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Small Solar Probe

J. C. Preble* and A. C. Tribble†
Rockwell International Corporation, Downey, California 90241-7009

The Small Solar Probe is a spacecraft that would utilize a Jupiter gravity assist trajectory to carry a solar physics instrument complement within 4 solar radii of the sun's center. The unique characteristics of the Small Solar Probe mission place significant challenges on spacecraft configuration, electrical power, telecommunications, and especially thermal control. This paper summarizes the main aspects of the mission and presents a conceptual spacecraft design. Key aspects of the point design are the deployable instrument platform, the integrated primary heat shield/high-gain antenna, the four-tiered thermal control subsystem, and the use of a solar array rather than a radioisotope thermoelectric generator as the power source. The results indicate that the Small Solar Probe mission is technologically feasible, can be performed within the life-cycle cost cap of $400 million, and can be ready for a year 2000 launch.

Introduction

TO date, solar studies have relied on either remote-sensing observations or in situ data collected several million kilometers from the sun's center. An in situ exploration of the near sun environment would provide a more thorough understanding of the sun's influence on the Earth and its climate and would also provide insight for application to other stellar systems.[1–3] Close-approach in situ observations were proposed as early as 1963, but did not receive in-depth analysis until the late 1970s.[4,5] In 1989, NASA organized a solar probe science team that defined a mission concept for a spacecraft that would pass within 4 solar radii (R_s) of the sun's center.[6,7] This concept utilized a conical heat shield with the apex pointed at the sun, and was launched from a Titan IV Launch Vehicle (LV). The science team also defined a set of science objectives for the mission (Table 1) and recommended a number of science payloads (Table 2).[8] These studies were expanded in the early 1990s.[9–13]

More recently, in the face of decreasing science budgets, the solar probe mission payload was decreased in both number and complexity in an attempt to devise a spacecraft that could be launched from a Delta II 7925 launch vehicle (Table 3). The conical heat shield was also replaced with a parabolic heat shield that could double as the spacecraft antenna, the objective being that the new Small Solar Probe (SSP) mission would have a life-cycle cost less than 50% of the 1989 concept. The requirements of the revised spacecraft concept bear several similarities to those of the Pluto Fast Flyby mission. Both spacecraft have extreme mission requirements, travel far from the Earth [the SSP utilizes a Jupiter gravity-assist (JGA) trajectory maneuver], and must utilize a low data rate to keep mass and power low. These two programs are currently the subject of international discussions between the United States and Russia, forming a "fire and ice" concept for solar-system exploration. In 1994, Rockwell International conducted a design study to develop an SSP concept (Fig. 1). This study concluded that an SSP was feasible. That is, the SSP could be 1) packaged within the mass and size constraints of a Delta II 7925 launch vehicle, 2) ready for a year 2000 launch, and 3) completed within NASA's life-cycle cost goal. The key features of the mission analysis and spacecraft design trades that were performed to support these conclusions are described in the sections that follow.

Mission Analysis and Operations

The SSP mission is bounded by three driving requirements: 1) perihelion of $4R_s$, the closest approach thought technologically feasible; 2) heliocentric polar orbit, to obtain out-of-ecliptic data and enable; and 3) sun–probe–Earth angle of 90 deg at perihelion, to ensure real-time downlink of data to Earth during the perihelion pass. Previous mission analysis studies conducted by NASA Jet Propulsion Laboratory (JPL) illustrated that although several trajectories can provide the required characteristics, a ballistic trajectory using a JGA is the most desirable. It uses the gravity-assist techniques proven during the Ulysses project, and it provides short flight times and lowers program cost and risk in comparison with other options.

