the probe to determine critical factors of probe design and to determine the feasibility of conducting such investigations at the current "state of the art".

The Ideal Velocity and Vehicle Propulsion

The "ideal" velocity for descent towards the Sun from the Earth's surface is

$$V_i = V_o - V_a + V_e + V_d + V_g - V_r - V_p \qquad (1)$$

V_o = Velocity of Earth in orbit
V_a = Apogee velocity of descent ellipse
V_e = Velocity of escape from Earth
V_d = Velocity loss due to air drag
V_g = Velocity loss due to gravity (during burning)
V_r = Velocity of Earth's rotation
V_p = Velocity change due to perturbing objects.

The orbital velocity of the Earth varies slightly over the year, and is nearly 30 km/sec [2].

It may be shown that the minimum energy descent path from an orbit to an inner point is by a co-tangent ellipse. For this, the velocity at apogee, the start of descent, is

$$V_a^2 = V_e^2 \left(\frac{1}{k} - \frac{1}{m+k}\right) \qquad (2)$$

V_e = Velocity of escape of parent body
k = Ratio, radius of apogee of the cotangent ellipse to radius of parent body
m = Ratio, radius of perigee of cotangent ellipse to radius of parent body
= per unit radius of closest approach.

For k equal to the Earth's radius, the value of V_a, and of $V_o - V_a$ are shown as a function of m, the distance of closest approach, in Fig. 1.

The velocity of escape from Earth is taken as 11.3 km/sec. The drag loss for typical vehicles may be assumed to be 0.25 km/sec, and the gravity loss to be 0.75 km/sec. It is assumed that the available component of Earth's rotation is the velocity of rotation times the cosine of the ecliptic angle, or 0.43 km/sec.

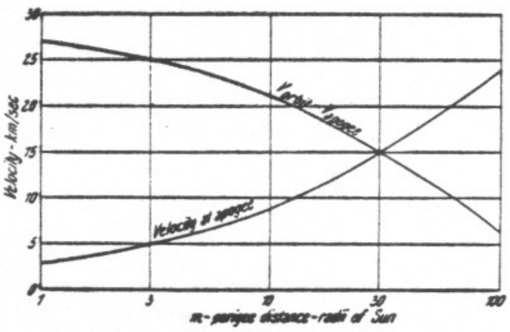

Fig. 1. Velocity for descent from Earth's orbit towards the Sun (values of V_a, $V_o - V_a$)

Perturbation velocities may be introduced by the Moon, Venus or Mercury, and may increase or decrease the effective velocity. Considerations of effectiveness of control systems indicate that it would be undesirable to permit perturbations by the planets. However, it is possible to use the perturbation of the Moon to reduce the ideal velocity by a maximum of 1.49 km/sec [3].

This maneuver will be assumed.

With these assumptions, the ideal velocity required to reach a given distance of closest approach is shown in Fig. 2.

It appears that the current state of the art in propulsive vehicles can be reasonably well represented by assuming that the propulsion system has a high altitude specific impulse of 275 seconds, and that the structural factor $\frac{\text{weight structure}}{\text{weight fuel}}$ is equal to 0.1. These values have been exceeded in liquid propellant units and attained in a few instances in solids.

Assuming that a five stage unit is the limit of practical design, it is found that a 250,000 pound takeoff weight is required to place a 50 pound payload at the distance of Mercury. A closer approach is possible by increasing the number of stages or the takeoff weight. However, since the Mercury approach weight is close to the reported takeoff weights of at least two of the largest rockets now in being, it appears that this may be taken as the reasonable limit of the current state of the art.

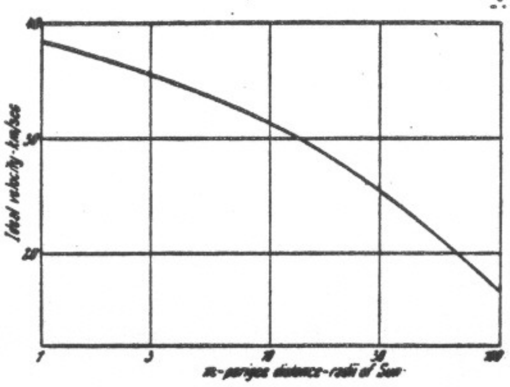

Fig. 2. Ideal velocity to reach a given closest approach to the Sun

A reasonable approach to this five stage design would use liquid propellant units for the first two stages, and solid propellants for the last three. All maneuvering would be accomplished in the early stages. It appears that guidance would be required for all stages if the Moon's perturbation is used. If this is eliminated it might be possible to eliminate guidance in the solid propellant stages by using spin.

