To Fly to the Sun: The Mission and Technology Challenges of the Solar Probe

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Abstract— To fly close to the sun (to a perihelion of 4 solar radii) represents many unique challenges to a mission and spacecraft design. The Solar Probe design is a result of over two decades of studies that have allowed the evolution of both the mission and trajectory design, as well as the spacecraft configurations. During these studies some of the most significant design challenges have been the trajectory design, the spacecraft shield design, the spacecraft configuration, the telecommunications near perihelion, science instrument accommodations, and minimizing the cost of the mission. This latter challenge (minimum cost) permeates all of the other design issues suggesting specific solutions consistent with this constraint. This paper will present the evolution and rationale that has taken place to arrive at the current design for this challenging mission.

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1. INTRODUCTION

The Solar Probe mission, that is under development and is scheduled to launch in early 2007, has been studied for over two decades as discussed in [1], [2], [3], [4], [5], and [6] in an effort to identify the technology challenges, as well as the mission and system design requirements. The gradual evolution of the elements of this mission has allowed optimizations of these elements over many years of tentative design concepts. This evolution has suggested design guidelines that can now be applied with a reasonable confidence leading to a more refined design. The technology challenges are driven by the current mission design concept that requires a high energy trajectory. Cost constraints for the mission demand the smallest launch vehicle possible while satisfying this high energy trajectory. This, in turn, requires that the spacecraft mass must be minimized. A key spacecraft element that allows a significant reduction in mass is the thermal shield/antenna (TSA) in the current concept discussed in [7] and [8] . The development of the TSA has been a major technology challenge but much progress has been made as given in [9] and will be summarized here. Another issue for this mission is the requirement of high rate telemetry at perihelion in real time while minimizing spacecraft power. Again, the TSA can provide a good solution to this issue by allowing a high rate downlink with radio frequency (RF) amplifier that consumes only a small amount of power. Because the antenna also acts as the spacecraft shield, it will be operating at very high temperatures (~ 2,000 K) at perihelion. The RF feed for the antenna must also operate at nearly that temperature. Significant progress in the development of this high temperature feed [10] will be discussed here. Related to the telemetry performance requirement at perihelion is the environment that the signal must pass through on the way to the earth. The spacecraft will be within the sun's "atmosphere" or corona at perihelion and the affects of the corona on the downlink performance must be considered when predicting the telemetry performance. Three experiments have been undertaken (see [11] and [12]) to record these effects on the downlink signals during earth-spacecraft solar conjunctions of three deep space missions. The results have enabled a new design philosophy for the Solar Probe downlink near perihelion that will significantly increase telemetry return.

Two science packages are under consideration for the Solar Probe missions listed in the Announcement of Opportunity (AO) [13]. The first package includes the in-situ instruments that will directly measure the environment around the spacecraft as it flies through the solar corona. The accommodation of these in-situ instruments within the TSA's shadow or umbra has been one of the key spacecraft design challenges. The other package consists of remote sensing instruments that will observe the sun's "surface" and corona using imagers with specific spectral bands that will produce unique images to determine the global context of the in-situ observations. Other unique requirements such as pointing control and stability near perihelion will also be discussed here.

2. MISSION OBJECTIVES AND DESIGN

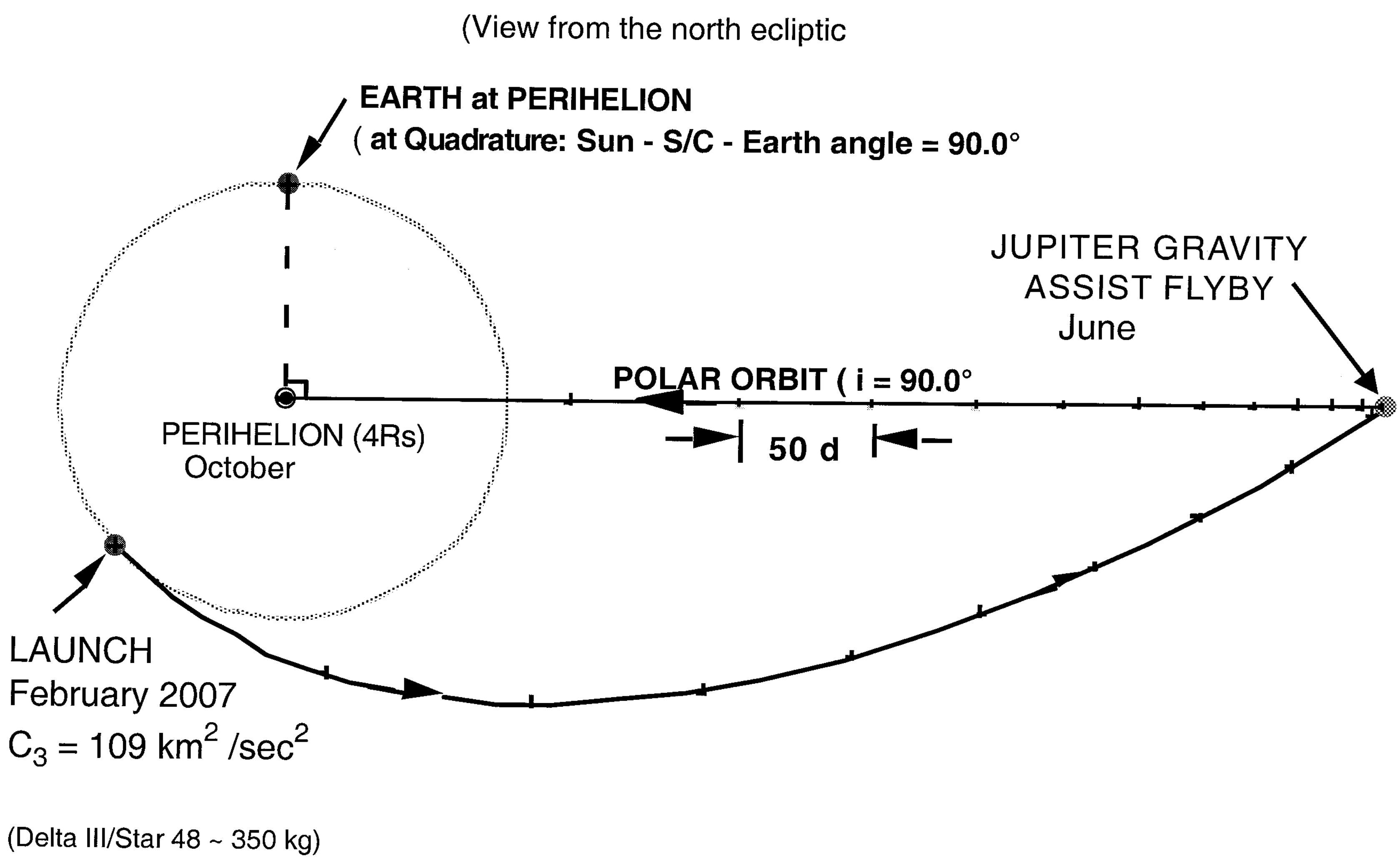
The fundamental objective of the Solar Probe mission is to fly through the corona with a highly instrumented payload at a distance that is as close to the sun as is possible with modern delivery capabilities and with the latest spacecraft technology capabilities. The scientific objectives are discussed in [13] in detail.

