Thermal Design Considerations for the Solar Probe Electric Field Antenna

Edited by

Neil Murphy & Ralph McNutt

With contributions from

Stuart Bale
Bob Ergun
Bill Farrell
Justin Kasper
Bill Kurth
Ralph McNutt
Earl Scime
Ed Sittler
Charles Swenson
Bruce Tsurutani
O.C. St. Cyr

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1. Introduction

The Solar Probe mission will fly a spacecraft as close as 3 solar radii from the Sun's surface to determine the processes responsible for the heating of the solar corona and the acceleration of the solar wind. For spacecraft in formulation, there is a need to specify fully the instruments comprising the scientific payload and how they will interface with and influence the spacecraft itself. Such initial considerations may be subject to change as the design matures but form the basis for determining the cost and potential scope of the mission. The evolution of the interplay between spacecraft and payload design varies from benign to significant, depending upon the various constraints and requirements on both the spacecraft and the payload. As part of the payload, Solar Probe will carry an electric field and plasma wave experiment that will examine waves in both magnetic and electric fields over a wide range of frequencies. One of the requirements for the plasma wave experiment is that the electric field antennas are partially illuminated during the perihelion pass, which leads to several unique design issues. The purpose of this document is to provide background information on the environmental requirements for these antennas, a short updated and focused review of PWI science requirements, recommended implementation strategies for antenna design and the potential consequences of the environmental requirements and implementation strategies on instrument performance. This document should also be viewed as an addendum to the baseline design presented in the STDT report. The science rationale for obtaining these measurements can be found in the STDT report.

2. Mission Requirements

The combination of the recent work by the Science and Technology Definition Team (STDT) for Solar Probe (NASA 2005; McComas et al.) and the work reported above indicate the challenges for the implementation of appropriate antennas for the Plasma Wave Instrument (PWI).

The PWI is considered to be very important to meet Solar Probe's primary objectives, but previous studies had short antennas that were contained within the umbra. During the course of the STDT study, direct plasma coupling of the antennas was identified as an important measurement requirement needed to meet the science requirements. As a result, the determination of a feasible concept to directly couple to the plasma and not endanger the spacecraft and instrument, or adversely affect other instruments, received particular attention during the STDT effort.

Antenna issues are driven by the mission requirements, viz. (1) provide prime science data from 20 solar radii (R_S) to a perihelion of 4 R_S (measured from Sun center) on an orbit roughly perpendicular to the plane of the ecliptic, (2) make at least two perihelion passes, and (3) do not significantly modify the orbit between pass one and pass two. The first requirement is set by the combination of traveling "as close as possible" to the Sun and providing a full cut in latitude subject to technological constraints and prudent margins. This requirement, in turn, drives the thermal environment that will be encountered (which is well known). The second requirement was derived by the need to observe the Sun during at least two parts of the solar cycle. This requirement, in turn, means that any actions/features meant to deal with the thermal environment must be capable of being "redone" for the second pass, e.g. one cannot help mitigate the thermal environment with one-time deployments that would preclude measurements on the second pass. The third requirement is driven by launch mass and vehicle availability considerations. Significant changes would require corresponding significant on-board propellant. Associated constraints and trades were studied by the STDT in some detail, and the conclusion was that such a capability is neither prudent nor desirable. The consequence is that the spacecraft will return to deep space (Jupiter's orbital distance of ~5 Astronomical Units – AU) between passes. This reinforces the consequences of the second mission requirement that any action of say, a mechanism, to deal with the thermal environment on perihelion pass one cannot be undone for pass two. In other words, any type of actuators and their associated movements used to mitigate the thermal environment near the Sun must be viewed in the context of required reliability and risk for the entire mission (two perihelion passes, separated by a deep space excursion).

3. Solar Probe PWI Electric Antenna Requirements

The PWI electric field experiment should measure fluctuations in the *in situ* electric field from close to DC to above the plasma frequency at 4 R_S (quasi-DC to 10 MHz was chosen for the strawman instrument). Measurement of weak type III radio bursts to higher frequencies (~20 MHz) is also an important goal as it may indicate the existence of a population of faint type III bursts associated with microflares. To achieve the required low-frequency sensitivity, part of the antenna must be in sunlight to generate photo-electrons. Each sensor should sit at the same place on the current-voltage curve, therefore the illuminated portion (and hence photoelectron flux) of each antenna should be equal - in the strawman design this is achieved using an isolated 1-m segment at the end of each antenna. The science requirements given here will separate into a high-frequency regime and a low-frequency regime, since different design considerations must be considered for both regimes. Other issues for consideration are wake effects and spacecraft noise (electromagnetic interference or EMI) problems.

3.1 Relevant Solar Probe Plasma Wave Experiment Parametars

Estimates of ambient solar wind and resulting antenna parameters relevant to the Solar Probe Plasma Wave experiment can be found in Table 1. These data do not take account of the potential effects of the spacecraft wake on the plasma parameters

4. Science Implementation

4.1 Instrument Design Issues

The design issues presented below are intended to provide a general picture of the issues confronting the design of an electric field instrument for the Solar Probe mission and highlight any design considerations that might be impacted by design choices driven by thermal considerations.

An electric field and wave instrument consists of antennas, preamplifiers, current and surface bias, and signal processing. The main functions of the electric field instrument are (1) to measure the in-situ wave electric field to characterize plasma waves and spatially-coherent structures that can heat plasmas, (2) to measure *in-situ* plasma emissions to determine the properties of the ambient plasma, in particular, the plasma density, (3) to measure the floating potential of the spacecraft, which is both a proxy for electron density and is used to correct thermal electron measurements, and, (4) to sense remotely radio emissions and the direction of propagation to characterize the nearby solar corona.

Table 1. Estimated Values of Solar Probe Plasma Wave Parameters

	Quantity	4 R_S	12 R_S (over poles)	20 R_S
n_e	Electron Density	2×10^4 cm^{-3}	10^3 cm^{-3}	500 cm^{-3}
T_e	Electron Temperature	200 eV	100 eV	50 eV
$\mathbf{B}$	Magnetic Field	3×10^4 nT	3×10^3 nT	1000 nT
v_{SW}	Solar Wind Velocity	300 km/s	800 km/s	300 km/s
v_{SC}	Spacecraft Velocity	310 km/s	180 km/s	140 km/s
Length Scales				
λ_D	Debye Length	0.7 m	2.3 m	2.5 m
$\delta_e = c/\omega_{pe}$	Electron Skin Depth	35 m	160 m	240 m
ρ_i	Ion Gyroradius	50 m	330 m	700 m
$\delta_i = c/\omega_{pi}$	Ion Skin Depth	1500 m	7000 m	10000 m
Fundamental Frequencies				
f_{pe}	Electron Plasma Frequency	1400 kHz	300 kHz	200 kHz
f_{ce}	Electron Gyrofrequency	1000 kHz	100 kHz	30 kHz
f_{pi}	Ion Plasma Frequency	30 kHz	7 kHz	5 kHz
f_{ci}	Ion Gyrofrequency	500 Hz	50 Hz	16 Hz
$f_{Doppler}$ (δ_e)	Doppler Shift (Electron Scale)	12 kHz	6 kHz	2 kHz
$f_{Doppler}$ (δ_i)	Doppler Shift (Ion Scale)	300 Hz	150 Hz	50 Hz
Antenna Currents				
j_{photo}	Photoelectron Current	28000 nA/cm^2	1600 nA/cm^2	600 nA/cm^2
j_{th}	Electron Thermal Current	230 nA/cm^2	16 nA/cm^2	5 nA/cm^2
Antenna Properties 1 m x 0.5 cm diameter: In sunlight, 75° angle.				
j_{photo}	Photoelectron Current	350 μA	20 μA	8 μA
j_{th}	Electron Thermal Current	36 μA	2.5 μA	0.8 μA
C	Capacitance to Plasma.	20 pF	20 pF	20 pF
Cb	Base Capacitance [goal]	~ C	~ C	~ C
R	Resistance *Ibias* = 1/2 *jphoto*	10 kΩ	150 kΩ	400 kΩ
f_{RC}	R/C Crossover	800 kHz	50 kHz	20 kHz
R	Resistance *Ibias* = -*jth*	100 kΩ	1200 kΩ	4000 kΩ
f_{RC}	R/C Crossover	80 kHz	6 kHz	2 kHz
Signals				
v x **B**	Induced electric field.	12 V/m	2.5 V/m	0.5 V/m
$E_{max,\ DC}$	Max DC Signal	1 V/m	0.1 V/m	0.1 V/m
$E_{max,\ 100\ kHz}$	Max Signal at 100 kHz	10^{-3} V/m Hz$^{-1/2}$	10^{-4} V/m Hz$^{-1/2}$	10^{-4} V/m Hz$^{-1/2}$
V_n	Noise at 100 kHz	10^{-6} V Hz$^{-1/2}$	3×10^{-7} V Hz$^{-1/2}$	2×10^{-7} V Hz$^{-1/2}$

