# Integrated Thermal Vacuum Testing of the Solar Array Cooling System for Parker Solar Probe

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Parker Solar Probe (PSP) will explore the inner region of the heliosphere through in situ and remote sensing observations of the magnetic field, plasma, and accelerated particles. PSP will travel closer to the sun (9.86 solar radii  $[(R_s)]$ ) than any previous spacecraft in order to obtain repeated coronal magnetic field and plasma measurements in the region of the sun that generates the solar wind. The baseline mission will entail 7 years from launch in 2018 until the completion of the 24th orbit; if delays necessitate, a backup 8-year, 26-orbit mission will be flown, with launch in 2019. During its lifetime, the spacecraft will be exposed to wide-ranging thermal environments, from the cold of Venus eclipse to exposures to the sun's corona, which produces a perihelion solar constant in excess of 480 suns. Spacecraft power is generated using photovoltaic solar arrays that are actively cooled by the solar array cooling system (SACS), manufactured by Hamilton Sundstrand, Windsor Locks, CT. This paper will describe the equivalent "test-like-you-fly" environments that were simulated and the results achieved during the SACS qualification and thermal design verification vacuum testing that took place at Goddard Space Flight Center between 1 March and 16 March 2017.

# Nomenclature

APL	=	Johns Hopkins University Applied Physics Laboratory
AU	=	astronomical unit (the distance between the center of the Earth and the center of the sun)
C-1	=	thermal vacuum test configuration 1
C-2	=	thermal vacuum test configuration 2
CSPR	=	cooling system primary radiator
GSFC	=	Goddard Space Flight Center
ISO	=	latching-type isolation valve (designated 1, 2, or 3)
I&T	=	Integration and Testing
ITVT	=	integrated thermal vacuum test
m	=	meter
MLI	=	multilayer insulation
MOC	=	Mission Operations Center
PBA	=	Phase B accumulator
PSP	=	Parker Solar Probe
RIU	=	Remote Interface Unit

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R <sub>S</sub>	=	Solar Radii
SA	=	Solar Array
SACS	=	solar array cooling system
SES	=	Space Environment Simulator
SOC	=	Science Operations Center
TPS	=	Thermal Protection System
TSA	=	truss structure assembly
V	=	volt
W	=	watt

# I. Introduction

**P**ARKER Solar Probe (PSP) will be launched from Cape Canaveral Air Force Station on a Delta IV expendable launch vehicle, including a third stage. Immediately after separation from the third stage, the spacecraft will begin the post-separation sequence, which includes spacecraft orientation relative to the sun and Earth, solar-array deployment, and partial activation of the cooling system (two platens and two of the four radiators). The time to first perihelion will be devoted to early operations, including:

- Launch correction maneuver
- Subsystem checkouts
- Deployments
- Instrument checkouts and in-flight calibrations
- Warm-up and activation of the remaining two radiators
- Preparation for first solar encounter period
- First Venus flyby to target for the first solar perihelia

Approximately 3 months after launch, PSP will conduct the first of 24 solar encounters, with perihelion of this first orbit at 35.7 solar radii ( $R_s$ ). The six Venus flybys planned for the mission will decrease the perihelion of subsequent orbits, leading to an eventual minimum orbit of 9.86  $R_s$  for the final three encounters.

PSP is a three-axis stabilized spacecraft, and its most prominent feature is the thermal protection system (TPS) designed to protect the spacecraft bus and most of the payload within its umbra during solar encounters. The TPS will always be pointed toward the sun, except during the cooling system activation, communications, thermal control, specified instrument calibrations, pre-Venus eclipse maneuvers, and the trajectory-correction maneuvers that occur at solar distances > 0.7 AU.

The central feature of each solar orbit will be the solar encounter period, centered roughly  $\pm 5.5$  days around perihelion when the observatory is within a solar distance of 54 R<sub>s</sub>. During this time, the observatory will primarily be devoted to the scientific measurement campaigns. Because communications with the observatory will be limited during solar encounter, the science data and the bulk of housekeeping data will be stored on a solid-state recorder. After each solar encounter (solar distance > 0.25 AU), mission highlights include cruise/downlink segments, science data downlinks, spacecraft-to-ground communications, and flight-path correction maneuvers.

The Deep Space Network will be used to communicate with PSP to collect data required for navigation. Mission operations will be conducted at the Johns Hopkins University Applied Physics Laboratory (APL) from a single Mission Operations Center (MOC). The MOC will be responsible for the operation of the spacecraft and for providing the necessary data to the instrument teams, such as attitude, ephemeris, and precise time. PSP uses a decoupled approach that allows each instrument team to manage its own instruments, including commanding and telemetry management. The MOC is responsible for transmitting commands provided by the Science Operations Centers (SOCs) to the spacecraft and providing science telemetry data to the SOCs in a timely manner. This decoupled approach was successful during previous spacecraft operations controlled from APL (TIMED [Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics], MESSENGER [MErcury Surface, Space ENvironment, GEochemistry, and Ranging], STEREO [Solar TErrestrial Relations Observatory], and Van Allen Probes programs), allowing for simpler interfaces between spacecraft and science operations mission configurations.