Detail system studies would be required to determine suitable means of using available rockets, the adaption of control systems to the problem, and so on.

Thermal Design

The incident power on the probe vehicle will be:

$$P_i = \sigma\, e_i\, T_i^4\, A_i \left(\frac{r}{R}\right)^2 \quad \text{watts/cm}^2 \tag{3}$$

σ = Stefan-Boltzmann constant
e_i = Emissivity for radiation at temperature
T_i = Solar temperature
A_i = Absorbing area
r = Radius of Sun
R = Radius of vector of body.

Current estimates indicate that the Sun radiates nearly as a black body at a temperature very close to 6000° K. The value of P_i for $e_i = 1$ is shown in Fig. 3 for this temperature. (This is the value of the solar constant.)

The power radiated by a body at a temperature T_r is

$$P_r = \sigma\, e_r\, T_r^4\, A_r \text{ watts/cm}^2. \tag{4}$$

The temperature of the body assuming no internal generation of heat and high internal conductivity is:

$$T_b = 6000 \sqrt[4]{\frac{e_i}{e_r} \cdot \frac{A_i}{A_r} \cdot \left(\frac{r}{R}\right)^2}. \tag{5}$$

For black or grey bodies the ratio e_i/e_r is unity. Other surfaces exhibit selective absorption. For example, for magnesium oxide, the value of e_i (for solar radiation) is 0.08, while the value of e_r would be 0.8 if the body temperature were 250° K [4].

The ratio A_i/A_r depends on the geometry of the body and on the distance from the Sun. For a flat plate the same on both surfaces $A_i/A_r = 1/2$. For a sphere of

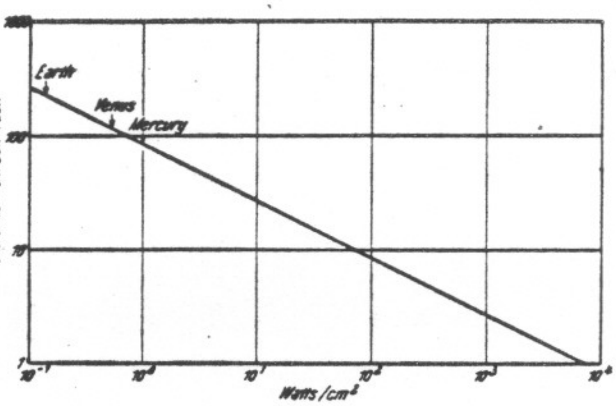

Fig. 3. Solar constant

uniform surface the value is $1/4$ for large distances, and $1/2$ at the Sun's surface. Over the range of interest it does not depart sensibly from $1/4$.

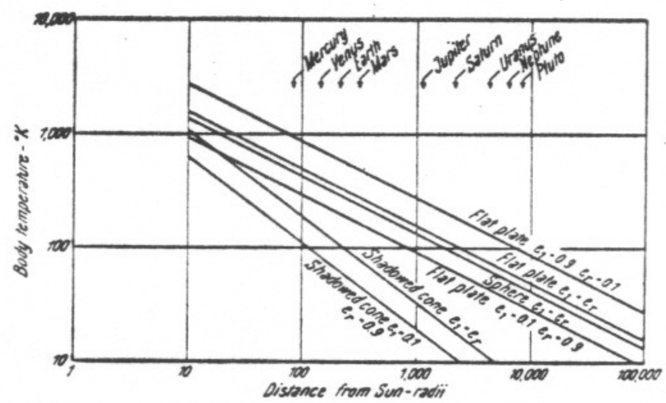

Fig. 4. Equilibrium temperatures

A possible shape is a cone with the base toward the Sun, and with the cone angle chosen so the sides are always shadowed. For this the area ratio is approximately equal to r/R. The equilibrium temperature relations of eq. (5) for these three cases are:

For flat plate

$$T_b = 5050 \sqrt[4]{\frac{e_t}{e_r} \cdot \left(\frac{r}{R}\right)^2}. \tag{6}$$

For sphere

$$T_b = 4250 \sqrt[4]{\frac{e_t}{e_r} \cdot \left(\frac{r}{R}\right)^2}. \tag{7}$$

For shadowed cone

$$T_b = 6000 \sqrt[4]{\frac{e_t}{e_r} \cdot \left(\frac{r}{R}\right)^2}. \tag{8}$$

The equilibrium temperature is plotted in Fig. 4 as a function of distance for several body types. Over the range interest, from 30 to 100 radii, it is possible to secure a body temperature between about 150° and 1000° K. Temperatures of approximately 600° K may be obtained with simple body shapes which do not require orientation. Over this range the allowable temperature limit is determined more by components used for instrumentation than by structural factors.