The navigation techniques developed by previous Jupiter gravity-assist missions such as Mariner Venus-Mercury 1973 and Mariner Jupiter-Saturn 1977 (a.k.a. Voyager) have been exploited by the Solar Probe mission designers. It was clear in early planning [2] that to deliver a reasonable payload close to the sun with a small (inexpensive) launch vehicle, a Jupiter gravity-assist (JGA) maneuver would be necessary. Figure 1 is a plot of a typical JGA trajectory to the sun for the most likely launch and arrival dates of the current baseline mission.

Three characteristics of JGA Solar Probe trajectories are illustrated here. First, the launch energy is quite high ($C_3 \sim 120$ km^2/s^2) for the direct trajectory to Jupiter. The arrival velocity at Jupiter must be quite high (~ 13 km/s) to properly determine the trajectory to the sun. Secondly, the JGA maneuver accomplishes three goals. It rotates the ecliptic inclination (i) of the trajectory from $i \sim 0°$ to $i = 90.00$ degrees. The precision of the final inclination is important as discussed below. The second goal is to reduce the perihelion radius to four solar radii (4 R_s) allowing the spacecraft to fly through the region of the solar corona where the solar wind is born [1]. Thirdly, the timing of the JGA trajectory can be designed such that the angle between the plane of the final orbit and the direction to the earth at perihelion can be 90.00 degrees or at "quadrature" with the earth which is necessary for the TSA design. One of the key issues of the initial design to JGA trajectories was the navigation accuracy of Jupiter. The aiming point at Jupiter had to provide the necessary final trajectory at the sun. The results of a navigation analysis [7] to determine the allowable errors in this aiming point is shown in Figure 2.

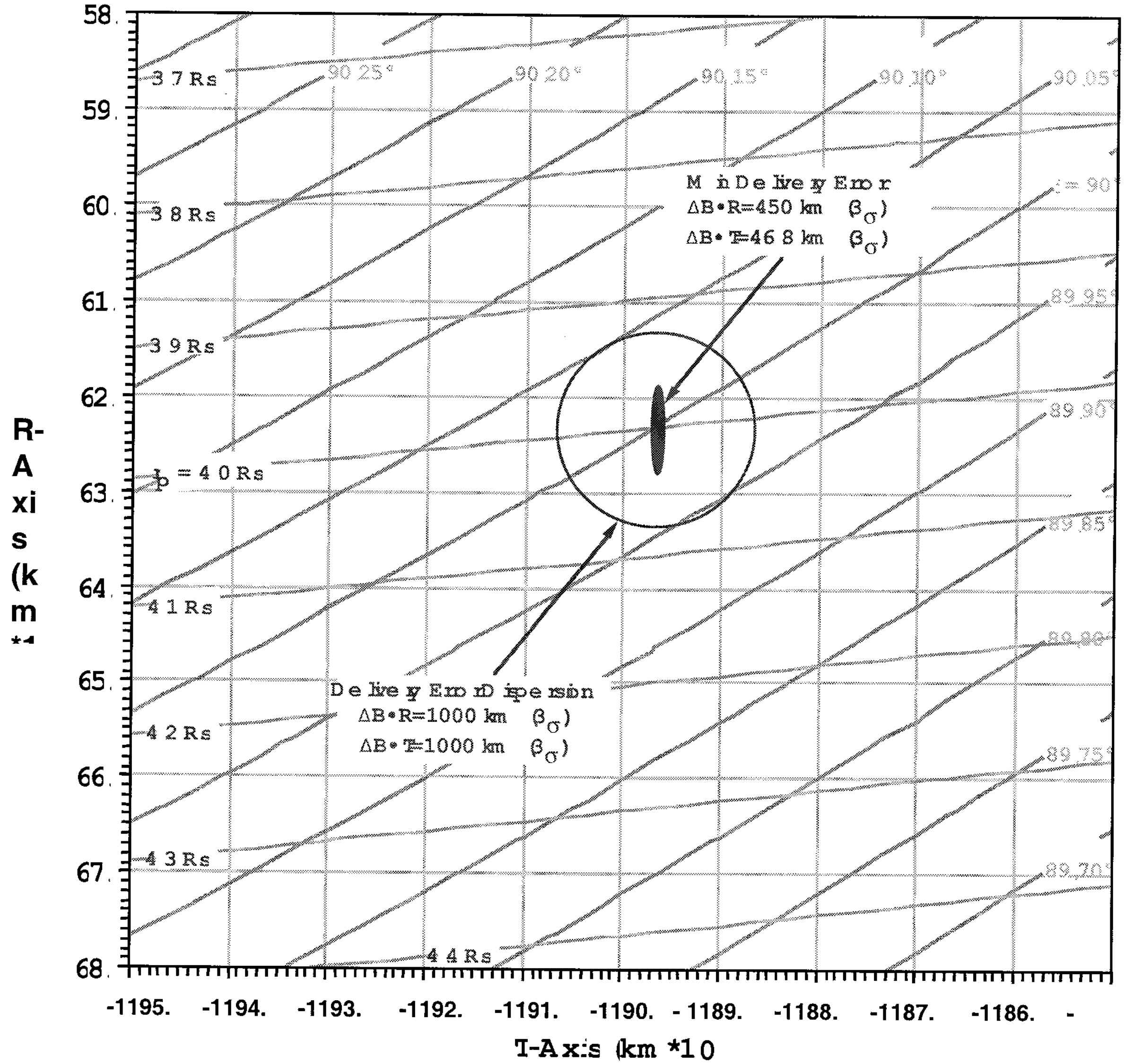
This is a plot of the so-called "B" plane or aiming plane at Jupiter. The B vector is measured from the center of Jupiter's projection into this B plane and represents the aiming distance in kilometers from center of Jupiter to produce a JGA trajectory to the sun. (This figure should be considered typical of a B plane analysis and could change slightly for different launch opportunities.) The R and T axes display the Jupiter centered aiming coordinates in kilometers. Of significance are the parametric curves plotted on this B plane. The near horizontal set of curves maps the Jupiter aiming point into perihelion radius (r_p) and the 4.0 R_s line shows the goal for the Solar Probe mission. The upward diagonal lines map the final orbit inclination as a function of the Jupiter aiming point and the i = 90 degrees line shows the goal for the Solar Probe mission. Clearly, the exact intersection of the 4 R_s and i = 90 lines is where the aiming point should lie. There will, however, be errors in the orbit determination as the spacecraft approaches Jupiter and these will map in the B plane as shown by the small black (error dispersion) ellipse at the desired aiming point. The error along the R axis (Δ B • R) is estimated at about 450 kilometers as shown as the major axis of the ellipse. Similarly the T axis error is the ellipse's minor axis. As listed in the box on the figure, the errors mapped into perihelion are quite small and well within spacecraft design margins at perihelion.