#### **II.** Test Objectives

The overarching objective of the integrated thermal vacuum test (ITVT) described in this paper is to verify the thermal design of the PSP spacecraft before launch by exposing the spacecraft's solar array cooling system (SACS) to environments simulating those expected to be encountered during flight. Thermal boundary conditions relevant to critical SACS operation will be provided by the TPS simulator, the spacecraft simulator, and the warming plate located between the central spacecraft and platens. Tests will be executed for the two solar-array deployment

configurations characteristic of near- and distant-helios operation. All of the operational aspects of the SACS expected in flight, including transient operations (i.e., radiator activation, launch corrections, and Venus eclipse), will be replicated and verified during the ground test. The most visible portion of the PSP spacecraft is the SACS, which is illustrated in Figure 1. The SACS is a pump-driven closed-loop single-phase system that uses water under pressure as the working fluid to actively cool solar cells, which are mounted to custom-made platens, and transport the waste heat to either two or four active radiators.



Figure 1. The PSP SACS.

The major components of the ITVT hardware are the flight TPS simulator, the four flight primary radiators, two test solar-array platens, and the SACS flight components located on both sides of the spacecraft top deck (Figure 2). The SACS top-deck components consist of the following:

- a redundant pump package, total and redundant delta pressure transducers, three latching valves, and the accumulator
- three latching valves that allow for the initial system activation and the strategic wetting of the final two radiators



Top Side of Top Deck (Faces toward TPS)

Underside of Top Deck (Faces toward inside of spacecraft)

#### Figure 2. The PSP SACS top-deck components.

The spacecraft and its components are designed to operate within the wide range of thermal environments defined in Table 1. The SACS budget at perihelion has been reduced from the original design capacity of 6400 W (no fin multilayer insulation [MLI]) to ~5900 W because of the addition of MLI over fins on radiators 1 and 4. The fin MLI was added to reduce low-temperature risk at aphelion; however, it will also decrease the perihelion maximum capacity from  $6400^{1.2}$  to 5900 W (illustrated in Figure 3) by reducing the unallocated margin. The test configuration has ~0.53 m<sup>2</sup> of MLI over the top and bottom fins on radiators 1 and 4 to warm bias the system during the time when only two radiators are active. This configuration represents the first 42 days of the mission when the solar flux is the lowest. From Figure 3, the estimated thermal load from the TPS to the PSP cooling system will be ~533 W, and the thermal load from the TPS and truss structure assembly (TSA) to the spacecraft bus, represented by the spacecraft simulator during this test, will be ~30 W. These quantities were directly measured and will be discussed further in section VII:

- Configuration 1 (C-1): the minimum, with two active radiators, SACS temperature achieved when applying the expected aphelion condition input power (933-W solar-array waste heat)
- C-1: the maximum SACS thermal capacity (SACS at 125°C)
- C-1 and configuration 2 (C-2): the heat leaks into the spacecraft simulator (SACS at 125°C/TPS simulator at 300°C)
- C-2: the TPS simulator heat leak into the SACS (SACS at 125°C/TPS simulator at 300°C)

Orbit Location	Heat Flux
Maximum solar flux at solar encounter (9.86 $R_S$ )	651 kW/m <sup>2</sup>
Minimum solar flux at aphelion (1.02 AU)	$1314 \text{ W/m}^2$
Maximum solar flux during communication slew (0.7 AU)	$2790 \text{ W/m}^2$
Venus albedo	$0.8 \pm 0.02$
Venus infrared emission	$153 \text{ W/m}^2$
Coronal heating at solar encounter $(9.86 R_S)$	$<3 \text{ W/m}^2$

# Table 1. Variation in environmental heat load.

The operation of the SACS will begin before launch when the water in the accumulator is heated to a temperature of ~50°C to overcome any cold spots in the inlet/outlet tubing when the system is initially activated after launch. The accumulator heater will be powered by the launch vehicle umbilical at L-24 hours and only be used prior to L-4 minutes. The temperature will be controlled by three series redundant thermostats ( $T_{low} = 40^{\circ}C/T_{high} = 50^{\circ}C$ ), ensuring that the temperature will not exceed 65°C during pre-launch heating when the pressure is highest. Post launch, this heater will never be used again.

The TPS will be pointed directly at the sun except for communication slews, scientific instrument calibration, and instances when the spacecraft is purposely slewed to point specified radiators toward the sun for warming. Approximately 1 hour after launch, the spacecraft will slew to allow solar warming of radiators 1 and 4 before

wetting. Similarly, radiators 2 and 3 will be warmed by the sun, enabling wetting around 41 days after launch to enable the totality of SACS operation. Between perihelion and 0.7 AU, the flap angle of the solar arrays will be autonomously controlled. As the spacecraft approaches the sun and requires ever-diminishing solar-array surface areas for power generation, the solar arrays will be tucked into the shadow of the TPS until only the secondary solar arrays are exposed to the sun during near perihelion operations. PSP will encounter Venus on several occasions to allow reduced perihelion distances, and before and after each 11-minute-long Venus eclipse, the spacecraft will slew to warm radiators 2 and 3.



Figure 3. Pre-ITVT SACS thermal budget at perihelion (SACS system temperature at 125°C). EOL, end of life.