Past experience (e.g. ISEE-1 [*Mozer et al.*, 1978], Polar [*Harvey et al.*, 1996], FAST [*Ergun et al.*, 2001], Cluster (*Gustafsson et. al.*, 1997], Wind [*Bougeret et al.*, 1995], Ulysses, and STEREO [Bale et al., 2007]) on antenna design has given rise to well-known criteria for a successful electric field measurement:

- The antenna must be articulated normal to, and as far away as possible from, the spacecraft body. Generally, the antennas must extend several spacecraft radii (or Debye lengths) from the surface of the spacecraft. Noise due to the floating potential of the spacecraft falls off as $1/r^3$.
- The antenna must not be in the spacecraft wake.
- The capacitive and resistive coupling of the antenna to the plasma must be adequate to measure the expected signals over the required frequency range.

It is almost impossible, to satisfy any of the above criteria without extending the antennas out of the shield's shadow. The Debye lengths and spacecraft radius are both on the order of ~1 m, so the antennas should be ideally ~5 m, and least ~ 3 m from the spacecraft. For short antennas, it is best to extend the active part of the receiving element away from the spacecraft (rather than use the entire boom for the antenna) to improve the effective baseline and reduce the influence of the spacecraft potential on the antenna's signal.

4.2 Spacecraft Wake

The second criterion is particularly strong, since rocket and satellite experience shows that, while an antenna in ram flow can measure the electric signals accurately, an antenna in the wake will measure low frequency noise associated with the spacecraft/plasma interaction. Spacecraft wake noise can be difficult to differentiate from natural plasma waves and may depend on plasma parameters. The best strategy is to keep the antenna sensors away from any potential wake turbulence. To this end, the strawman Solar Probe designs use three antennas equally spaceed in azimuth and inclined 15° to the spacecraft-sun axis. The high speed of the spacecraft at perihelion forces any plasma wake to be highly inclined reltive to this axis and the antenna are oriented such that two of them are in the plasma ram direction. With this said, further study of potential wake effects should be undertaken prior to the detailed design of an antenna system.

4.3 Resistive and Capacitive Coupling

The solar probe antennas will have a resistively-coupled frequency range extending to a minimum of 80 kHz and possibly much higher at perihelion, well above the ambient ion gyro-frequency. Thus the antenna design needs to incorporate many of the features typically seen in "DC" electric field instruments. For example, electrostatic effects such as spacecraft generated fields and nearby surface potentials need to be controlled as well as current exchange between the spacecraft and the antenna.

4.4 Base Capacitive Coupling

It is important to maintain a base capacitance comparable to the antenna free-space capacitance to maintain sensitivity at high frequencies. For example, the free-space capacitance of a thin, 1-m antenna comprable to the isolated 1m segments in the strawman design is roughly 20 pF: with a base capacitance as low as 60pF (eg the STEREO PWI antena) the transmission could be as low as 25%. The antenna design should either (a) "bootstrap" the signal shield or (b) locate the preamplifiers near to the receiving element. Additionl stray capacitance should therfore be avoided in the antenna deployment structure.

4.5 Antenna Currents

The resistive coupling and the AC noise depend strongly on the currents between the antenna and the plasma. The primary currents are the (1) photoelectron current, (2) the electron thermal current, (3) thermal conduction currents driven by temperature gradients in the antenna and (4) applied bias currents. Thermionic currents on the spacecraft and antennas should be kept to a minimum. Photoelectron current exchange between the spacecraft and antennas should be avoided.

4.6 Exposure to Sunlight

Exposure to sunlight is very important for a low-frequency electric field measurements. If not exposed to sunlight, the coupling between antennas and plasma depends on the smaller and highly variable electron thermal current (which varies with plasma density and temperature).

The photoelectron current densities in the near-Sun environment are roughly 100 times the electron thermal currents (see Table 1). While currents between the antenna and plasma increase the coupling, the AC noise (shotnoise) is increased. For this reason, it is best to reduce the cross section of the antenna to sunlight but maintain as much length as possible. Bias currents need to be applied to the antennas to maintain a floating potential of a few volts. The photoelectron emission will add roughly a factor of 3 to the noise level in the capacitive-coupled regime if the antennas are tilted away from the Sun.

4.7 Effective Baseline (Separation) of the Antennas

Clearly, a long antenna baseline improves the signal to noise ratio; DC offsets due to spacecraft-generated fields fall away as $1/r^3$. The noise from the spacecraft photoelectron current and the antenna currents are estimated in Table 1. A baseline of 1 m will allow only the strongest electric fields to be measured, but with limited dynamic range (~60 dB) and accuracy. To improve the dynamic rage, the baseline and the antenna capacitance need to be increased. A baseline and length of 3-5 meters is desirable.

4.8 Primary Design Considerations

The measurement requirements lead to the following primary design considerations for the solar probe electric field antenna system:

- Length of antennas
- Diameter of antennas

- Position of antennas
- Sun angle of the antennas
- Bias Currents
- Base Capacitance

5. Solar Probe Electric Antenna Thermal Environment

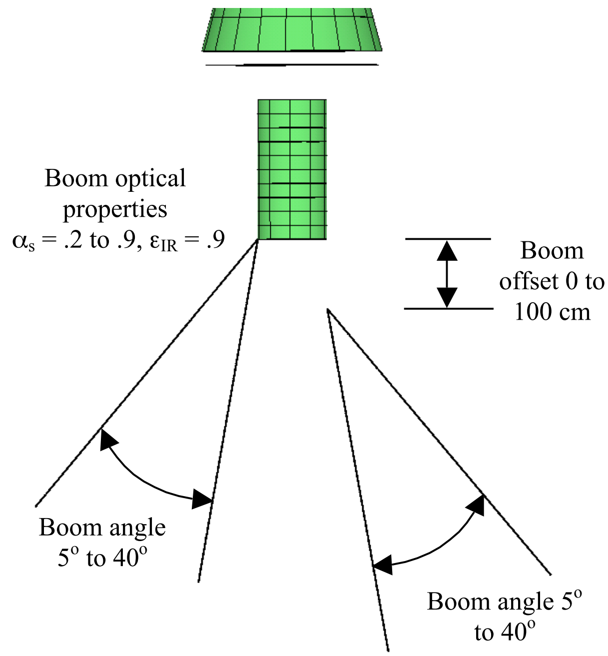


Figure 1. The geometry used for thermal modeling of the Solar Probe Electric field antenna