This navigation analysis demonstrated that the "quadrature" geometry at perihelion was readily achievable and that had a significant effect on the spacecraft configuration design. In addition, the i = 90 degrees trajectory takes the spacecraft over the solar poles which is a key scientific objective allowing direct vertical observations over the "coronal" holes expected to be the source of the high speed solar wind. Also, the 4R_s perihelion takes the spacecraft deep into the region of the birth of the low speed solar wind near the solar equator. These regions are shown in the near perihelion trajectory plot in Figure 3. Here the trajectory is illustrated from ± 30 days around perihelion. The view is from the earth and illustrates the effect of the quadrature geometry on the spacecraft design with the TSA. Because the TSA is an off axis parabolic antenna, its boresight is chosen such that the spacecraft nadir pointing maneuver rotates around the TSA boresight. This means that real time telemetry to



(View from the north ecliptic
EARTH at PERIHELION
(at Quadrature: Sun - S/C - Earth angle = 90.0°
JUPITER GRAVITY
ASSIST FLYBY
June
POLAR ORBIT (i = 90.0°
PERIHELION (4Rs)
October
50 d
LAUNCH
February 2007
$C_3 = 109\ km^2 /sec^2$
(Delta III/Star 48 ~ 350 kg)

Figure 1



58.
59.
60.
61.
62.
63.
64.
65.
66.
67.
68.
R-Axis (km
37 Rs
38 Rs
39 Rs
= 40 Rs
41 Rs
42 Rs
43 Rs
44 Rs
90.25°
90.20°
90.15°
90.10°
90.05°
i = 90°
89.95°
89.90°
89.85°
89.80°
89.75°
89.70°
Min Delivery Error
ΔB•R=450 km (3σ)
ΔB•T=46.8 km (3σ)
Delivery Error Dispersion
ΔB•R=1000 km (3σ)
ΔB•T=1000 km (3σ)
-1195.
-1194.
-1193.
-1192.
-1191.
-1190.
-1189.
-1188.
-1187.
-1186.
T-Axis (km *10