# **III. SACS Requirements**

The primary requirement of the thermal control system is to provide solar-array cooling from 9.86  $R_s$  to 1.02-AU solar distances. The system will provide this cooling with an operating fluid (water) temperature of +10°C (test minimum) to +150°C (test maximum/platen) depending on heat load and mission timeline. The system is designed for a survival fluid temperature of +200°C. A custom-designed accumulator is used to store the water from launch until initial system activation (~90 minutes after launch) as well as maintain a pressure cap on the system to prevent boiling for temperatures up to 210°C. At 9.86  $R_s$ , the cooling system must be able to accommodate ~5900 W of solar load over two solar-array platen "wings." The maximum platen surface temperature at that heat load cannot exceed 150°C at the platen surface where the solar cells will mount. The highest portion of this heat load will occur on the tips of the solar platen; the heat load is equivalent to 35 suns. A second cooling capacity case occurs at the 0.7-AU hot slew; in that case, the spacecraft is slewed toward the sun and additional heat load is imparted to the solar array. In addition, the total heat load is ~4500 W on the two solar arrays, with an additional 800 W on two of the four radiators.

After a thorough trade study, deionized water was selected as the working fluid for the cooling system. Water provides the best thermal properties for this mission. However, after launch the spacecraft will go through an eclipse, and its temperature will drop below freezing for a period of time. To ensure the system will survive, it will be launched dry with the water stored in the accumulator, which will not be exposed to the harsh temperatures. However, other components in the system will become rather cold, thus driving the survival temperatures of the radiator and cold plate down to  $-130^{\circ}$ C and  $-80^{\circ}$ C, respectively. To ensure the system does not freeze, it will be activated as it is warmed up after the eclipse. The solar arrays will be deployed and heated by the sun at the same time that one or two radiators are heated. Referring to Figure 4, once the components are sufficiently warmed, isolation valve 1 (ISO1) will be opened to flood the pumps, the two solar-array platens, and two of the four cooling system primary radiators (CSPRs) (radiators 1 and 4). With one-half of the system loaded with water, spacecraft

operation will commence, eventually warming and flooding the final two radiators (radiators 2 and 3) ~42 days after launch.



Figure 4. SACS flight system schematic.

The final system driving requirement is a Venus flyby eclipse. The eclipse is expected to last 11 minutes. During this time, the system will have no solar heat load, and fluid temperatures are expected to drop. The system must be able to maintain a minimum water temperature of 10°C in order to ensure that freezing will not occur.

As mentioned previously, the fluid for the cooling system is water. To survive the 7-year mission requirement and the 3-year ground life, the majority of the wetted materials will be titanium. This will reduce the chance of corrosion-induced contamination in the system. All other materials in the system are water and temperature compatible. To ensure the correct positions of the solar cells and the protection provided by the spacecraft TPS, the solar platens will need to be held to tight tolerance for flatness, which must be maintained within 0.006" over any  $10" \times 10"$  area, and 0.012" over the wing width over the temperature range of  $-80^{\circ}$ C to  $200^{\circ}$ C.

# **IV. ITVT Hardware**

The ITVT was the system verification test for the SACS and provided performance data and configuration verification before the observatory system-level vacuum test to follow. The overarching objective of this test was to verify the thermal design of the PSP spacecraft on the ground by exposing the spacecraft cooling system and its interfaces to environments simulating those to be encountered during flight. These operations include the postlaunch maneuvers, activation of the cooling system, Venus eclipse maneuvers, in-transit operations, and perihelion configuration management. As shown in Figure 5, two basic hardware configurations were tested: one with the platens on (C-1) and one with the platens removed (C-2). C-1 represents the most realistic flight hardware arrangement that properly simulates heat input, radiating area, and cooling system resistance (pressure drop). Tests with this configuration will be used to verify critical thermal requirements and characterize system performance as a function of temperature, flow rate, and active radiator status (either two or four panels in operation). C-2 represents the observatory thermal vacuum configuration that allows the solar-array actuators to move the solar-array booms (not possible due to 1-g with platens on). Testing of C-2 in the ITVT enables comparison to C-1 to support the observatory testing. During C-1 and C-2 testing, data will be gathered to allow the development of critical flight software related to solar-array control and safing; this software will be subsequently flight certified during the observatory-level thermal vacuum testing. The hardware modifications allowing the switch from C-1 to C-2 will be accomplished within the Space Environment Simulator (SES) 290 vacuum chamber.



**Figure 5. ITVT Configurations** 

# V. Test-Like-You-Fly Considerations

The ITVT article consisted of both actual flight and non-flight hardware elements. Actual flight hardware evaluated during the ground test consisted of the radiators, the radiator heaters (included for the ground test but not used in flight), the TSA, and the cooling water supply system. Electric heaters were used to emulate various PSP mission environments (see Figure 6). These heaters were energized by a combination of high-power (100 V/50 A) and low-power (60 V/5 A) supplies. Low-power heaters were used to provide energy to the radiator panels, the spacecraft simulator, the solar-array boom heater, the plumbing electronics box, and the water connection lines. A combination of low- and high-power heaters will be used to warm the platens. Additionally, high-power heaters will be used to provide energy to the TPS thermal simulator. The TPS thermal simulator provided back-side temperatures in the range of 300°C to 350°C. The "test-like-you-fly" environmental simulation for the test platens, TPS, SACS radiators, and spacecraft simulator heater plate is described in a bit more detail in the following sections.



Figure 6. SACS test configuration. Radiators 2 and 3 are fully visible. S/A, solar array; S/C, spacecraft.