In the near-Sun environment, the Solar Probe electric antennas may reach temperatures as high as 1400°C. This places severe constraints on the materials used for antenna construction and also on the control of heat transfer to the spacecraft. As part of the Solar Probe STDT activities, engineers at the Applied Physics Laboratory modeled the thermal environment of the antennas to estimate the range of temperatures experienced by the antennas and their potential thermal impact on the spacecraft. Figure 1 shows the geometry of the model used: The antennas were modeled as 40-cm-long 2-cm-diameter tubes with varying thermo-optical properties. Three antennas were assumed, placed symmetrically at the base of the spacecraft, and the angle to the spacecraft axis (the sun-spacecraft line) was varied between 5° and 40°. Potentially offsetting the boom in the anti-sunward direction was also modeled, but this offset had little effect on the thermal impact of the antennas on the spacecraft.

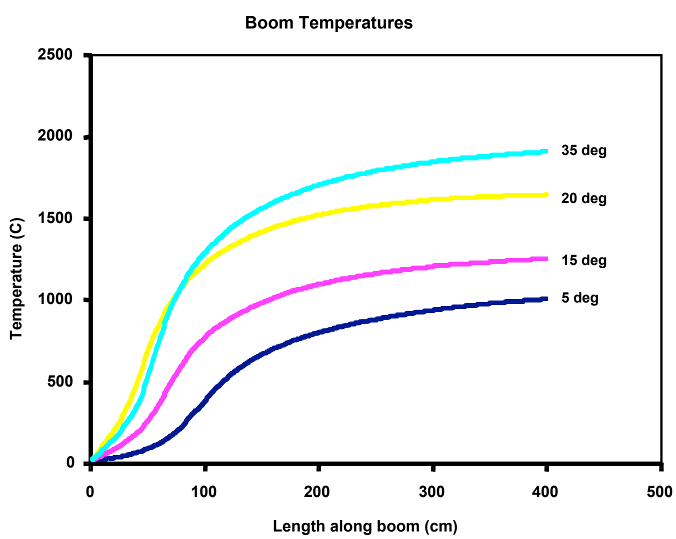


Figure 2. Temperature distribution along the antenna as a function of angle to the spacecraft axis at a perihelion distance of 4 R_S

The thermal modeling effort had two major goals: to determine the peak temperature reached by the antennas and to estimate the amount of heat deposited from the antennas into the spacecraft bus

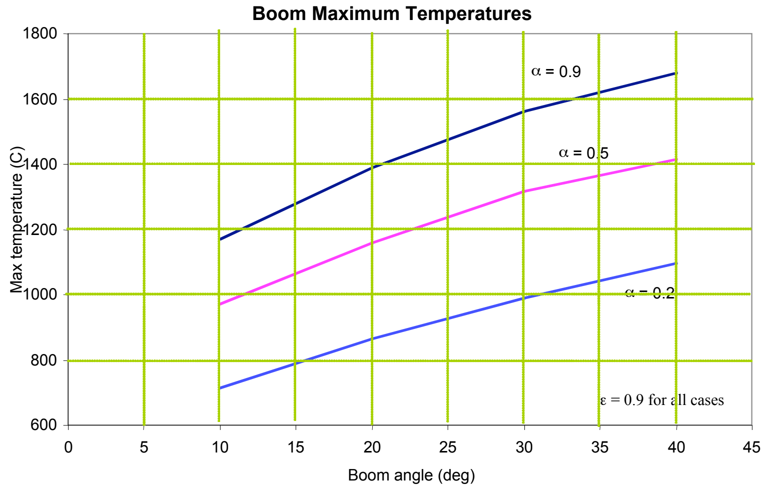


Figure 3. Maximum antenna temperature as a function of antenna angle for several values of absorptivity (α). The emissivity (ε) is assumed to be 0.9.

for various antenna geometries and materials. Figure 2 shows the temperature distribution along the antennas as a function of angle to the spacecraft axis, with the assumption that the ratio α/ε is 1. A critical result that is clear from this figure is that at the point at which the antenna attaches to the spacecraft, the temperature is not elevated, regardless of the peak temperature at the antenna tip. This occurs because the lower section of the antenna is in shadow and is radiatively cooled. This has two important consequences – (1) the antennas are not a source of conductive heat flow into the spacecraft and (2) the pre-amplifier, which is connected to the base of the antenna, does not have to work at high temperatures.

While one of the main performance drivers for an electric antenna system is to maximize the antenna tip separation and thus, for Solar Probe, the angle to the spacecraft axis, this may lead to unacceptable antenna peak temperatures. Figure 3 shows the results of an analysis to determine the peak antenna temperature as a function of antenna angle, as defined in Figure 1. The figure shows model results as a function antenna angle for three choices of absorptivity, α, that bracket the expected range for potential antenna materials (with ε kept constant at 0.9). For the strawman design an angle of 15° was chosen at perihelion as the optimum tradeoff between potential peak antenna temperature and desired tip separation. At larger distances the strawman concept allows the antennas to be articulated to increase tip separation with increasing distance from the sun.

Modeling the radiative heating of spacecraft from the electric antennas (not shown) at perihelion demonstrated that this source of

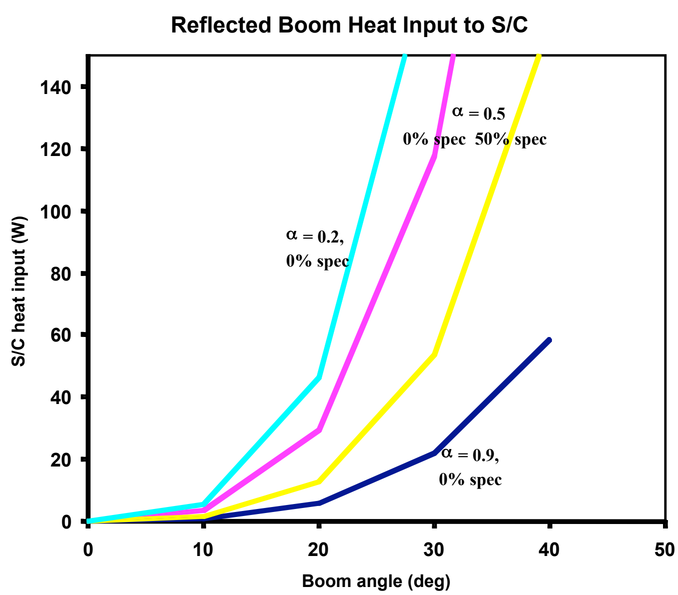


Figure 4. Estimates of heat input into the spacecraft as a function of antenna angle for different assumptions of absorptivity and scattering from the antenna.

heat input was negligible, leaving scattered sunlight as the main source of heat load. Figure 4 shows scattered light heat-load estimates as a function of boom angle at perihelion. Models were run with different assumptions about the thermo-optical properties of the antennas and the degree of specular reflection. As the antenna angle increases the heat scattered into the spacecraft increases rapidly, particularly for angles greater than 20°. This analysis further validates the choice of a 15° antenna angle at perihelion.