Figure 2

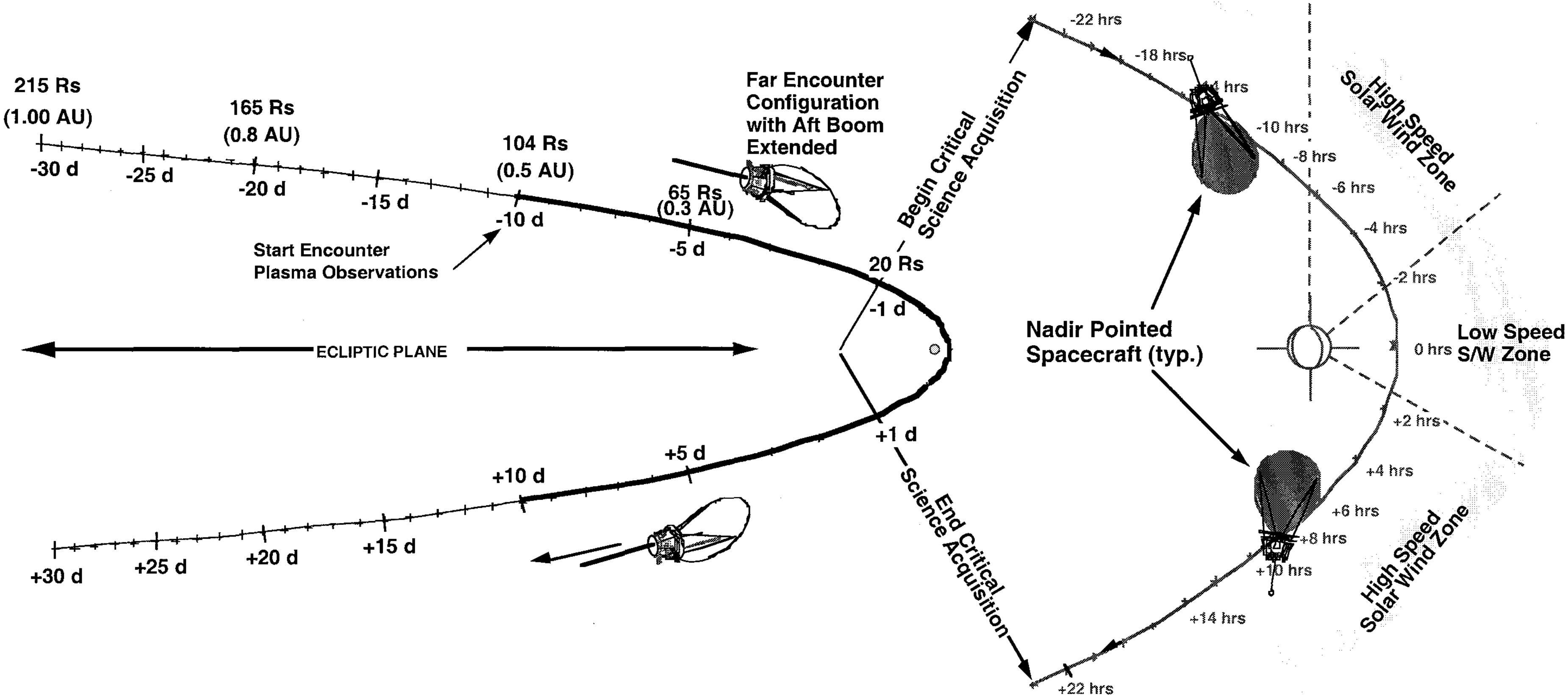

215 Rs
(1.00 AU)
165 Rs
(0.8 AU)
104 Rs
(0.5 AU)
65 Rs
(0.3 AU)
-30 d
-25 d
-20 d
-15 d
-10 d
-5 d
Far Encounter Configuration with Aft Boom Extended
Start Encounter Plasma Observations
Begin Critical Science Acquisition
20 Rs
-1 d
ECLIPTIC PLANE
+1 d
End Critical Science Acquisition
+5 d
+10 d
+15 d
+20 d
+25 d
+30 d
-22 hrs
-18 hrs
-10 hrs
-8 hrs
-6 hrs
-4 hrs
-2 hrs
0 hrs
+2 hrs
+4 hrs
+6 hrs
+8 hrs
+10 hrs
+14 hrs
+22 hrs
High Speed Solar Wind Zone
Low Speed S/W Zone
Nadir Pointed Spacecraft (typ.)
High Speed Solar Wind Zone