Mission analyses were performed (Figs. 2 and 3) to define the 1) ecliptic and noncliptic motion, 2) probe distance profiles,

Table 1 Solar probe science objectives

Coronal structure
Coronal heating and acceleration of the solar wind
Plasma turbulence within the coronal envelope
Acceleration, storage, and transport of energetic particles
Sources, skins, and dynamics of interplanetary dust

Table 2 Original solar probe instrument suite

Instrument	Mass, kg	Power, W	Telemetry, kbit/s	Time resolution
Fast plasma	8	8	5	5 ms
3D ions	12.5	17	10	1 s
3D electrons	3.5	3	5	1 s
Magnetometer	5.5	6	2.5	40 ms
Plasma wave	19	11	15	5 ms
Thermal composition	12	8	5	1 s
Suprathermal composition	13	9	10	1 s
Medium-energy particles	10	8	10	1 s
High-energy particles	10	8	1	1 s
Coronal spectral imager	25	15	5	30 min
Neutron	10	5	1	10 s
Dust	5	5	0.5	10 s
Total	133.5	103	70	N/A

Table 3 Revised solar probe instrument suite

Instrument	Mass, kg	Power, W	Telemetry, kbit/s
Plasma detector	3.7	2.7	1.2
Magnetometer	2.5	1.5	0.5
Plasma wave	4.25	4	0.5
Energetic particles	2.5	2.5	0.5
Coronal photometer	0.75	0.5	0.1
Hard x rays	0.7	1	0.3
Shared DPU	3	3	——
Total	17.4	15.2	3.1

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*Study Manager, Advanced Programs, Space Systems Division.
†Project Scientist, Advanced Programs, Space Systems Division. Senior Member AIAA.

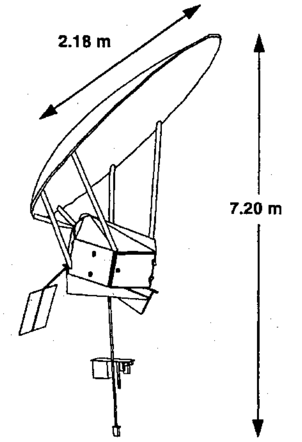

Fig. 1 SSP spacecraft.

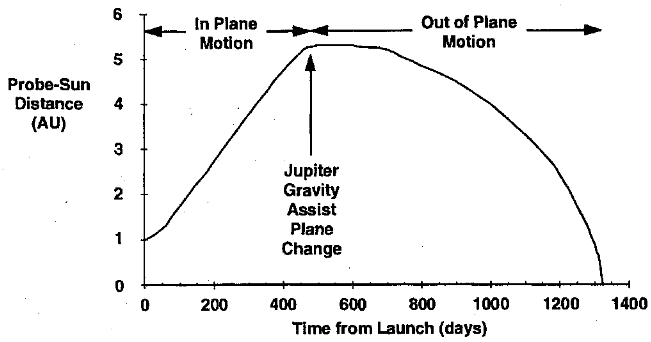

Fig. 2 Probe–sun distance during flyout and approach.

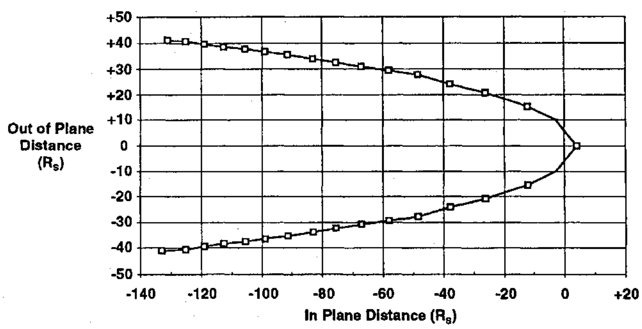

Fig. 3 Spacecraft trajectory during perihelion; data points are at 1-day intervals.

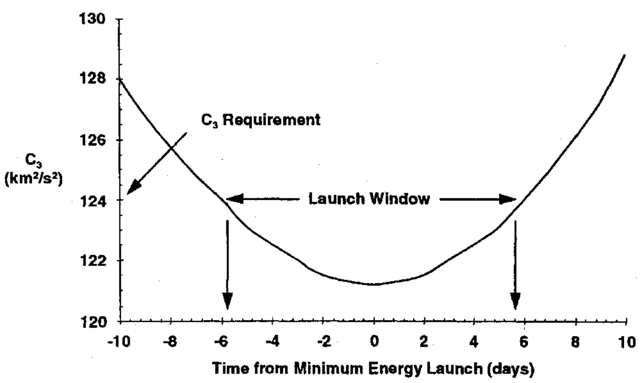

Fig. 4 Eleven-day launch window required for $C_3 < 124\ km^2/s^2$.