The analysis summarized above identified the major thermal drivers in the development of the strawman antenna concept, however, more detailed design efforts may also have to consider other issues. For example, depending on the construction of the sunlit section of the antenna, there may be a significant temperature gradient around the antenna circumference, leading to warping of the antenna – this could lead to unwanted mechanical stress and/or coupling into spacecraft dynamics.

6. Electric Antenna Implementation Approaches

Based on the discussion above, we can summarize the following requirements for the Solar Probe electric antennas:

(1) capable of surviving and operating in the near-Sun environment
(2) maximally separated at the tips
(3) coupled to the surrounding plasma by a photoelectron sheath
(4) illuminated symmetrically by the Sun
(5) longer than several Debye lengths in the measurements region
(6) more conductive (greater electron mobility) than the surrounding plasma
(7) have a photoelectron/thermionic emission current to not more that ~10 times the thermal current to limit the shot noise to the preamplifiers
(8) minimize the cross sectional area exposed to potential dust impacts
(9) minimize exposure to plasma wake produced by spacecraft

Requirements (1) and (3) are the primary drivers for material selection in that they imply illumination in the near-Sun environment, which will drive at least part of the assembly to high temperatures. Requirement (6) also drives the material selection in that conductivity must be assured for the range of temperatures that will be encountered from 20 R_S inward to 4 R_S and back out again on both the passes (the cold encounter near 5 AU following the first perihelion pass must not cause any irreversible conductivity changes or hysterisis effects that cannot be calibrated out). Requirement (5) is driven by the expectation that the Debye length will be a maximum of ~0.6 m (although this could be larger in a wake plasma); in this case "several" is ideally 5 to ~10 corresponding to lengths of 3 to 5 meters, although the experiment performance is expected to degrade gracefully with antenna length. Requirement (4) means that any spacecraft shadowing should be symmetric on any of the elements and the same for each. Requirements (7) and (8) imply achieving minimal diameter of the active element.

6.1 Accommodation Requirements

To survive the large solar flux close to perihelion, Solar Probe has a sophisticated thermal protection system. A conical primary shield, made from carbon-carbon re-radiates most of the incoming heat flux and a secondary shield reduces the heat flow into the spacecraft to acceptable levels. The spacecraft bus and bulk of the scientific instrumentation remain in shadow behind the thermal protection system and operate at close to room temperature during the solar periapse pass. To be consistent with the driving requirements above and allow packaging during launch, a strawman implementation with three antennas separated by 120° in azimuth, tilted back toward the spacecraft-sun line, with deployment/articulation and attachment systems within the spacecraft umbra and with a 1-m electrically isolated end segment, was developed.

With the above general implementation approach the overarching accommodation issue becomes one of minimizing the thermal input to the spacecraft to realistically manageable levels due to the active elements exposed to sunlight.

The engineering assessment, based upon the three antennas separated by 120° azimuth and all tilted from the axis of symmetry of the spacecraft thermal shield by equal angles, is

(1) Solar energy scattered from the antennas is the most significant source of heating to the S/C
(2) Actual S/C heating will be a function of multi-layer insulation (MLI), radiator and aperture location
(3) Spacecraft heating is a function of antenna angle and optical properties
(4) The allowable range is an estimate and assumes some local MLI shielding will be required:
    i. 100% diffuse reflection is worst case
    ii. Heat input reduces linearly as specularity increases
    iii. Reflected boom heating is not significantly affected by increasing the offset distance from the S/C
(5) Antennas may exhibit other thermal effects, such as azimuthal temperature gradients, resulting in additional design issues.
(6) Temperature gradient along antenna axis at shadow boundary needs to be investigated for possible antenna warping.

6.2 "Proof-of-Principle" Examples

As part of the engineering study to identify a feasible design space for the plasma wave subsystem antennas, consistent with thermal constraints established by Solar Probe study team, the engineering team looked at two concepts;

(1) A stacer type antenna and
(2) A rigid carbon-carbon antenna

with the carbon-carbon approach being chosen as the strawman during the STDT studies.

Carbon-carbon has the advantages of (1) good thermo-optical properties, (2) retains its physical characteristics at high temperature, and (3) is well characterized, but the disadvantage that it is rigid and so needs a non-standard deployment mechanism. Stacer antennas have the advantages of (1) high heritage, (2) are light, and (3) have low mass deployment systems. For use on Solar Probe, they must be fabricated from a refractory metal, however, which does not have heritage from previous missions.

For the study of both approaches, the following requirements were adopted:

- Antenna design requirements
  - A minimum of 1 meter of exposed antenna at 20 R_S
  - The 1-m section of the boom electrically isolated from the remainder of the boom (care must be taken not to unduly increase the base capacitance and thus reduce gain).
  - The distance between the centers of the active portions of the antennas > 2m.
  - 3 antennas required, 120° apart
- Engineering design requirements
  - Function up to >~1300K (which has implications for α and ε) during closest solar approach
  - Acceptable heat transfer to spacecraft

6.2.1 Stacer Approach.
For the stacer approach the following concept was developed:
(1) Thermal Constraints
1. Boom angle limited to ~15°
2. Boom optical properties:
   a. _ ~ 1.0
   b. $\_{IR}$ ~ 1.0
3. Boom operating temperature ~ 1600° C
   a. Remain conductive from boom root thru active length
   b. The design ensures that the boom will survive in an extreme environment that is very difficult to test
   c. Material property information is not readily available for temperatures in these ranges – some developmental work will be required
   d. The material temper will be affected by the high temperatures and may change its mechanical characteristics.

(2) Baseline Design
1. 3 Stacer booms at 15°

2. Booms ~4.5 m long
3. 1.1 m active length
4. 2.15 m separation of center of active length

The resulting concept is shown in Figure 5.

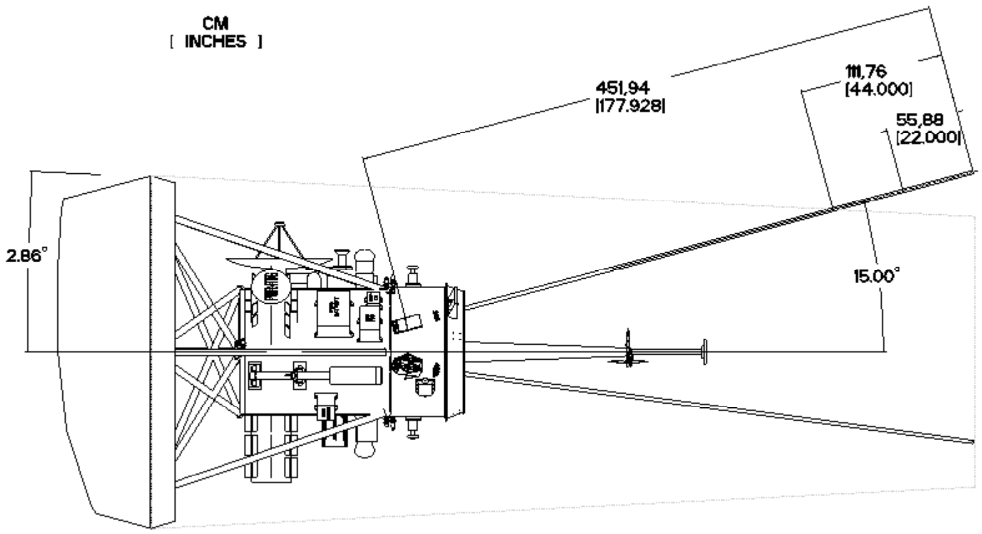


Figure 5. PWI antenna concept using stacer deployment approach.