# A. Test Platens

To properly simulate the high-intensity solar load experienced by the flight solar arrays, the test platens were heated using a combination of ceramic heaters in high-heat-flux areas (the angled platen areas) and Kapton heaters in low-heat-flux regions (Figure 7). For completeness, Figure 8 shows the tested configuration with the back-side MLI installed. For cases following pump down and cooling system initial activation, only the ceramic heaters were used because they have a more reliable heat path and are capable of simulating the high heat flux need to properly simulate solar-array heating. Figure 9 illustrates the environments experienced by the solar arrays over the mission, and Figure 10 illustrates the heat profile applied to each platen simulating the solar heating at closest approach (9.8 R<sub>s</sub>). The test and flight solar-array platens are fabricated from diffusion bonded CP Grade-4 titanium and utilize internal mini-channels to collect the waste heat from the solar cells (or heaters). The mini-channel design is different for the secondary and primary segments of the flight and test platens to minimize pressure drop during maximum flow. The short secondary segment (254 mm in length) uses a small-diameter densely packed mini-channel design that efficiently removes the waste heat when at the highest flux; however, this comes with a pressure-drop penalty. To reduce the pressure-drop penalty, the primary segment (864 mm in length) transitions to a larger-diameter minichannel design, in the region of the plane change, because the heat flux profile is very benign (see Figures 9 and 10) and the pressure drop would be more substantial due to the greater length. For transient considerations, the mass of the test platens matches the as-built mass of the flight solar arrays to within 1%. As reference, the test and flight platens are uniform in width, measuring 635 mm and 686 mm, respectively, and both are 1118 mm in length.



Figure 7. Typical solar-array test platen populated with test heaters. For clarity, the back side is shown with the MLI removed.



Figure 8. A typical test platen with MLI installed. Note that, by design, the inlet and outlet manifolds are uncovered.



Figure 9. Solar-array heat load based on position and solar distance.



Figure 10. Solar-array applied heat load and platen temperature predictions simulating closest approach  $(9.8 R_S)$ .

## **B. TPS Simulator**

Approximately 90" in diameter, the flight TPS points toward the sun during closest solar approach, and shields the spacecraft bus from direct solar impingement. The top (i.e., sun-facing) surface of the TPS is predicted to reach temperatures in excess of 1200°C. To sustain these high temperatures, the TPS is made of lightweight carbon–carbon foam sandwiched between two carbon–carbon facesheets. The top facesheet is coated with a specialized coating, but the bottom facesheet is uncoated. The bottom (i.e., bus-facing) surface of the TPS is predicted to reach temperatures near 300°C, creating an important source of incident infrared heating on the bus. Because the very high incident heat load and temperature of the top surface of the TPS cannot be replicated in available test facilities, the verification of the PSP spacecraft thermal design is performed in stages.

The thermal simulator was designed and tested to meet the thermal design requirements and contamination limits of the bottom TPS surface. The TPS simulator was successfully used during the ITVT and will be used again during final spacecraft thermal vacuum testing to enable testing of the spacecraft in thermal vacuum with the predicted infrared heat load. Figure 11 illustrates the mechanical and thermal structural and packaging design, and Figure 12 shows the thermal predictions and expected power required as a function of temperature that were verified during ITVT.



Figure 11. TPS simulator mechanical and thermal design.



Figure 12. TPS simulator thermal zones and predicted power and bottom temperature at 350°C.

# **C. SACS Radiators**

The SACS radiators (shown in Figure 13) were heated in symmetric pairs using nine circuits per radiator pair. Uneven fin heat loading was applied as prescribed by thermal modeling of the conical solar illumination geometry and was critical for verification of the warm-up scenarios that will be experienced during initial system activation, final system wetting (R23), and the pre-Venus eclipse heating of the SACS. Titanium 3Al-2.5V tubes, 6.35 mm in diameter, are bonded to a 6061 aluminum facesheet and saddle using EA9394 epoxy to create each fin assembly. Nine fin assemblies make up a completed radiator assembly. Each radiator assembly is 1 m<sup>2</sup> and weighs ~5 kg.



Two heater assemblies per fin Wired in parallel :  $R_{T_{in}}=24\Omega$ 

Similar fins 1-9 on radiator 1&4 and 1-9 on radiator 2&3 are then wired in parallel  $R_{T_{circult}} = 12 \Omega$  (24 $\Omega$  for fin #1 on Rad 2&3)

Circuit max power = 300 W (60V/5A) Total of 18 radiator fin circuits





Parallel Wiring Nominal Circuit Nom in al Circuit ocation Resistance (Ω) Resistance (Ω) Radiator 1 Fin 1 to Radiator 4 Fin 1 Radiator 2 Fin 1 to Radiator 3 Fin 1 24 Radiator 1 Fin 2 to Radiator 4 Fin 2 Radiator 1 Fin 3 to Radiator 4 Fin 3 Radiator 2 Fin 2 to Radiator 3 Fin 2 Radiator 2 Fin 3 to Radiator 3 Fin 3 12 12 12 12 Radiator 1 Fin 4 to Radiator 4 Fin 4 12 Radiator 2 Fin 4 to Radiator 3 Fin 4 12 Radiator 1 Fin 5 to Radiator 4 Fin 5 Radiator 2 Fin 5 to Radiator 3 Fin 5 12 12 Radiator 2 Fin 6 to Radiator 3 Fin 6 Radiator 2 Fin 7 to Radiator 3 Fin 7 Radiator 1 Fin 6 to Radiator 4 Fin 6 12 12 Radiator 1 Fin 7 to Radiator 4 Fin 7 12 12 Radiator 1 Fin 8 to Radiator 4 Fin 8 12 Radiator 2 Fin 8 to Radiator 3 Fin 8 12 Radiator 1 Fin 9 to Radiator 4 Fin 9 Radiator 2 Fin 9 to Radiator 3 Fin 9 12