Figure 3

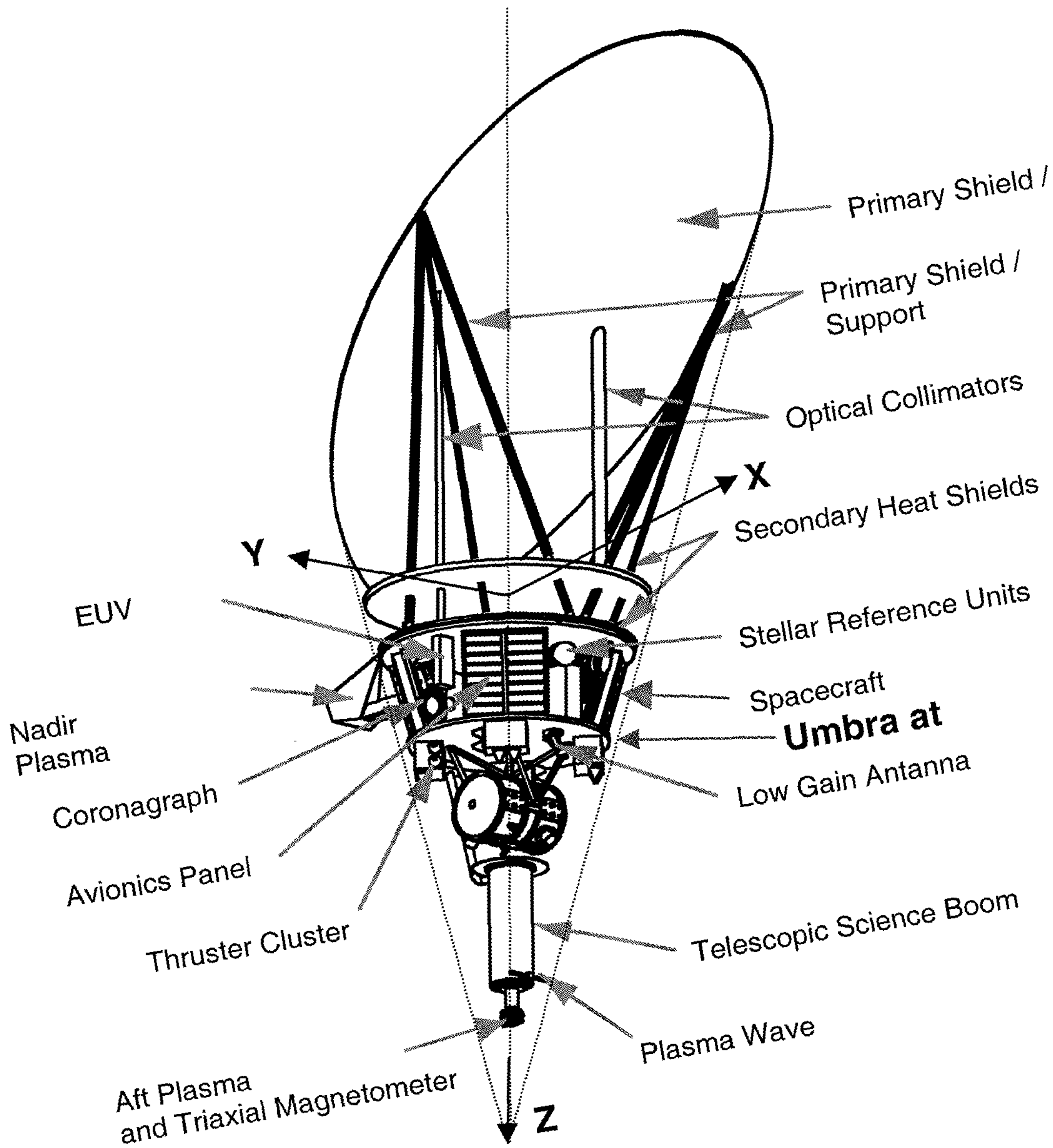

Primary Shield /
Primary Shield /
Support
Optical Collimators
X
Y
Secondary Heat Shields
EUV
Stellar Reference Units
Spacecraft
Umbra at
Nadir
Plasma
Low Gain Antanna
Coronagraph
Avionics Panel
Thruster Cluster
Telescopic Science Boom
Plasma Wave
Aft Plasma
and Triaxial Magnetometer
Z

Figure 4.

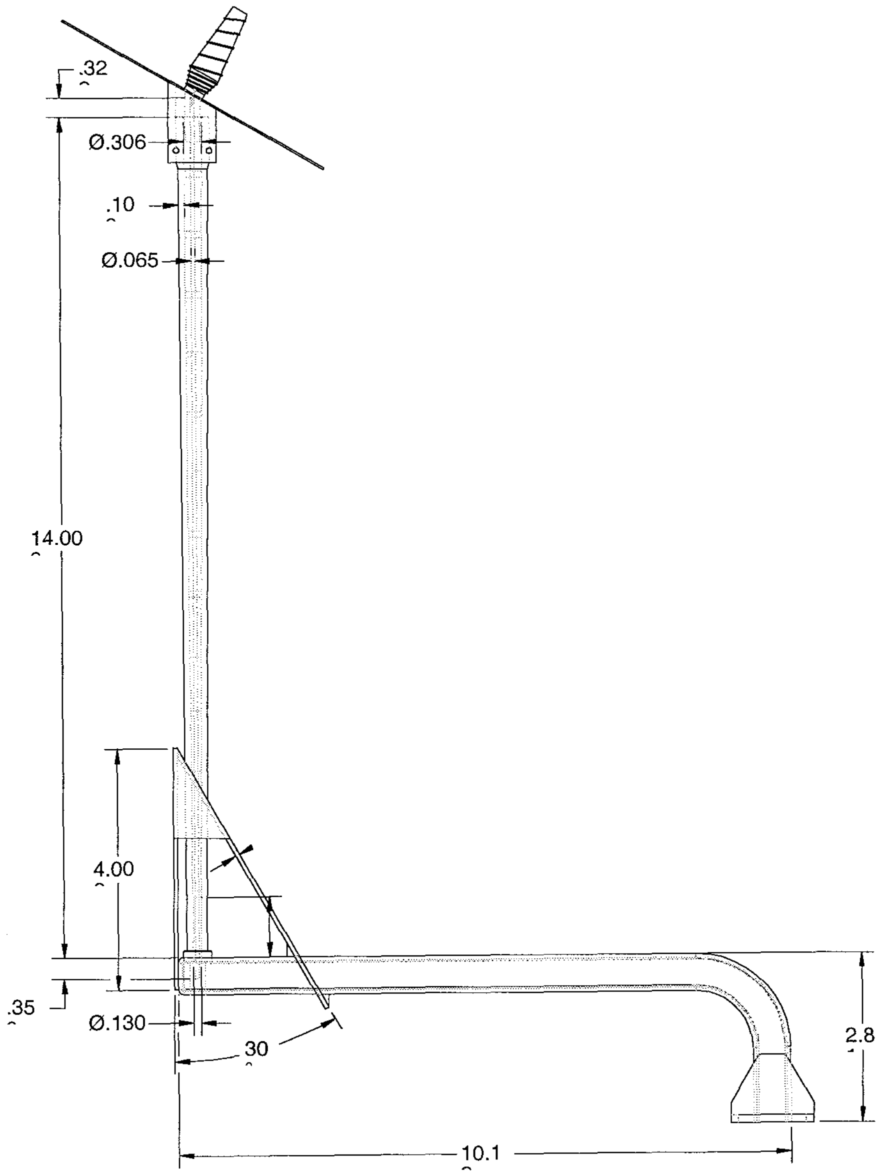


Figure 5.

the earth will be possible at all times near perihelion even when the shield is continually maneuvered to maintain precise nadir pointing. The high speed at perihelion (~ 300 km/s) is also suggested in Figure 3. The spacecraft travels from the north pole to the south pole of the sun in about 13 hours and travels from 1 AU to perihelion in just 30 days.

3. Spacecraft Technology Development

Before discussing some of the technology innovations and development, the detailed architecture of the spacecraft should be understood. Figure 4 is an isometric view of the spacecraft configuration that illustrates some of its major characteristics.
The upper section of the spacecraft is dominated by the large Carbon-Carbon TSA system. Below the TSA is the octagonal bus structure which holds the avionics of the spacecraft and some of the bus mounted instruments. The radioisotope power source mounts under the lower plate of the bus as shown. Finally, near the tip of the umbra is the aft instrument package mounted on a telescopic boom.