The stacer deployment would occur once near 1 AU, shortly after launch and during initial instrument checkout. All risk issues identified with this approach concern the requirements for the stacer material maintaining required mechanical properties following deployment and maintaining optical properties commensurate with low thermal input to the spacecraft bus.

6.2.2 "Stickboom" Appoach.

To use rigid antennas, nominally carbon-carbon, the deployment must use a mechanism

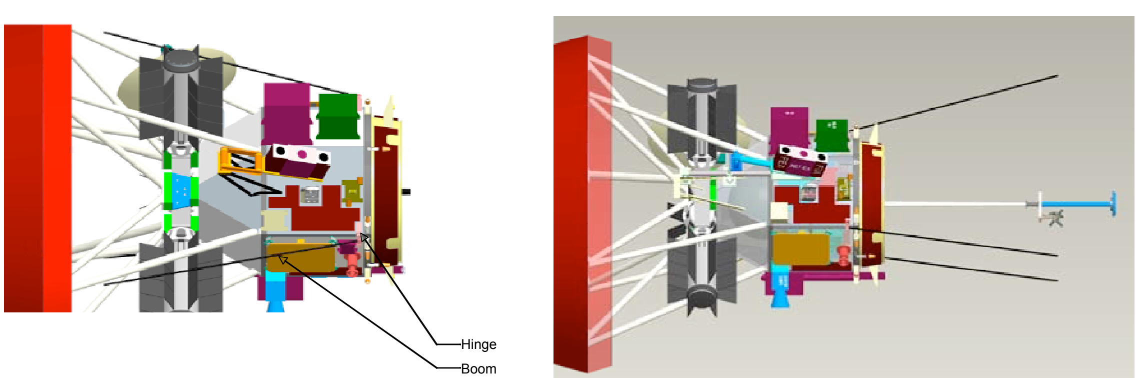


Figure 6. PWI antenna concept using one-time deployment stickboom antennas.

to “unfold” the antenna elements from the spacecraft. The spatial constraints, require, in turn that the overall length be limited unless there are multiple joints that would have to unfold and lock. The strawman design chosen was to avoid the mechanical issues with such a “carpenter’s rule approach” and not include intermediate hinges that would introduce both mechanical and electrical issues. The simplest approach uses a one-time deployment as illustrated in Figure 6.

An extension of the concept in Figure 6 one adds the capability to move the antennas so they can follow the umbra as it changes over the course of the spacecraft trajectory during the prime mission (within 20 R_S), allowing a 1-m illuminated segment during the entire perihelion pass, but without the need for excessively long antennas. The approach is illustrated in Figure 7 for two spacecraft-sun ranges. Care must be taken thast the mechanism used in this approach does not generate electrical interference and does not unduly add to the base capacitance of the antennas.

Advantages associated with the moveable concept include the fact that additional boom length is available, and for additional mass, risk, and cost (over the simple hinge), the booms can be driven to follow the umbra and keep 1 meter exposed. It was found that while support of the boom against the strut is workable, there are few options. Packaging of the spacecraft is very tight and is exacerbated by the addition of the stickbooms. Some

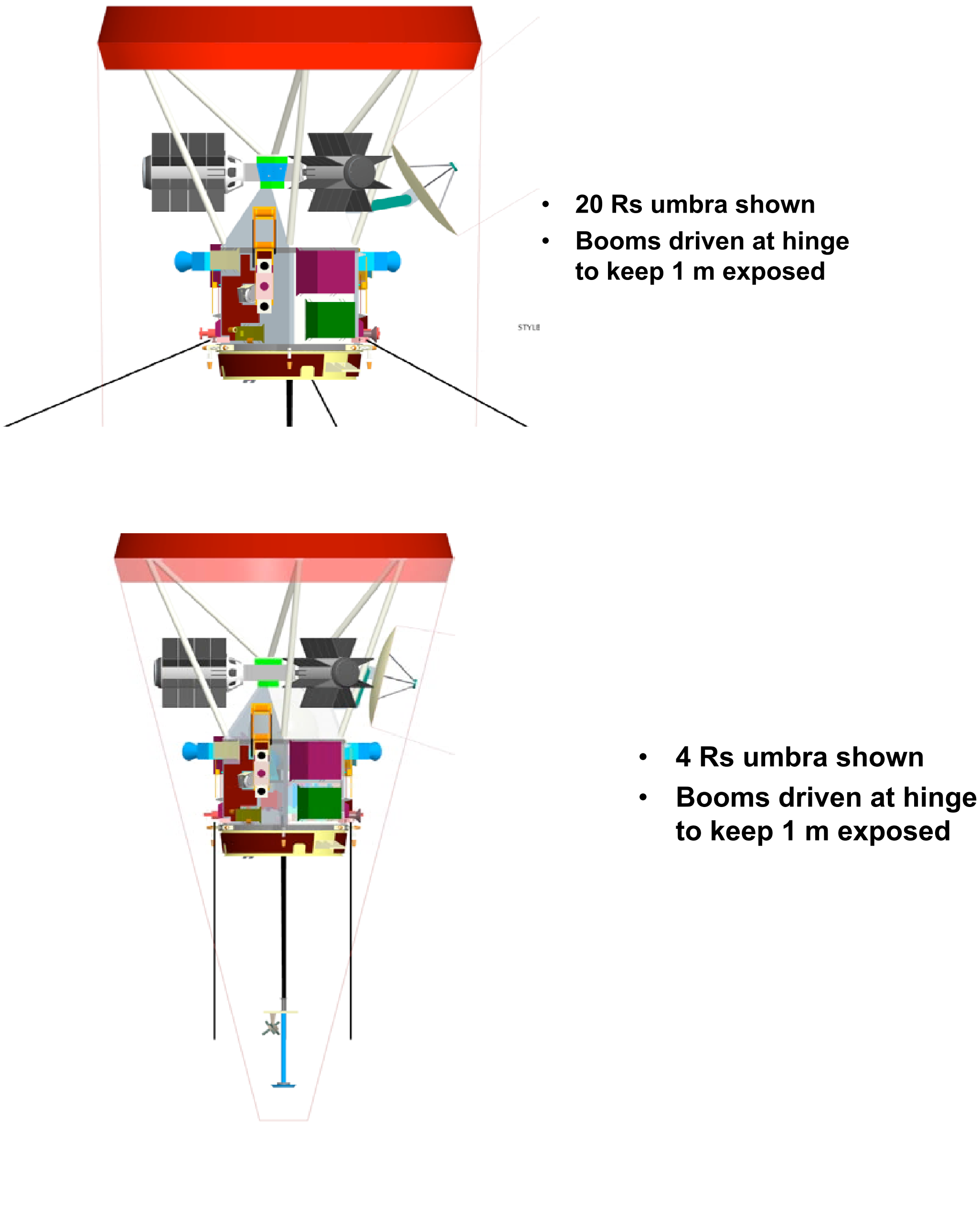


Figure 7. (Top) Stickboom deployment at 20 R_S. (Bottom) Stickboom deployment at 4 R_S.

reconsideration of some of the packaging of spacecraft subsystems would be required to thoroughly vet the design (this work was done later and funding precluded a fully self-consistent mechanical and packaging design by the engineering team).