# **D.** Spacecraft Simulator

The spacecraft simulator utilized two plates, with ten Kapton heaters per plate to measure the heat leak into the simulated spacecraft bus from the SACS and TPS simulator. The simulated spacecraft boundary condition of 20°C was held for both test configurations (C-1 or C-2), and the power needed to maintain this temperature setting was measured. As the SACS and TPS simulator warmed, the conducted and radiated heat caused the input power needed by the spacecraft simulator to decrease. Delta changes in control heater power needed to maintain 20°C on the plates provided direct measurements of heat flow into the underlying structure. Figure 14 illustrates the relationship of the simulator to the flight top deck that is populated with SACS components.



Figure 14. The spacecraft simulator plate as seen by the spacecraft top deck.

# E. Test-Like-You-Fly Deviation: Use of the Phase B Accumulator

#### 1. Flight Accumulator Failure

On 26 December 2016, the PSP flight accumulator failed during the second of three (planned) expulsion tests (pre-vibration) at UTC Aerospace Systems. The failure occurred during the fill process. APL was notified immediately, and a Failure Review Board was convened the next day. The root cause was determined to be insufficient bonding to the titanium center tube shell, which caused blisters to form at voids in the Rulon liner that created an interference with the bellows sweeper guide leading edge while stroking, causing the bellows to fail (Figure 15). The sweeper guide with the leading edge intolerant of imperfections in the Rulon bonding also contributed to the failure. The initial program impact was enormous; this was viewed as a catastrophic failure, and the risk to the July 2018 launch was very high. Plans were being made to absorb a slip and begin preparing for the 21 May to 3 June 2019 launch backup. Goddard Space Flight Center (GSFC) was informed and put on standby for the SACS ITVT that was scheduled to start in late February 2017. The SES shroud replacement (April 2017 and lasts 6 months) and PSP observatory thermal vacuum testing (December 2017) were in series with this test. Cost impact to NASA was expected be exceptionally high, and mission risk would increase due to the longer flight time and no viable backup mission. The accumulator was scheduled to be installed into the SACS in mid-January 2017, and the completed SACS was to be delivered to APL (GSFC) the first week of February 2017. This was the last major component requiring integration before subsystem delivery.



# Figure 15. Flight accumulator failure.

#### 2. The Accumulator Function

The flight accumulator stores the fluid required for the mission and is used for initial post-launch system activation (solar arrays + two radiators) and final radiator activation (L+42 days). A fixed gas charge provides system pressure to prevent cavitation at the pump, and the nominal maximum system water temperature is  $135^{\circ}$ C (designed for a maximum temperature of  $210^{\circ}$ C). The accumulator compensates for fluid volume changes due to thermal excursions for the expected operating range of the SACS.

Before launch, water in the accumulator is heated by an external heater to a temperature between  $40^{\circ}$ C and  $50^{\circ}$ C to overcome any cold spots in the radiators and tubing and ensure that the system does not freeze when it is initially wetted. The heater is controlled by three thermostats in series for fault tolerance, and power is supplied by the launch vehicle (Figure 16).

#### 3. Preserving the 2018 Launch Date

Before the failure, there were eight Dynatube connections being used on the SACS:

- Two per each solar array
- Two per each 1/2" latch valve (ISO2 and ISO3)

Adding an additional mechanical field joint was programmatically viewed as low risk and divorced the nearly integrated SACS from the redesign and manufacture of the replacement flight accumulator.

APL and UTC Aerospace Systems worked together to define the orientation of the mating Dynatubes and verified that this installation could be readily done at APL during a predetermined segment of integration and testing (I&T). This decision also allowed for the implementation of the Phase B accumulator (PBA) during ITVT.



Figure 16. The flight accumulator in the SACS.

#### 4. Using the Phase B Accumulator

During Phase B (2012–2013), a development accumulator was built to demonstrate the bellows design, and this accumulator was successfully used during the SACS half system thermal vacuum test (October to November 2013). Functionally, the PBA (Figure 17) provides the same expulsion and pressure functions as the flight unit. However, the PBA *thick-walled aluminum* shell is cylindrical and larger, and the bellows can hold ~20% more fluid (a good thing because there is more ground support equipment volume); PBA can only be stroked vertically; and the PBA cannot be physically substituted for the flight accumulator because of its size and form factor. The SACS and PBA arrived at GSFC on 21 February 2017 (~7 days later than was scheduled before the failure).

During the test, the PBA was positioned external to the SACS and temperature controlled to the levels expected during the different conditions set forth by the test cases. Thermal vacuum testing started on 2 March and concluded on 15 March 2017. The SACS was removed from SES 290 on 20 March and arrived at APL on 28 March (on schedule). The replacement flight accumulator was eventually completed and was integrated with the SACS using

the Dynatube interface on 14 June 2017. Since its integration, several expulsions and re-fills have been successfully completed during I&T. The accumulator was present for the system-level observatory mechanical (mass properties, vibration, and acoustic) and thermal vacuum testing, where its performance was verified.