3) sun–probe–Earth geometry, and 4) sensitivity of the C_3 requirement. The C_3 requirement was found to be quite sensitive to launch date (Fig. 4), but relatively insensitive to arrival date. These analyses indicate that a launch window of 11 days is required to maintain a $C_3 \leq 124\,km^2/s^2$. The C_3 requirements for later launch opportunities are roughly the same. Based on available Delta II data, the probability of being able to launch within an 11-day window was estimated to be acceptably high at $\geq$0.9991 (Ref. 14).

Table 4 SSP mission requirements

Phase	Requirement
Launch	Aug. 11, 2000 Delta II 7925 Star 30C fourth stage $C_3 < 124\,km^2/s^2$
In-plane cruise	Four TCMs Weekly contact to 34-m DSN
JGA	Nov. 27, 2001 $v_{inf} > 13$ km/s 90-deg plane change
Out-of-plane cruise	Two TCMs Weekly contact to 34-m DSN
Perihelion	Solar pointing at $P - 15$ days Full-time science at $P - 5$ days Daily contact to 34-m DSN Real-time return to 70-m DSN ($P - 24$ h to $P + 24$ h) $4R_s$ perihelion March 27, 2004 Store perihelion data
Postperihelion	Return stored perihelion data
Mission	1324 days

Mission operations were defined for the cruise and perihelion phases of the mission. The primary events occurring during cruise were four trajectory correction maneuvers (TCMs) and the nonpowered JGA swingby. Telecommunications coverage during cruise is accomplished using the 34-m Deep Space Network (DSN) stations. Continuous coverage is assumed for the TCMs (TCM -10 days through TCM $+$ 1 day). Weekly coverage for 5 h is assumed for other portions of the cruise. Calibration and checkout of all instruments are planned every six months.

Perihelion (P) operations occur for 20 days from $P - 10$ days to $P + 10$ days. The spacecraft must be protected from direct exposure to the sun during this phase and is kept in the umbra of the heat shield. Primary data collection occurs during this period, and the instruments are operated continuously for those 20 days. From $P -$ 10 days to $P - 1$ day data are collected, stored in the solid-state recorder, and downlinked daily to the 70-m DSN stations. From $P - 1$ to $P + 1$ day, data are both recorded and downlinked in real time. The stored perihelion data are returned following $P + 1$ day. From $P + 1$ to $P + 3$ days, data are stored and returned daily. On account of spacecraft pointing constraints, the telecommunications link cannot be closed from $P + 3$ days to $P + 10$ days. All data are stored during this 7-day period and returned when the link is closed at $P + 10$ days. These mission requirements are summarized in Table 4.

Spacecraft Design

A critical design requirement that affects all spacecraft subsystems is the launch vehicle. As specified by NASA JPL, the launch vehicle is the three-stage Delta II 7925 with a Star 30C fourth stage. The capability quoted by McDonnell Douglas is 188 kg for a C_3 of $124\,km^2/s^2$. The baseline concept is consistent with this capability and provides a margin of 22.2 kg. Although this margin is considered acceptable, control of the spacecraft mass properties is critical and will be an area of special concern throughout this program (Table 5).

Configuration

The spacecraft configuration meets instrument location and field-of-view requirements, provides compatibility with the launch vehicle, and accommodates the spacecraft subsystem requirements (Fig. 5). The plasma detector, energetic-ion instrument, coronal photometer, plasma-wave instrument, and shared data-processing unit are located on a deployable instrument platform approximately 130 cm from the spacecraft bus. The magnetometer is located along the same deployable boom, but approximately 70 cm further from the bus. This location reduces the interference detected by the magnetometer that may be produced by the other instruments and the spacecraft subsystems. The hard-x-ray instrument is located within the spacecraft bus to eliminate possible shielding by the spacecraft structure and components.