Component Temperature Limits

The maximum operating temperature for typical components as of 1957 is shown in Table I. It is seen that these tend to fall into two groups. For one,

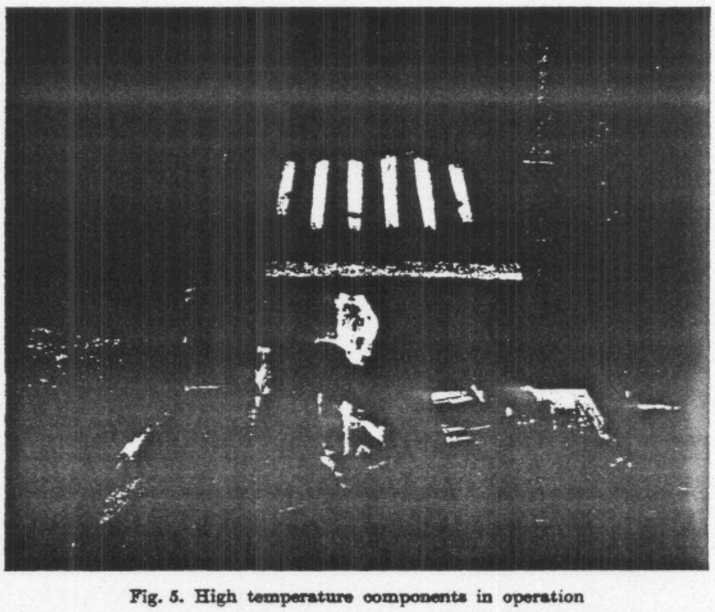

Fig. 5. High temperature components in operation

components are available for operation up to about 500° C (773° K), while for the other, the temperatures are limited to approximately the boiling point of water (373° K). An example of the high temperature technique is shown in Fig. 5.

Operation at the higher temperatures would eliminate the possibility of using semi-conductor devices. This would increase the power demand for instruments, since the overall electrical efficiency of vacuum tubes is relatively lower. It would also mean that semi-conductors could not be used to absorb solar power, and it would appear necessary to use the lower efficiency thermocouple system in this case.

Operation at the higher temperatures would also introduce a problem in the design of sensing instruments, since available radiation sensing equipment is limited to operation below about 100° C. This appears to be due to the fact that the need for high temperature sensors has not been great, since there are several sensing principles which appear to be basically useable to at least 500° C.

If the low temperature limit is chosen, the limit of approach with simple non-oriented bodies is approximately the orbit of Venus, which does not appear to be close enough for good experimental programs. Oriented bodies can approach as close as about 20 radii, which appears ample.

It appears that there is little reason for preferring one group of components over the other, since the lower temperature group offers the greatest design freedom in component selection, but requires stabilization, whereas the high temperature group introduces problems of power supply, efficiency and, therefore, weight, but eliminates the need for stabilization of the body. Pending further study, therefore, it is arbitrarily assumed that the low temperature components are used.

Table I. *Temperature Limits of Components*

Item	Maximum Temperature, 1957
Bearings	540° C
Resistors	700° C
Potentiometers, precision	200° C
Transformers	500° C
Capacitors	500° C
Tubes	600° C
Radiation Detector Tubes	100° C
Semi-conductor-silicon	200° C
Semi-conductor-germanium	75° C
Semi-conductor-selenium	125° C
Wire, insulated	500° C
Relays	125° C
Batteries	225° C
AC generators	250° C
AC servo motor	500° C
Torque motors	450° C
Selsyn	450° C
Circuits	500°—600° C
Thyratron Tubes	450° C

Stabilization

It appears that the stabilizing sensing system should be operated by the Sun's radiation. Sensing and control are required in pitch and yaw, but are not needed in roll.

One of the many possible sensors is sketched in Fig. 6. In this sensor a Sun shade is mounted above an assembly of four photocells, so that the outputs of all cells are equal at the correct orientation. For other orientations the difference in output levels forms an error signal.