Figure 17. The PBA installed for ITVT. GSE, ground support equipment.

# VI. Test Profiles

The ITVT simulated the flight environment of the spacecraft from the coldest point of the Venus eclipse, as well as aphelion conditions before activation of the full cooling system, to the hottest perihelion approach in the sun's corona. Testing verified the thermal performance of the SACS during steady-state and critical transient operational scenarios. The testing also verified the heat flow from the TPS into the SACS radiators and into the spacecraft top deck.

During ITVT, the thermal performance of the SACS was evaluated using test configuration 1 (C-1) (test platens installed) and test configuration 2 (C-2) (test platens removed), which were designed to bracket the entire mission phases expected during flight, including the critical transients.

Critical mission cases were simulated by applying the calculated heat load to the radiators in order to achieve the desired water temperature. The critical mission cases are as follows:

- A1: Cold case at 10°C with two active radiators and no heat into active radiators/platen heating olny
- B2: Cold case super aphelion slew at ~20°C with four active radiators and heat added to two of the four active radiators and the two platens

- B5: Hot case 0.7-AU communication slew at ~95°C with four active radiators and heat added to two of the four radiators and the two platens
- B6: Hot case at perihelion (9.8 R<sub>s</sub>) with four active radiators and simulating the maximum SACS water temperature at the maximum system heat input through the platens (no radiator heating)

The C-1 configuration was used to verify critical thermal requirements and characterize system performance as a function of temperature and water flow rate for both the two and four active radiator configurations with the equivalent of two radiator fins per radiator blanketed on radiators 1 and 4. This configuration will also be used to demonstrate solar-array-related autonomy responses to heat flux and measure temperatures.

The C-2 configuration represents the configuration that will be used during observatory testing to allow test cases with solar-array gimbal motion, including solar-array flap-angle control, solar-array feathering at aphelion conditions, solar-array motion during the post-separation sequence, and solar-array safing. The data gathered during these tests will be used to help set future SACS parameters and verify proper SACS operation for operational scenarios without the platens. All SACS thermal control (heating) was done using the four radiators in a symmetric manor.

The platens or radiator heaters were used to provide water at prescribed temperature levels at the exit of the pump for configurations C-1 and C-2, respectively.

#### A. Definition of SACS Water Temperature

The SACS water temperature was measured at the pump check valve outlet, and this will represent the SACS temperature (see Figure 18 for reference). During the test, the water temperature was controlled within  $\pm 2^{\circ}$ C of the desired target temperature.



**Test Configuration-1** 

Test Configuration-2

Figure 18. The "as-run" test profiles for the two configurations.

## **B.** Configuration 1 Summary

The C-1 profiles shown in Figure 18 are labeled A, B, and C. Profile A corresponds to spacecraft simulations with two active radiators (radiators 1 and 4), and profiles B and C represent spacecraft configurations with four active radiators. Profile C repeats selected SACS temperature set points from profile B but with the TPS simulator in operation. Before profiles A and B, spacecraft transients will be simulated.

Before profile A, the initial dip (on the left part of Figure 18) corresponds to the transition to vacuum experienced during launch, and the flat portion of the profile before the first ramp corresponds to the mission timeline where the post-launch wetting of radiators 1 and 4 will be simulated. Before profile B, the wetting of the remaining two radiators (radiators 2 and 3 on day 41 of the mission) will be evaluated. The Venus eclipse and preand post-Venus eclipse maneuvers will be evaluated in profile B.

Each of the initial states A1–A4 and B1–B4 corresponds to a water temperature downstream of pump 1 bracketing the expected temperature during flight, which is set by varying the power level of the platens. The profile C tests in Figure 18 are executed with the TPS simulator on, in part, to evaluate energy flow from the TPS into the SACS and spacecraft simulator. Changes in spacecraft simulator power were measured and quantified as SACS inputs (at temperatures >20°C during profile B) and combined SACS and TPS (during profile C). The requirement is that the combined thermal load into the spacecraft must be <40 W (30 SACS/10 TPS).

Critical mission transients executed during C-1 are defined as follows:

- Post-launch cooldown and initial SACS wetting
- Final system activation: wetting of radiators 2 and 3
- Venus eclipse

Figure 19 summarizes graphically the implementation of each of the described critical transient cases tested.



Figure 19. Critical transients during C-1.

# C. C-2 Summary

The C-2 configuration profiles shown in Figure 18 are labeled B, C, and D and represent temperature conditions expected during observatory-level thermal vacuum testing. Testing is simplified by eliminating the need for a flight-like fill after the C-1 configuration testing by removing profile A. Unlike C-1, the SACS was fully wetted at the start of C-2 before the initiation of the testing depicted in Figure 18. Profiles B and C are essentially repeats of those without the platens. Any testing accomplished in profile D will be to-be-determined tests that do not require the use of the platens. During profile C, CSPR heater power levels will be adjusted down to attain the same initial temperatures from profile B (e.g., C5 = B5 and C6 = B6), and the differences in the required power setting to reach these temperatures (comparisons between profiles C and B) will provide a direct assessment of the heat introduced into the SACS by the TPS simulator, as illustrated in Figure 3 (SACS thermal margin at perihelion). The spacecraft simulator was operated in the same manner as was done during C-1.

Table 2 summarizes the implementation of each steady-state case tested during C-2.