Table 5 SSP mass properties

Subsystem	Mass, kg
Instrument	17.4
Heat shield/antenna	20.0
Structure/mechanisms	21.3
Electrical power	28.5
Communication and data handling	8.5
Telecommunications	11.4
Attitude determination and control	9.3
Propulsion (dry)	10.2
Thermal control	18.5
Subtotal	145.1
Propellant	20.7
Spacecraft	165.8
LV capability	188.0
Margin	22.2

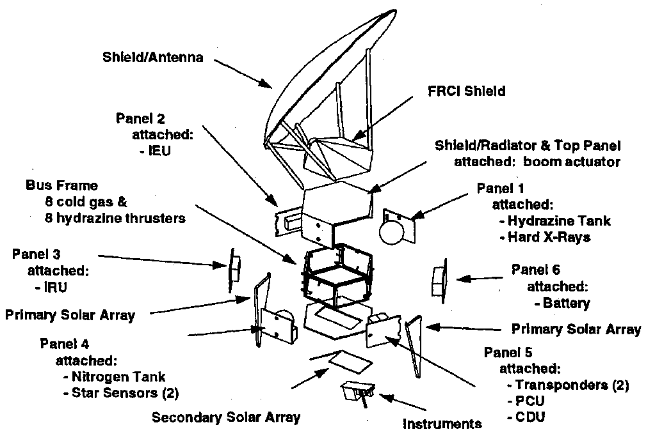

Fig. 5 SSP spacecraft (exploded view).

Three launch-vehicle fairings with different diameters were considered: 2.44, 2.90, and 3.05 m. All required 1.52-m cylindrical extensions. The 2.44-m-diam fairing was selected and provides the highest payload capability without significant penalty in usable volume.

The primary challenges of accommodating the spacecraft subsystems involved the design of the combined heat shield/antenna and the placement and orientation of the solar array. The heat shield/antenna is a full paraboloid with a circular projected diameter of approximately 2.1 m. The instruments are deployed at a 14.5-deg angle to maximize the distance from the spacecraft bus.

Heat Shield/Antenna

The design combines the functions of the primary heat shield and the high-gain antenna into a single component to reduce mass and power. Consequently, its performance must meet the requirements imposed by the structure-and-mechanisms, telecommunications, and thermal control subsystems. Trade studies were conducted to evaluate materials, configurations, and fabrication methods. The paraboloid geometry maximizes antenna performance and reduces mass loss through sublimation, which could contaminate the data. Its size is maximized by tilting the shield in the launch vehicle by 14.5 deg, which allows the longest possible deployed payload platform. The single-piece, monocoque, carbon–carbon (C–C) shell can be made from a chemical vapor deposition process or from high-stiffness graphite fabric prepreg using a carbonization and liquid reimpregnation cycle. A similar C–C perimeter frame is bonded onto the shell for stiffness and truss attachment. The truss tubes are made from T300 C–C fabric to maximize conductive resistance between the shield and bus.

The combined heat shield and antenna is used as an X-band offset reflector antenna. The antenna provides 39.9-dB gain for uplink and 41.4-dB gain for downlink. Losses because of asymmetry, polarization, feed horn, spillover, reflectivity, surface roughness, and tip deflection were considered.

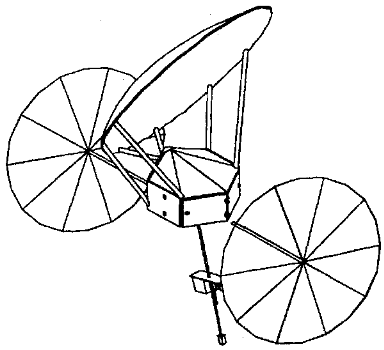

Fig. 6 SSP spacecraft, showing the primary solar arrays.

Structures and Mechanisms

The bus structure is designed to efficiently interface with the launch vehicle, spacecraft subsystems, and instrument payload. A hexagonal configuration provides good attachment points at corners for the launch-vehicle truss and primary shield, and flat surfaces for internal subsystem attachment. Mass is minimized by using minimum-gauge aluminum honeycomb sandwich panels and a graphite–epoxy framework. The payload platform is stowed as far as possible into the bus to reduce bus height and increase the size of the primary heat shield. A nonmagnetic Astro Aerospace bi-stem boom extends the payload platform during the mission.