6.2.3 Concept Trade Space.
The moveable approach allows more margin during the mission in that the antenna length and the angle outside umbra is actively controlled. This comes at the expense of requiring a reliable set of actuators that will rotate the antenna elements inward as perihelion is approached and then rotate them back outward as perihelion is passed, as well as reliably repeating this pattern at the second perihelion passage. There is thus a contingency to retract inside the umbra if spacecraft or instrument is getting too hot. If the actuators fail, an alternative fail-safe scheme may be required to automatically retract booms to zero degrees incidence or separate them from the spacecraft

For the stickboom approach the carbon-carbon antennas provide very good optical properties for mitigating potentially adverse thermal input into the spacecraft bus. An initial thermal analysis of potential stacer materials suggested higher thermal input based upon currently known optical properties. So with respect to overall accommodation of the antennas, while the stacer approach lends itself to more easily accommodating the measurement requirements and the carbon-carbon approach those of the spacecraft, studies to date have indicated that there exists a sufficiently large trade space, for all requirements to be accommodated, but with a need for further study of various materials and approaches in order to minimize mission risk.

In any case, these studies have shown that following PWI selection the instrument team and spacecraft team will have to work closely in implementing an optimized, fully consistent thermal-mechanical design for Solar Probe. Risk management for the selected antenna system must address issues of performance in the near-Sun environment (including charged particles and dust) and potential spacecraft impacts. In addition, appropriate management of risk must include appropriate high temperature testing.

7. Materials

7.1 Issues

The PWI electric antennas must operate over a broad temperature range, from ≈ 1400 °C at 4 R_S from the Sun to sub-zero temperatures beyond 1 AU. Implementation is affected by the possibility that different antenna material properties may have varying significance during different phases of the mission. For example, a specific modulus of elasticity or tensile strength may be more important at deployment, while melting point is clearly most critical at perihelion. In many cases, materials may undergo re-crystallization at temperatures well below their melting temperatures and must also be considered in the antenna design. Material properties that relate to antenna performance (work function or conductivity, for example) will need to be optimized for operation within 20 R_S, but even within this region material properties may change by an order of magnitude.

The development of high temperature materials for various engineering applications proceeded rapidly over the 20th century, driven by the need for strength and durability at increasingly higher temperatures. The development of gas turbines and jet engines led to the development of materials that can operate in the ≈500°C range, (various steels and titanium) to materials that need to withstand operating temperatures over 1000°C, such as the nickel superalloys (e.g. Nimonic-80 and Inconel). More recently intermetallic aluminides (e.g. Ni_3Al) have been developed that can operate at temperatures up to ≈1400°C as have ceramics (e.g. YSZ, SiC, Al_2O_3) and refractory metals and alloys that can be used up to several thousand degrees. Composite materials, such as carbon-carbon, have been developed that maintain their mechanical properties to well over 3000°C. Development of refractory materials continues at a rapid pace, for example Ir and Rh based intermetallics (e.g. Rh_3Ti) are being explored that will operate at over 2000°C.

A survey of available materials, based on the above development efforts, suggests that there are several candidates that might fulfill the requirements of an electric field antenna on the Solar Probe spacecraft, both from a thermal and mechanical perspective and an electric field experiment perspective. Different materials having differing properties could support a variety of potential antenna designs: for example, carbon-carbon for rigid, low reflectivity antennas, refractory metals for more flexible, perhaps thinner, antennas or a combination of several materials.

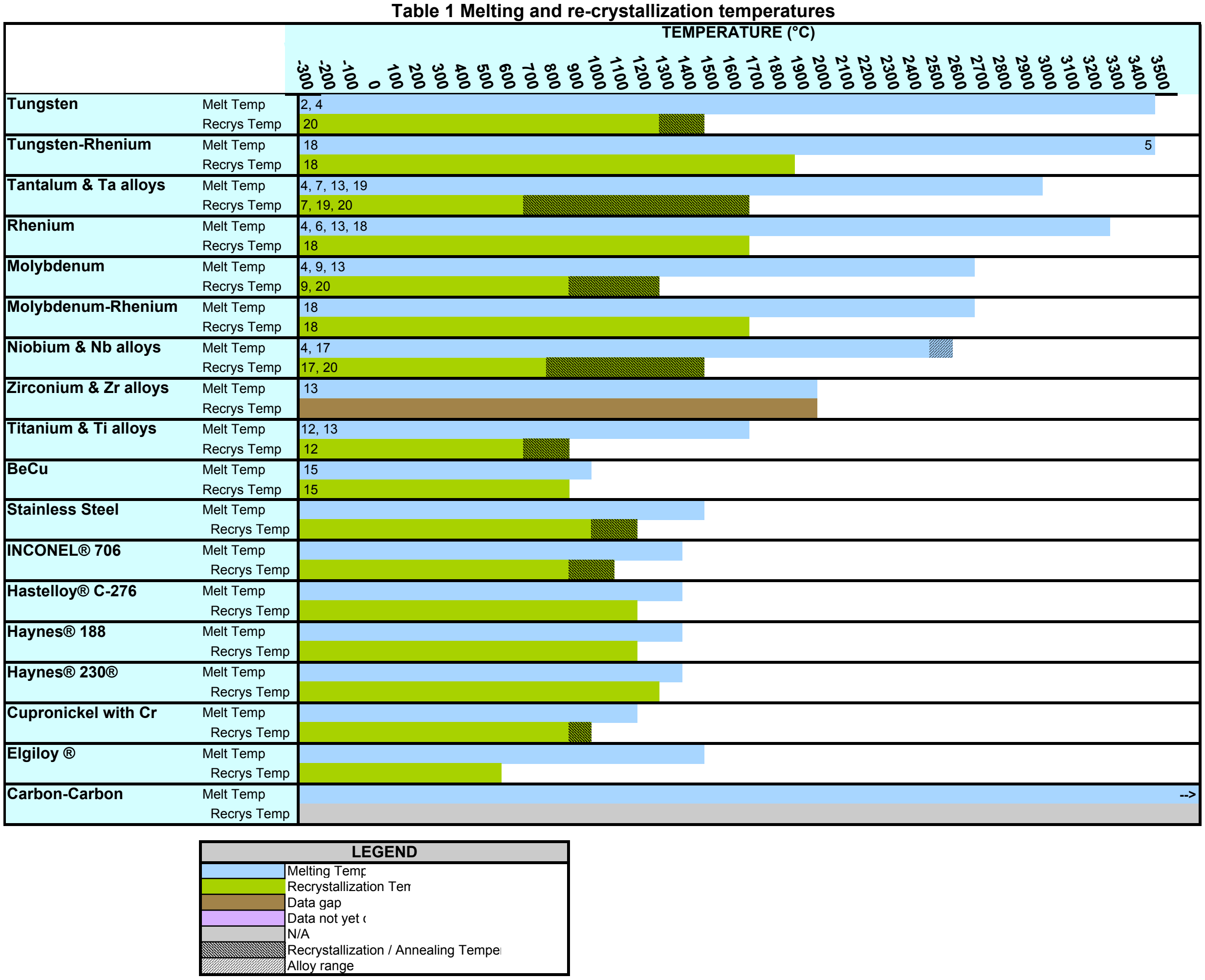
7.2 Data Summary

There are many potential materials that can be used to fabricate electric field antennas, but many of these will not survive the near-sun environment. It is important to identify a set of materials that will allow an effective plasma wave experiment to be designed, will survive the necessary environment and will not produce an excessive heat load on the spacecraft. The problem that we face is that, because of this non-typical set of requirements on the antenna materials, some of the properties we are interested in have not been tested over the whole range of temperatures that will be experienced by the Solar Probe antennas.