It appears that it would be best to secure control forces by a combination of two systems. For initial conditions forces would be secured by valving compressed gas to bring the angular velocity of the body to zero, or very nearly so. To save weight the gas may be stored in the probe body. The long term but smaller control forces would be secured by small electrical motors, operating as a momentum conservation system.

The control sensor can be made very small, say one inch in diameter. However, it appears that a larger sensor would be better, since it appears possible to secure sufficient power to operate the control motors directly from the sensing cells. In this case the momentum motors could be constructed with two windings on a common shaft. Lead networks for stabilization would have to handle appreciable power, and so would use R-L circuits instead of the conventional R-C types.

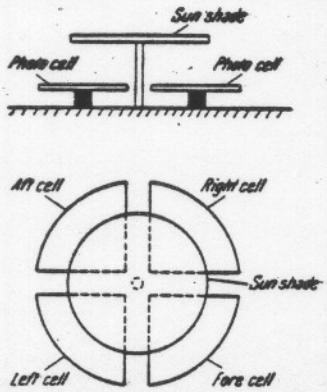

Fig. 6. Stabilization sensor

Telemetry [5]

Assuming that telemetry is only required until the probe is first eclipsed by the Sun, the maximum distance over which telemetry is required is equal to the distance of closest approach plus the radius of the Earth's orbit, a total distance of about 207 million km, or about 129 million miles.

Since the propagation path will pass close to the Sun, the telemetry frequency should be chosen in a region relatively free of solar noise; that is, in the centimeter wave bands. However, equipment is generally less efficient and reliable at the extreme frequencies. As a compromise, it is assumed that a frequency of 1000 mc/s is used.

The free space attenuation at the maximum range is about 260 db. This is very large, so it will be assumed that a narrow band CW system such as Microlock will be used. For 5 cps of band width the theoretical receiver sensitivity is —197 db below one watt. A 6 db noise figure, and 10 db signal/noise ratio appear reasonable. One hundred foot diameter ground receiving antennas are available, which give a gain of 47 db at the operating frequency. The transmitter power required is:

Theoretical Sensitivity	−197 dbw
Noise Figure	*6* db
Receiver Noise Level	−191 dbw
S/N Desired	*10* db
Input of Receiver Terminal	−181 dbw
Antenna Gain	− *47* db
Level at Antenna	−228 dbw
Path Attenuation	*260* db
Transmitter Power	+32 dbw

or, allowing one db for extra loss, a radiated power of two kilowatts.

The equipment required to generate this level of power does not need to be complex or heavy. However, the power supply would involve a considerable absorption problem. It appears that two approaches are open: one is to use a directive transmitting antenna, the other is to operate the telemetry intermittently to reduce the average demand.

A transmitting antenna of 4 feet diameter would provide a power gain of 20 db, which would reduce the transmitter power rating to 20 watts. However, this antenna would require orientation, since its beam width is about 16°. This would in turn require roll stabilization in addition to the pitch and yaw stabilization provided for thermal control and orientation sensing. This could be accomplished by "homing" on an Earth transmitted signal.

An alternate means is to restrict the time of transmission, to say one minute every two hours. This would reduce the average power demand to about 30 watts, at 50% efficiency. This would require a storage battery and a timing system.

Neither of these solutions are really satisfactory, but it appears that one or the other must be used since the remainder of the system has been carried to the practical limit. For the rest of this study it is assumed that intermittent transmission is employed.

With the narrow receiver pass band employed, data transmission would have to occur at a very slow rate. It appears that a suitable system is a modified pulse-width modulation, with the width of a pulse representing one quantity and the interval between pulses representing another. Pulse widths and spacing of 10 seconds duration would give an accuracy of approximately 2% full scale. A train of seven pulses would give a warning pulse and 12 active channels. This train would be repeated every two hours.

This type of telemetry system is certainly minimal, but is still capable of handling an appreciable amount of data. For example, during the period from launching to closest approach, about 25,000 individual readings, of 12 quantities would be transmitted.

Probe Experiments

It does not appear to be possible to make a direct determination of the density of the solar atmosphere. The experiments suggested here have been selected to provide a reasonable estimate of the atmospheric density and condition.