Configuration	<b>CSPRs</b> Active	Case Number	Description
		B1	CSPRs heated to achieve 10°C water
		B3	CSPRs heated to achieve 37°C water
		B4	CSPRs heated to achieve 75°C water
		B5	CSPRs heated to achieve 95°C water
C-2	All	B6	CSPRs heated to achieve 125°C water
			TPS simulator heated to 300°C and CSPRs heated to achieve
		C5	95°C water
			TPS simulator heated to 300°C and CSPRs heated to achieve
		C6	125°C water

Table 2. C-2 case list.

#### VII. Test Results Summary

Summary results for the Venus eclipse transient, C-1 and C-2, are represented by Figures 20–26. Overall the testing was very successful and provided valuable data that verified the SACS Level 4 requirements and were used to better refine the SACS and spacecraft thermal models. For the majority of the test, the results were as expected, except for the spacecraft simulator power measurement over the operational temperature range for the SACS. Spacecraft simulator power verified that particularly with C-1/profile B during SACS hot operation (+125°C), the heat load into the simulator plate was on the order of 100 W (three times greater than the requirement). This anomaly will be discussed further in the following section.

#### A. Venus Eclipse

During C-1 and before the start of the steady-state cases run during profile B, the last two critical transients (the activation of radiators 2 and 3 and the Venus eclipse) were performed. The Venus eclipse was important to verify that during the 11-minute eclipse, the unheated SACS would remain above  $15^{\circ}$ C. To mimic flight operations, 1 hour before the start of eclipse, the test heaters on radiators 2 and 3 were powered to simulate the spacecraft slew to  $45^{\circ}$ , increasing the SACS heat load to drive the system water temperature to ~70°C. At the end of 1 hour, all SACS heaters were powered off for 12 minutes to represent the eclipse. As shown in Figure 20, the minimum water temperature realized during the eclipse was  $38^{\circ}$ C, well above the required minimum  $15^{\circ}$ C. As shown, the slew heating is necessary because the temperature drop during the eclipse was ~40°C; before the slew, the SACS water temperature was ~53°C.



Figure 20. Venus eclipse transient simulation. S/C, spacecraft.

#### **B.** Configuration 1/Profile A

C-1/profile A represents the first 42 days of the mission when only the solar-array platens and radiators 1 and 4 are active and the spacecraft is farthest from the sun. Understanding the SACS temperature as a function of input power is essential to verify the minimum heat load to keep the SACS above ~15°C when in this configuration. During spaceflight, radiators 1 and 4 cannot be slewed (other than during initial activation) toward the sun, so the

only system heating is a result of the waste heat from the solar arrays, estimated to be 933 W. As a result of this constraint, ~0.53 m<sup>2</sup> of MLI was added to the top and bottom of the fins on radiators 1 and 4 to reduce the effective radiator area. From the results of the test it is shown, in Figure 21, that for 934 W (case A2) of thermal input, the SACS water temperature was measured to be 14°C. Because the desired water temperature is >15°C with 933 W of thermal input, based on the correlated thermal model, the MLI coverage will be increased to 0.62 m<sup>2</sup> for observatory thermal vacuum testing. Further "area tailoring" will be applied as needed based on results achieved during the observatory thermal vacuum test.



Figure 21. C-1/profile A summary. The fin MLI and back-side MLI coverage of radiators 2 and 3 is also shown. S/C, spacecraft.

# C. Configuration 1/Profile B

As C-1/profile A represents the first 42 days of the mission when only the solar-array platens and radiators 1 and 4 are active, C-1/profile B represents the fully activated system that is the long-term mission configuration. Unlike the heating constraints with only two active radiators, radiators 2 and 3 can be slewed into the sun to augment the solar-array waste heat and keep the SACS comfortably above 20°C, as shown in Figure 22, case B2, so cold-case concerns related to a fully active system are somewhat mitigated. However, the driving hot case is represented by the perihelion environments where the TPS, solar arrays, and TSA mounted instruments are at maximum heat flux and temperature; the predicted SACS capacity is shown in Figure 3, with the SACS temperature at 125°C. As shown in Figure 22, the measured heat capacity of the SACS when at 125°C and with 0.53 m<sup>2</sup> of fin MLI was 6153 W, or ~253 W better than expected when compared to the pre-ITVT predictions illustrated by Figure 3 (5900 W).



Figure 22. C-1/profile B summary. S/C, spacecraft.

# **D.** Configuration 2/Profile B and Profile C

The C-2 test configuration will be used during the observatory-level thermal vacuum testing to allow for autonomy and fault-management testing with the SACS operational (and with the platens removed to allow for unconstrained articulation of the solar-array booms). Thermally, during C-2 an accurate measurement of the heat load from the TPS into the SACS could be realized because the platens would no longer be in the field of view of the TPS simulator (as they were in C-1). Referring back to Figure 9, at 9.86 R<sub>s</sub> the platens are nearly vertical and are not influenced by the TPS. During case B6 (shown in Figure 23), the heat needed to reach 125°C SACS temperature was 4612 W. From case C6 (shown in Figure 24), with the TPS at 300°C, the heat needed to reach 125°C SACS temperature was 4037 W. Therefore, subtracting these SACS power measurements yields 575 W of TPS heat into the SACS, compared to the 533 W that was predicted and is illustrated by Figure 3.



Figure 23. C-2/profile B summary. S/C, spacecraft.