Thermal Shield Antenna (TSA Development)

An extensive materials development and testing program has been under way since the early 1990s [8]. The search for a lightweight shield material, that had the necessary thermal and mass loss properties, resulted in the selection of a Carbon-Carbon material that would be fabricated using a special densification technique [8]. Although materials testing continues to verify these desirable properties, a full scale shield fabrication program will begin in 2001 in order to deliver the flight shield in time to integrate it with the flight spacecraft.

High Temperature Antenna Feed

A high temperature RF feed is under development for the TSA antenna [10]. This feed is not shown in Figure 4 but is located at the focus of the parabolic antenna) must operate at about 1500 K at perihelion and has required an innovative materials development program. The results of that program [10] suggested that a ceramic matrix composite (CMC) material is the best candidate for the feed structure as shown in Figure 5.

The assembled CMC feed structure will be nickel plated to provide the necessary electrical conductivity at the operational temperature of 1500 K. The upper helical feed element will radiate the X-band signal that comes through the lower waveguide and coaxial vertical structure. A tungsten wire (that forms the helical element of the upper feed) passes through the coaxial section and its tip acts as an "emitter" (or in this case a receptor) of the signal in the waveguide. Fabrication of this feed system will be completed in late 2000. This feed is an integral part of the shield system.

Carbon-Carbon Optical Baffles

The tubes beneath the shield in Figure 4 that are parallel to the Z (nadir) axis of the spacecraft serve as optical collimators for the EUV and visible (magnetograph) imagers which must view the disc of the sun (i.e., nadir). The collimators have been designed to provide the correct field-of-view for the strawman instruments [14] and would be lined with a low reflectance material to provide the necessary optical attenuation. The collimators have been designed as "filled apertures" that maintain a constant temperature at their lower ends even though the range to the sun is varying near perihelion.

Nadir Viewing Plasma Spectrometer (NVPS)

Another unique technology development results from the scientific observational requirement (see 14, Appendix D) that one of the plasma spectrometers on the spacecraft must have an aperture in the nadir direction. This implies that the instrument must be compatible with the full solar flux (3000 suns) at perihelion. E. Sittler, et al (see [14], section 4) developed a unique design concept for this high temperature instrument system. A Carbon-Carbon "tent" shield (as shown in Figure 4 on the left side of the bus) was developed that would contain an electrostatic mirror beneath the shield. A pair of electrostatic mirrors form a reflection path for energetic plasma species to reach the plasma spectrometer located in the cooler region of the spacecraft bus. The two mirrors are approximately parallel and an angle (about 45 degrees) to the nadir direction forming a simple reflection system into the detector (see ibid). It is a fundamental scientific requirement that a complete plasma distribution function be observed at all times. This nadir viewing plasma spectrometer provides that view of the distribution function in the nadir direction. The remainder of the distribution function would be observed from instrument on the aft boom of the spacecraft.

Aft Viewing Plasma Spectrometer (AVPS)

Mounted at the tip of the aft boom shown in Figure 4 is another plasma spectrometer that can observe the plasma distribution function at all angles except in the near nadir direction. It is desired to observe as close to the nadir direction as possible to be complementary to the NVPS [13]. When the spacecraft is far from the sun (when the sun is nearly a point source) and the umbra is nearly cylindrical, the boom will be extended to locate the AVPS about 4 m from the bus. As the spacecraft approaches perihelion the boom is retracted, keeping the AVPS as close to the tip of the umbra as possible at all times. Figure 4 illustrates the fully retracted configuration of the aft boom (at perihelion).

The technology issue about the aft boom is the reliability of its deployment and retraction: Many booms have been developed for single time deployment but few have had