Since the major component of the atmosphere is hydrogen, a suitable means of determination of density is the measurement of the intensity of hydrogen Lyman alpha radiation as received from the Sun and from the atmosphere, as suggested by Chubb, Freedman and Kupperian [4]. For this experiment two photon counters would be used, one which its window looking toward the Sun, the other looking at right angles to the Sun. The second counter would receive scatter radiation from neutral hydrogen, and also recombination radiation from ionized hydrogen. The absolute magnitude of these would vary with the process cross-section, the recombination rates, and the solar intensity. The solar intensity will vary appreciably, particularly during flare activity. This variation, measured by the first counter, plus the ratio of counter readings which will vary with intensity and recombination rate provide means for separating the effects, and thereby determining the density of the atmosphere and its condition.

An independent measurement of ionized particle intensity can be secured by separate counters sensitive to electrons and protons, by a system developed by workers at the State University of Iowa [6]. For this a pair of Čerenkov detectors are used. In one a magnetic field is used to sweep electrons aside, so the counter is sensitive to protons and heavy particles. In the other shielding is used to block protons, and the electrons are focused on the detector by a magnetic field. The intensity of the field can be varied to secure a measure of the electron energy.

For these counters the total number of particles and the total intensity received during a transmission interval would be telemetered. Additional information on the ion content of the transmission path can be determined by measurements on the telemetry signal itself.

Of the many other possible experiments, only a meteor impact experiment has been included. This uses an impact microphone technique developed by DUBIN [4].

These experiments are suggested as typical and it is expected that specialists in rocket measurement would suggest more definitive experiments in some cases. and also suggest other important fields of investigation. Ref. [4] contains a compilation of such suggestions.

General Features of the Probe Vehicle

A check of Fig. 4 indicates that no special surface treatment is required for a highly conductive cone for the approach distance and maximum operating temperature selected. However, attainment of high conductivity would be troublesome, and the length of the cone would increase weight problems. A better solution would use surface treatment, which allows a short cone. For example, if the conductivity were large, a cone having sides equal to the base diameter would not exceed the maximum allowable temperature if the emissivity ratio e_i/e_r is 0.25 or greater.

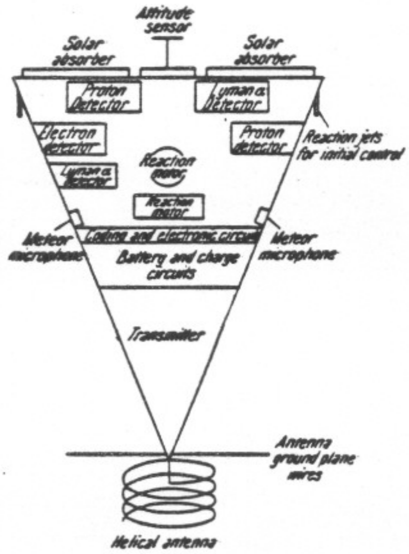

Fig. 7. Schematic diagram of the probe

A schematic representation of a suitable vehicle is shown in Fig. 7. Here the transmitter and storage batteries are amounted at the vertex of the cone for good cooling. The solar cell absorbers, and the attitude sensors are on the base of the cone, which always faces the Sun. Good cooling paths to the sides of the cone would have to be provided for these cells by metallic conduction.

The stabilization motors are mounted near the center of the vehicle. Stabilizing jets for initial use are mounted near the base. Measuring elements and cooling and servo amplifier elements are located in the remaining space.

A helical antenna with ground plane has been chosen for telemetry, since this eliminates most of the effects of roll on the signal and also makes for easy pattern shaping. The antenna is mounted at the cone vertex.

The cone would be approximately 2 feet across the base by 2 feet high. The shell would be built of a magnesium or lithium based alloy. Preliminary weight estimates indicate that the entire assembly can be fabricated for appreciably less than 50 lbs.

Summary

It appears that it is possible to construct a solar probe which is within the present state of the art. The distance of closest approach to the Sun, and the payload capacity are adequate to permit definitive experiments to be conducted. One group of such experiments are suggested.

The major present limitation appears to be in the propulsive vehicle, which limits the distance of approach for a reasonable payload to approximately the orbit of Mercury. Telemetry techniques are also limiting in small vehicles, but need not be in larger ones capables of carrying and orienting directive transmitting antennas.

Thermal problems do not appear to be limiting in any way: in fact most of the needed high temperature components are available for a design capable of approaching to a distance of 5 radii of the Sun, or to within about 4 million miles of the solar surface.

It is suggested that investigations of the type outlined here would form a sound basis for the extension of the Geophysical Year.

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