We have compiled data on range of potential materials in a format that highlights both the properties of interest and also the temperature ranges in which there is a paucity of data. Table 2 shows the melting and re-crystallization temperatures for a range materials, together with metals and alloys typically used in high temperature environments for comparison. Numbers in each bar refer to reference materials in the legend of Table 3.With the requirement that materials not only survive the Solar Probe environment, but also maintain their critical properties, it is clear that traditional high-temperature materials, for example Inconel or Hastelloy C-276, are inadequate for our purposes. Other materials, such as carbon-carbon, tantalum, tungsten, and alloys using these materials, are potential candidates for the electric field antenna

Table 2. Melting and Re-crystallization Temperatures for Some Materials

Table 1 Melting and re-crystallization temperatures



In Table 3, we have summarize the currently available information on a wider range of properties for the more promising materials from Table 2. These include tungsten and its alloys with rhenium and molybdenum, tantalum and its alloys including those with molybdenum and tungsten, rhenium, rhodium and alloys including with platinum, molybdenum and its alloys with tungsten, titanium, and rhenium, niobium and its alloys including those with tantalum, zirconium, and tungsten, zirconium and its alloys including that with tin, titanium and its alloys, nickel, carbon-carbon, and beryllium copper. Seven materials from table 2 are not listed, and in light, of the summarized requirements, the traditionally used BeCu is of questionable utility.

A significant question to resolve is, of these materials, (excluding carbon-carbon), which, if any, can be deployed from a stacer at 1 AU, and (§6.2.1) can maintain required mechanical properties following deployment and optical properties commensurate with low thermal input to the spacecraft bus.

TEMPERATURE (°C)

		-300	-200	-100	0	100	200	300	400	500	600	700	800	900	1000	1100	1200	1300	1400	1500	1600	1700	1800	1900	2000	2100	2200	2300	2400	2500	2600	2700	2800	2900	3000	3100	3200	3300	3400	3500
Tungsten	T Strength				4, 13	13	13	2, 13	2, 13	2, 4, 13	2, 13	2, 13	2, 13	2, 13	2, 4, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2, 13	2	2	2			13			
	T Yield							2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2	2										
	E Modulus	13	13	13	2, 4, 13	2	2	2	2	2, 4	2	2	2, 13	2	2, 4	2	2, 13	2	2	2	2	2	2						13											
	Hemi Emtnc					16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16				
	Spec Emtnc													16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16							
	Reflectivity																																							
	Absorptivity								21						21					21					21															
	Emissivity							21	21	21						21				21	21	21	21	21	21				21	21	21			21	21				21	
Tungsten-Rhenium	T Strength				3				3				3				3		1, 3			1		1	1															
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																				16		16		16		16		16		16		16		16					
	Spec Emtnc																16	16	16	16	16	16																		
	Absorptivity																																							
	Emissivity																				21									21					21					
Tungsten + Molybdenum	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																												16		16		16		16		16			
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Tantalum & Ta alloys	T Strength	7	7	7	4, 7, 13	7, 13	7, 13	7, 13	7, 13	4, 7, 13	13		13	13	4, 7, 13	7, 13	7, 13	7	7	7	7	7	7, 13	7	7	7, 13	7	7	7	7, 13	7	7	7, 13	7	7					
	T Yield				13	13	13	13	13	13	13		13	13	13	13	13						13			13				13			13							
	E Modulus		13	13	4, 13	13	13	13	13	4, 13	13	13	13	13	4, 13	13	13	13	13	13	13	13	13																	
	Hemi Emtnc									16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16							
	Spec Emtnc				16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16					
	Reflectivity																																							
	Absorptivity							21	21						21	21																								
	Emissivity		7	7	7	7	7	7	7, 21	7	7	7	7	7	7, 21	7,21	7	7		21		21			21					21				21						
Tantalum + Molybdenum	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																																							
	Spec Emtnc																	16				16			16		16		16		16		16							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Tantalum + Tungsten	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																																							
	Spec Emtnc																				16				16				16				16							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Rhenium	T Strength				4, 6, 13					4					4, 6, 13					13					6, 13															
	T Yield																																							
	E Modulus				4, 6, 13	13	13		13	13			13	13	4, 6										6															
	Hemi Emtnc				16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16									
	Spec Emtnc								16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16					
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Rhodium + Rh_3X	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Th Conduc							23		23	23		23	23	23	23																								
	Hemi Emtnc															16	16		16	16		16		16		16														
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Platinum + Rhodium	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc												16	16	16	16	16	16	16	16	16	16																		
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Molybdenum	T Strength			9	4, 9,13	9	9, 13	9	9, 13	4, 9	9, 13	9	9, 13	9	4, 9, 13	9	9, 13	9	9, 13																					
	T Yield			9	9	9	9							13	13	13																								
	E Modulus	13	13	13	4, 9, 13	9, 13	9, 13	9, 13	9, 13	4, 9, 13	9, 13	9, 13	9, 13	9, 13	4, 9, 13	9, 13	9, 13	9							13															
	Hemi Emtnc					16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16									
	Spec Emtnc				16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16									
	Reflectivity																																							
	Absorptivity							21		21					21																									
	Emissivity								21						21					21	21	21			21					21										
Molybdenum-Rhenium	T Strength				10								10				10				8																			
	T Yield				10								10				10																							
	E Modulus		14		8												8				8				8															
	Absorptivity																																							
	Emissivity																																							
Molybdenum + Titanium	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc									16	16		16		16	16	16	16		16	16	16	16		16		16	16	16	16		16	16							
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							

Molybdenum + Tungsten	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc													16	16	16	16	16	16																					
	Spec Emtnc																	16				16			16		16		16		16									
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Niobium & Nb alloys	T Strength				4, 11	11	11, 1	11, 1	11, 1	4, 11, 13	11, 13	11, 1	11, 1	11	4, 11, 13	11, 1	11	11	11	11																				
	T Yield				11	11	11, 1	11, 1	11, 1	11, 13	11, 13	11, 1	11, 1	11	11, 13	11, 1	11	11	11	11																				
	E Modulus				4, 11, 13	13	13	13	13	4, 13	13	13	13	13	13	13	11, 1	13	13	13	13	13	13	13	13	13	13													
	Hemi Emtnc									16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16													
	Spec Emtnc												16	16	16	16	16	16	16	16	16	16	16	16	16	16	16													
	Reflectivity																																							
	Absorptivity																																							
	Emissivity													21	21					21	21				21	21	21													
Niobium + Tantalum	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																16	16	16	16	16	16	16	16	16	16			16											
	Spec Emtnc																16			16				16			16			16										
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Niobium + Tungsten	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																							16		16		16		16										
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Niobium + Zirconium	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc								16	16	16	16	16	16	16	16	16	16	16	16	16	16																		
	Spec Emtnc																																							
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Zirconium & Zr alloys	T Strength				13	13	13	13	13	13	13	13	13	13	13																									
	T Yield				13	13	13	13	13	13	13																													
	E Modulus		13	13	13	13	13	13	13	13	13	13	13	13	13																									
	Hemi Emtnc												16	16	16	16	16	16	16	16	16																			
	Spec Emtnc			16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16																	
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Zirconium + Tin	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Hemi Emtnc																																							
	Spec Emtnc																16	16	16	16	16	16	16	16	16															
	Reflectivity																																							
	Absorptivity																																							
	Emissivity																																							
Titanium & Ti alloys	T Strength	13	13	13	12	12	12, 1	12	12, 1	12	12, 13		13		13																									
	T Yield	13	13	13	13	12	12, 1	12	12, 1	12	12, 13		13																											
	E Modulus		13	13	13	13	13	13	13	13	13																													
	Hemi Emtnc		16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16	16																					
	Spec Emtnc												16	16	16	16	16	16	16	16																				
	Reflectivity																																							
	Absorptivity																																							
	Emissivity													21								21																		
Nickel	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Absorptivity							21		21					21	21																								
	Emissivity							21		21					21	21																								
Carbon-Carbon	T Strength																																							
	T Yield																																							
	E Modulus																																							
	Th Conduc				22		22	22	22	22	22	22	22	22	22		22		22			22	22			22														
	Sp Heat				22					22					22					22					22					22					22					22
	Reflectivity																																							
	Absorptivity														21				21	21					21															
	Emissivity							22		22		22	22		21, 22		22		21, 2	21,22	22		22		21, 22		22		22	22			22		22					
BeCu	T Strength				15	15	15	15	15	15																														
	T Yield											15	15	15																										
	E Modulus																																							
	Absorptivity																																							
	Emissivity																																							