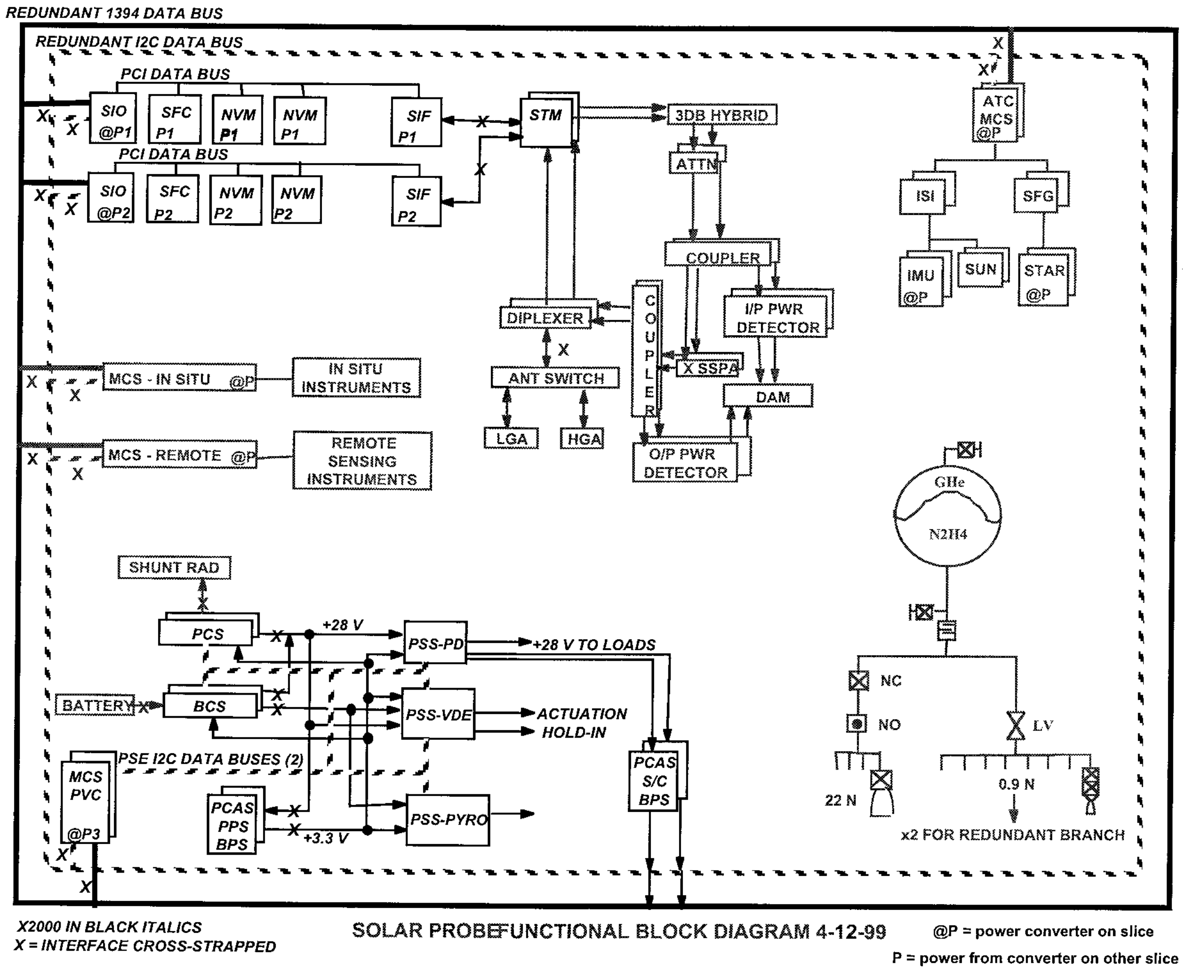

REDUNDANT 1394 DATA BUS
REDUNDANT I2C DATA BUS
PCI DATA BUS
SIO @P1
SFC P1
NVM P1
NVM P1
SIF P1
SIO @P2
SFC P2
NVM P2
NVM P2
SIF P2
STM
3DB HYBRID
ATTN
COUPLER
I/P PWR DETECTOR
DIPLEXER
COUPLER
X SSPA
DAM
ANT SWITCH
LGA
HGA
O/P PWR DETECTOR
ATC MCS @P
ISI
SFG
IMU @P
SUN
STAR @P
MCS - IN SITU @P
IN SITU INSTRUMENTS
MCS - REMOTE @P
REMOTE SENSING INSTRUMENTS
GHe
N2H4
NC
NO
LV
22 N
0.9 N
x2 FOR REDUNDANT BRANCH
SHUNT RAD
PCS
+28 V
PSS-PD
+28 V TO LOADS
BATTERY
BCS
PSS-VDE
ACTUATION
HOLD-IN
PSE I2C DATA BUSES (2)
MCS PVC @P3
PCAS PPS BPS
+3.3 V
PSS-PYRO
PCAS S/C BPS
X2000 IN BLACK ITALICS
X = INTERFACE CROSS-STRAPPED
SOLAR PROBE FUNCTIONAL BLOCK DIAGRAM 4-12-99
@P = power converter on slice
P = power from converter on other slice

Figure 6

the retraction requirement. A program is under way to address this question.

3. SPACECRAFT SYSTEMS REQUIREMENTS AND CONCEPTS

The Solar Probe mission will travel to a region near the sun where the environments are either quite unique or not well understood. (This is why the science from this mission is so important.) When traveling to an unknown environment, certain special design practices must be employed to increase the probability of survival. First, and foremost, the Solar Probe shield system must be well characterized and the design must be well validated. This validation process must recognize that thermal testing of the full scale shield in its operational environment is not possible with any facility on earth. Thus, only a high fidelity analytical validation is possible. (This is similar to the Galileo probe shield design which could not be tested as a full scale shield.) The extensive materials and components testing program discussed above is a key element of this validation process. Some of the other key system design concepts are discussed below.

Attitude Control Design Concept

The attitude control subsystem for the Solar Probe must maintain the precision pointing control (± 0.5 degrees) for the high gain antenna pointing. This range can never exceed the umbra design margin on the configuration of ± 1.0 degrees. The key concerns near perihelion relate to the reliability of maintaining sensor accuracy in the hostile environment near the sun. Figure 6 from [13] illustrates some of the block redundancy used to assure reliability at perihelion.

The blocks in the figure with "mirror images" behind them implies a block redundant element. The attitude computer (ATC) with its micro controller (MCS) will be given signals from the stellar reference (STAR) subsystem and its frame grabbar (SFG) that will be sufficient to determine the 3 axis stabilized attitude while away from the sun. However, when the STARs are within P ± 8 hours, it is predicted that they cannot be used as the only attitude reference source because of environmental problems. During that time interval, an inertial measurement unit (IMU) or gyro element will become the source for the attitude reference. The issue here is that gyros drift. The unit proposed for the Solar Probe at this time is similar to the gyro element that is flying on the Cassini mission - the Hemispherical Resonance Gyro (HRG). This HRG gyro package (four skewed gyros providing internal redundancy) has a drift rate specification of about 0.001 degrees/hour per axis. Thus, in the 16 hours of drift after the final "calibration" with the STARs, the gyro would drift a total angle of about 0.016 degrees per axis. With this high quality gyro, maintaining the ± 0.5 degrees pointing control during the 16 hour interval should be straightforward and the gyro drift would not be the key attitude error source. The whole issue of fault tolerance to assure no attitude failures is obviously crucial for the Solar Probe. Considerations of whether simple block redundancy is sufficient to protect against all of the possible environmental effects that could perturb the attitude control, remains to be studied in depth. Referring back to Figure 6, precise hand-shaking must exist between the system flight computer (SFC) and the ATC through the high speed (1394) bus shown in this avionics architecture that is nearly identical to the X-2000 concept under development at JPL [13]. More information about this architecture can be found in the [ibid].