Electrical Power

The electrical power subsystem (EPS) provides the electrical power, pyro switching, and regulated voltages to the spacecraft bus and instruments. Because the spacecraft is subjected to a 5-order-of-magnitude variation in the solar flux, radioisotope thermoelectric generators were the power supply of choice for previous design studies.[15] Our concept examined the possibility of performing the mission with GaAs solar arrays, a power conditioning unit, a lithium carbon monofluoride battery, and an instrument power-supply module located on the instrument platform (Fig. 6). The solar arrays are incapable of surviving the perihelion pass and are stowed at $P - 10$ days. At this point, a secondary power source is enabled. The preferred solution consisted of a second, high-temperature solar array that would be offpointed from the sun to maintain a temperature of $< 250°$C. The second solar array would be used from $P - 10$ days until $P - 38$ h. At this point, the second array would also be stowed, and the spacecraft would be powered by the battery until $P + 38$ h, at which time the process would be reversed. Although the baseline photovoltaic (solar array) method is more mature, solar concentrators using optical waveguides and thermoelectric generators both warrant further study.

The LiCF battery is baselined at 43 A-h and has internally redundant cells for improved reliability. The power conditioning unit and the instrument power-supply module are based on advanced technology in electronic hybrid packaging to meet mass and power targets.

Command and Data Handling

The command and data handling (C&DH) subsystem provides the processing, data recording, storage, and playback, and capability for the spacecraft bus components and provides instrument control. C&DH supports the telecommunications subsystem with data formatting and buffering, timing, signal conditioning, I/O interfaces, and control of the spacecraft. The C&DH subsystem components are the integrated electronics unit (IEU) and an instrument data transfer module. The IEU has an SCI internally redundant modular architecture (featuring a Loral R6000SC). The data bus is a 1773 protocol tolerant to electromagnetic interference. The processor has error detection and correction capability with a 256-kbit boot programmable read-only memory (PROM), 1 Mbit of PROM, and a 1-Mbit static random access memory to handle power on reset functions. The use of advanced technology in electronic packaging and low-power interface drivers is required to achieve the mass and power targets.

Telecommunications

The mission requirements to transmit science data in real time during the perihelion pass and at the large Earth–Jupiter distance during the gravity assist maneuver challenge the X-band telecommunications design.[16] Assessing the effects of amplitude and phase scintillation created by transmission through the solar corona and providing reliable communication returning perihelion data in real time were the focus of this effort. The mission geometry was evaluated to assess the expected magnitude of the scintillation effects, and numerical methods were used to quantify the performance of the communication channel. The analytical results show that perihelion data may be returned in real time at 4 kbit/s with a bit error rate of 1×10^{-4} (the initial requirement from the science community), using a concatenated coding scheme with NASA standard $r = \frac{1}{2}$, $L = 7$ convolutional code, and a (255, 223) Reed–Solomon code resulting in more than 5 dB of system margin. Numerical analysis was also used to assess the effect on link performance of using more aggressive coding schemes. The improved coding scheme that will be validated on the Cassini and Mars Pathfinder missions, an $r = \frac{1}{6}$, $L = 15$ convolutional code with a (1023, 959) Reed–Solomon code, will allow a telemetry bit error rate of 1×10^{-6} at 4 kbit/s with more than 5 dB of system margin in the presence of the expected worst-case amplitude scintillation. The mission requirements are achieved through the use of a 2.1-m high-gain antenna/heat shield that provides better than 41-dB gain, a 1.5-W solid-state amplifier, and related telecommunications components.