Table 3 (concluded). Legend

LEGEND	
(orange)	Tensile Strength (ultimate)
(teal)	Tensile Yield Strengh
(light blue)	Elastic Modulus (tensile)
(pink)	Thermal Conductivity
(light yellow)	Specific Heat
(tan)	Hemispherical Total Emittance
(blue)	Normal Spectral Emittance
(grey)	Reflectivity
(light green)	Absorptivity (alpha)
(light cyan)	Emissivity (epsilon)
(white)	Data gap
(lavender)	Data not yet compiled
(hatched)	We have data from which this parameter might be calculated

1	http://products.asminternational.org/hbk/do/highlight/content/V02 "Refractory Metals & Alloys"
2	http://products.asminternational.org/hbk/do/highlight/content/V02 "Properties of Pure Metals, Tungsten"
3	Alloy Digest, September, 1965 "Chase 75W-25 Re"
4	http://www.ultramet.com/refr.htm "Refractory Metals"
5	http://www.rhenium.com/tung_rhen.asp?menu_id=2&pic_id=9 "Tungsten-Rhenium"
6	http://www.ultramet.com/reprop.htm "Thermophysical Properties of Rhenium"
7	http://products.asminternational.org/hbk/do/highlight/content/V02 "Properties of Pure Metals, Tantalum"
8	"Stress-Rupture Strength and Creep Behaviour on Molybdenum-Rhenium Alloys," Freund et al
9	http://products.asminternational.org/hbk/do/highlight/content/V02 "Properties of Pure Metals, Molybdenum"
10	"Investigation of Mechanical Properties and Microstructure of Various Molybdenum-Rhenium Alloys", Leonhardt et al
11	"Nb / Nb Alloy Products; C-103, Overview" by Wah Chang
12	Material Properties Handbook: Titanium Alloys
13	Pure Metals Properties, A Scientific-Technical Handbook, A. Buch
14	"Low Temperature Mechanical Behavior of a Molybdenum-Rhenium Alloy", Agnew & Leonhardt
15	http://products.asminternational.org/hbk/do/highlight/content/V02 "Copper and Other Beryllium-Containing Alloys"
16	Touloukian Data Book (get citation)
17	http://www.rembar.com/niobium.htm
18	"Rhenium and Molybdenum / Tungsten Based Alloys: an Overview of Database", Bryskin & Carlen
19	http://www.sciencedirect.com/science?_ob=ArticleURL&_udi=B6TXD-44DHYK2-W&_user=120530&_coverDate=11%2F30%2F2001&_rdoc=1&_fmt=&_orig=search&_sort=d&view=c&_acct=C00000969
20	http://products.asminternational.org/hbk/do/highlight/content/V04 "Heat Treating of Refractory Metals & Alloys"
21	Microsoft Excel File: Hi-Temp Props1.xls, Robert Miyake
22	"Preliminary Design of the Thermal Protection System for the Solar Powered Solar Probe Spacecraft," S. Yerushalmi, J. Peng, and R.B Dirling, Jr., 2002 January.
23	Thermophysical Properties of Rh_3X for Ultra-High Temperature Applications Yoshihiro Terada, et al.*Platinum Metals Review, 69-76*. 2006.

8. Conclusions

A plasma wave instrument, consisting of appropriate preamplifiers and other electronics and electric field antennas, is important for a solar probe mission. Such an instrument will make measurements of the electric field components of waves in the plasma that will be diagnostic of (1) potential solar wind acceleration mechanisms, e.g. ion cyclotron waves, (2) the ambient environment in which those mechanisms operate, e.g. electron plasma oscillations, and (3) remote conditions relevant to coronal heating and solar wind acceleration, e.g. type III radio bursts. At the same time, as we have shown, there are significant implementation constraints, which are driven by the necessary antenna geometry, mechanisms, and materials. All three of these are interrelated as well as being driven by additional constraints on form factors, packaging, and thermal loading driven by the spacecraft and the mission itself.

The requirement that the antennas be in physical contact with the *in situ* plasma outside of the shadow of the primary thermal shield during the most physically stressing part of the mission imposes challenging considerations for risk mitigation philosophy and testing protocols. These same issues must be dealt with for the primary thermal shield, except for the issue of scattered light to the spacecraft bus. Hence, implementation options must be considered early so that testing can be appropriately costed and planned. The lack of a robust analysis my lead to increased risk and cost during the detailed design phase of the mission.

Current considerations (in this report) indicate that – while not necessarily optimal – the required measurements can be made with antennas undergoing a one-time deployment from non-articulated interfaces with the spacecraft bus. This is true of both the stacer and "stick-broom" approaches, the former longer, but requiring prepackaged elastic elements, and the latter inherently shorter due to the use of rigid elements. Both approaches have packaging, clearance, and launch-environment issues that require further analysis

Given its high heritage and capability for longer antennas, the stacer approach offers distinct advantages as well as the potential for cleaner measurements, i.e. with more margin. However, while there are materials with potential for this application, appropriate boom material(s) remain(s) to be appropriately tested and qualified for stacer deployment and high-temperature survival.

While the strawman choice for stick boom approach, carbon-carbon antennas need to tested and qualified for the types of launch loads they will experience prior to deployment. If structural issues are identified, then some of the high-temperature materials identified for potential stacer use could potentially be substituted.

We concur with the STDT report that an effective low mass, low risk approach is the “stick boom” approach with carbon-carbon elements, although the use of other materials may allow antennas with better performance (with smaller cross sections, for example). More analysis is required to verify this, and we do know that the measurement margins are less than optimal.

9. Recommendations

Two potentially viable approaches to plasma wave antenna implementation on Solar Probe have been identified, with a range of possible material choices, however, neither approach has been developed to a TRL 6 level. As part of the ongoing Solar probe risk mitigation activities, we suggest two tasks:

(1) Demonstrate by test of a full-scale protoype that a latched carbon-carbon antenna deployment mechanism can be flight qualified for appropriate loads encountered during launch.
(2) Demonstrate by deployment of a full-scale prototype, that a stacer boom made from an appropriate high-temperature material can be packaged and deployed in an implementable package.
(3) Complete a more detailed study of potential spacecraft wake effects