Another important parameter of the attitude control design is the stability of the spacecraft pointing. The nadir viewing optical instruments will require relatively long exposures of about 0.1 seconds because they will be very narrow band imagers. The instruments require stability that would keep the attitude rate of the spacecraft below 1 pixel of smear during the exposure. This stability must be less than 100 μ radians in 1 second as specified in the AO [13]. One of the determining factors of the attitude rate is the minimum impulse bit of the attitude control thrusters. A thruster development program for the Europa Orbiter mission [15] is expected to produce precise minimum impulse bit thrusters (MIT) with a small enough bit to allow the necessary stability even though the thruster force level is reasonably high (0.9N). A high thrust is necessary to maintain adequate attitude control during the near perihelion pass in the presence of the high photon pressure. This pressure on the TSA system will cause an overturning moment on the spacecraft that must be compensated for by the thruster system. Thus, the thrusters must be large enough to account for this moment but have a small enough MIT capability to provide the stability for the optical instruments.

Telecommunications Subsystem Design Concept

An X-band telecommunications subsystem for the Solar Probe introduces many unique design issues. First the existence of the sun in the telecommunications path must be considered at both ends of the path. On the ground the sun gets in the beam width of the ground station causing "hot body" noise which must be accounted for but is reasonably predictable. At the spacecraft the existence of the coronal perturbations on the link is not so predictable. (A K_a band carrier frequency would reduce the effects of these perturbations but deleterious effects of the weather at the stations may be even more unpredictable than the solar corona!) In an effort to better understand the effects of the solar coronal perturbations on an X-band link a series of conjunction experiments have been completed to capture a high quality X-band signal from the Mars Global Surveyor (MGS) spacecraft when it was in conjunction with the earth [12]. With this geometry the MGS signal passed through the corona on the way to the earth and was perturbed by the corona. This carrier signal containing these perturbations or "scintillations" was recorded with modern wide-band digital recorders (the intermediate frequency (IF) signal at the station was recorded with a new prototype recorder known as the Radio Science Recorder (RSR) with a higher bandwidth that captures not only the carrier but also the telemetry subcarrier on the IF signal). The IF is a "raw" data signal existing before the station receiver so it has the most information and is not modified by the receiver itself. The preliminary analysis of this data suggests a new time dependent phenomenon that had not been seen in previous data. Previous work had assumed that the signal was noisy but that the noise was stationary and classical stationary analysis techniques could be applied to the analysis. This is only partially true. What was found in this data was two distinct characteristics. First, and most important, was that a major component of the signal was non-stationary. That is, large transients or "fades" of 20-30 dB occurred every few minutes in the data that had a duration of a few seconds at most. The depth of these fades was so large that "covering" them with design margin would be virtually impossible. Because the fades were so infrequent, simply ignoring them and accepting the loss of data during these fades and the time to relock the signal would be a reasonable design decision. In the past large margins (~ 10 dB) were introduced in the link design in an effort to recover all data even in the presence of these fades. This was very conservative and penalized the telemetry rate at perihelion by a factor of 10! Although a final decision has not been made, a tentative design philosophy ignores these fades and accepts the periodic losses as inevitable and acceptable. Further analysis of these conjunction data may suggest techniques that would even capture most of the data during these enormous fades. The second characteristics seen in the time domain analysis of the FSR data is that there is indeed a background perturbation or scintillation which appears to be very stationary and lends itself to classical frequency analysis techniques. Thus, margin in the link design to cover this stationary noise is justifiable because this background noise would occur continuously near perihelion.

4. Current Program Plan

With the initial stages of the technology development program completed and with the Announcement of Opportunity [13] proposals already received by NASA, the Solar Probe program is making progress toward a 2007 launch date. The success of the shield material development program [9] will lead to the issuance of RFP for shield development in early 2001. A fabrication contractor will be selected by mid 2001 with the fabrication, assembly, and testing of the shield completed by early 2004 at the time of the spacecraft critical design review (CDR).

The Solar Probe spacecarft will be designed and fabricated by an industry contractor for JPL. The completed shield will be delivered to the contractor prior to the beginning of the final spacecraft fabrication (CDR). The selection of the flight's instruments will occur at the time of the release of the shield RFP when the relationship of the shield and spacecraft concepts will be sufficient to understand any major interface issues with the instruments. The instrument development and the shield fabrication will be contemporary allowing interaction with the shield fabrication and the instrument design concerned with the interfaces to the shield such as the optical instruments that must have baffles that pass through the shield system (See Figure 4). Both the shield system and the instruments will then be available, with sufficient schedule margins, at the time of the final assembly and testing of the spacecraft system. The testing of the entire system will include environmental testing under the flight simulated environments such as vibration and acoustics but with the exception of flight thermal-vacuum testing. There is no test facility in the world that can simulate the 3000 suns flux at perihelion in vacuum. Thus, the full scale thermal vacuum tests will be done at the highest flux possible (the large JPL test chamber can generate about 9 suns). It is expected that this testing will be sufficient to understand the thermal interactions of the spacecraft subsystems. The thermal shield design validation will be accomplished at the coupon testing level of the materials combined with the highest fidelity analytical modeling. This process is expected to confirm the reliability of the shield design in its operational environment at perihelion.

5. Conclusions

The technology is in reach to enable a mission to the region inside the solar corona. Modern materials such as carbon-carbon for the shield system have only recently been tested in the environments that will be experienced at perihelion. With this new understanding of the shield materials, the TSA system will be designed with a high level of confidence that it will function very well at perihelion. This three axis stabilized concept for the spacecraft will also rely on new inertial sensors such as the HRG to maintain the necessary pointing control near the sun over a period of many hours. The stability of the attitude control will be enhanced by the MIT thrusters that will enable the long exposures of the optical instruments. Those same thrusters will have a large enough peak force capability to control any overturning moment caused by the photon forces at perihelion. New instrument technologies [14] will enable nadir viewing capability for plasma spectroscopy and optical observations while it is expected that the instruments will have low enough mass and volume to be accommodated in the light weight spacecraft that is required for the Solar Probe mission.

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