Attitude Determination and Control

The attitude determination and control subsystem (ADCS) features low system complexity with selected internal and block redundancy. The design utilizes reaction jets to provide control torques and relies upon a Hughes–Danbury optical star tracker and an inertial reference unit (IRU) to provide guidance information. The star tracker provides a reliability of 0.985 over 4 years and is updated at the rate of 10 Hz. The Delco hemispherical resonator gyro IRU has a reliability of 0.9997 over 4 years and a bias stability of 0.006°C/h (3σ) at constant operating temperature.

The solar-pressure torque, not including solar wind, during perihelion passage is the dominant disturbance (over two orders of magnitude higher than the next highest torque—gravity gradient). The system has been sized to provide impulse to acquire and maintain attitude during perihelion. During $P \pm 10$ days, ADCS must keep the spacecraft in the heat-shield umbra and simultaneously maintain Earth pointing for communications. Proper spacecraft orientation is maintained through a combination of simultaneous rotation about the sun–probe line and about the normal to the sun–probe line, such that Earth pointing is always maintained. Analysis has indicated that the spacecraft can remain in the shield umbra and track the Earth from $P - 9$ days to $P + 3$ days. Consequently, our mission operations concept terminates the downlink at $P + 3$ days and resumes it at $P + 10$ days.

Propulsion

The propulsion subsystem provides impulse for translation maneuvers (Δv) and attitude maneuvers. Translations maneuvers are required for the four TCMs. The total Δv requirement for the TCMs is 200 m/s and is allocated among the four maneuvers. Attitude maneuvers are required to despin the spacecraft following separation from the launch vehicle, to provide attitude maintenance during cruise, and to acquire and maintain attitude during perihelion operations.

The design combines the features of a blowdown monopropellant hydrazine system with those of a cold-gas GN_2 system into a single system. The monopropellant portion provides impulse for Δv maneuvers and meets the high-torque requirements of the ADCS, while the GN_2 system meets the low-torque requirements. The design is similar to a concept being considered for the Pluto Fast Flyby.

Thermal Control

The design concept for the thermal control system is a four layered passive system (Fig. 7). It has evolved within a domain of competing requirements to minimize mass, mechanical complexity, mass loss at $4R_s$, bus interior temperatures at $4R_s$, and heater power throughout the entire mission while maximizing antenna aperture. The system includes a C–C primary heat shield to reflect the incident solar load, a secondary infrared (IR) shield to absorb and radiate IR backload from the primary shield, a tertiary shunt and radiator, and a quaternary insulation shield.

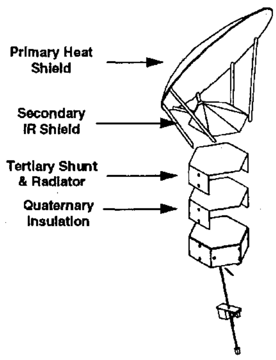

Fig. 7 Thermal control subsystem.

Modeling in I-DEAS/TMG, TRASYS, and SINDA shows that the concept is both thermodynamically feasible and materially manageable. The maximum shield temperature expected, 2380 K, should not give rise to an unacceptably high mass loss on the outer surface. By interleaving thermally conductive layers the thermal control system will maintain required bus temperatures both at solar encounter and during interplanetary cruise.

Technology Development

Several technology development projects were identified as means of reducing program risk. There is some synergy between those technology development items identified by the Pluto Fast Flyby mission, but several projects are unique to the SSP. The following technology development projects are recommended: 1) C–C heat shield/antenna, 2) attitude-control simulation, 3) integrated multilevel heat shield, and 4) analysis of data transmission in the presence of amplitude scintillations. In addition, the issue of which radiation hardness level to enforce, in view of the possibility of a mission-ending solar proton event during perihelion, requires further examination.

Conclusions

The SSP mission provides unique challenges in several key areas: mission design, configuration, electrical power, telecommunications, and thermal control. Our analyses indicated that 1) the mission is technologically feasible, 2) the conceptual design is consistent with the Delta II launch vehicle with adequate mass margin, and 3) two flight systems can be produced within the $130 million cost goal. The key to the technical success of the mission is the development of the parabolic C–C heat shield/antenna. Four technology development projects have been identified as means of mitigating risk. Successful completion of these studies will contribute to the eventual success of the SSP mission.

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