Figure 24. C-2/profile C summary. S/C, spacecraft.

# E. Test Anomaly: SACS Excessive Heat Leak into the Spacecraft Simulator

During C-1/profile B, it was very apparent that the heat from the SACS into the spacecraft bus with the SACS at 125°C was much higher than expected. Upon analysis of the test data, the effective leak into the spacecraft simulator was measured to be grossly 100 W, approximately three times greater than that expected from the pretest thermal modeling, and is illustrated by Figure 25. As shown in Figure 3, the fluid-flowing side of the SACS resides on the top side of the top deck and is covered with low-emissivity tape. It was determined that the majority of the unexpected heat leak was due to much higher thermal conductivity in the component mounting hardware, which, in turn, caused the underside of the top deck to become considerably warmer and radiate three times the expected heat into the spacecraft simulator. The design fix, which is being verified during observatory-level thermal vacuum testing, was to add an MLI blanket between the underside SACS components (keeping them radiation coupled to the internal spacecraft) and the underside of the top deck. Black Kapton tape was also added to the underside components to increase the surface emissivity and help with thermally coupling these components to the spacecraft bulk temperature. Thermal modeling indicates that this additional MLI coverage will considerably reduce underside area x emissivity of the top deck and should get the heat leak back into the range of 30 W. Figure 26 shows the fix as it was implemented before integration with the spacecraft. The remaining underside MLI was installed after SACS integration (the underside of the top deck is now completely covered). One important note, two Remote Interface Units (RIU) are located on the top side of the top deck in the old design were controlled by the deck interface. Isolating the top deck protects the majority of the spacecraft components from SACS heating but also isolates the RIU's from the rest of the spacecraft resulting in high RIU temperatures. The lowest risk and most deterministic solution is to thermally isolate the RIU's from the top deck and create a thermal short using a commercial copper heat strap to a cooler zone of the spacecraft and will be verified at the Observatory Level TVac test.

- B1:
  - > SACS Temp 10°C
  - > S/C Sim Power 133 W
- B6:
  - > SACS Temp 124°C
  - > S/C Sim Power 34W



Figure 25. SACS heat leak into the spacecraft simulator is quantified to be ~100 W. S/C, spacecraft.



Top Side of Top Deck final configuration (Faces toward TPS)

Underside of Top Deck with partial MLI installed (Faces toward inside of spacecraft)



# VIII. Model Correlation

A thermal model was developed that represented both of the ITVT test configurations. The model, C1 shown in Figure 27, has the appropriate 0.525 m<sup>2</sup> of mli coverage on radiators 1&4 and the platens in the fixed horizontal orientation for test configuration 1. Initial thermal model runs showed marginal agreement, but over-predicted water temperatures for the test power input over all cases: Configuration 1: 4 °C; Configuration 2: 8 °C. Adjustments were made to the thermal model to better represent the test parameters:

- Increased platen thermal nodes from 1 to 22, with water node under each
- Input power to the radiator fins adjusted to account for fin efficiency effects of adding heat to the edges of the radiator fins
  - Verified with detailed fin model, indicating loss of efficiency due to non-uniform temperature across the fin

The adjusted ITVT thermal model shows good agreement ( < 1  $^{\circ}$ C) as compared to the test data over all power levels, for 2 and 4 radiators and TPS off (~1  $^{\circ}$ C) and TPS on (~3  $^{\circ}$ C). These results are illustrated in Figure 28.

Results of the model also indicated that to meet 18 °C with two active radiators and 933 W of solar array waste heat that the mli coverage would need to be increased to  $0.62 \text{ m}^2$  for the observatory level TVAC test.



Figure 27. The thermal model representing the SACS in the C1 configuration.



Figure 28. Model results are in good agreement with data taken for both test configurations .

# IX. Conclusion

The very successful ITVT provided verification that the SACS subsystem performed as expected over a broad range of active radiators, temperatures, input power, and transient responses, both thermal and electrical. The test helped to close a majority of the Level 4 requirements that were based on hot and cold system performance with two and four active radiators, hot and cold expected heat loads, and the aggregate system time constant. It also provided an end-to-end functional test that allowed all of the flight hardware to be integrated and operated together over a wide range of temperatures, radiator configurations, and pump speeds. Data gathered for the two different test configurations provided the flight system with measured parameters regarding system delta-P and pump and

electronics power usage as a function of temperature and pump speed, flight temperature sensor verification, and transient timing parameters associated with the initial system warm-up and activation.

The design flaw regarding anomalous top-deck heat leak was unambiguously exposed and quantified, and a simple fix was implemented before SACS integration. Without performing an "integrated thermal test" where the spacecraft was thermally represented by the simulator plate, this problem would not have been found until the observatory-level thermal vacuum test, and implementing the fix would have been an enormous undertaking from a technical (the fix would have needed to be made during the chamber break so that the fix was tested in C-2, probably requiring the spacecraft being removed from the SES) and schedule perspective.

As a comparison to ITVT, Figure 29 shows the integrated PSP spacecraft with the flight solar arrays installed (left) along with the solar-array booms being articulated during flight system software testing during C-2. Also evident is the MLI covering the fins on radiators 1 and 4.



Figure 29. The PSP spacecraft before observatory thermal vacuum testing. C-1 is on the left, and C-2 is on the right.

Finally, Figure 30 shows the SACS assembly being removed from the SES on 20 March 2017 after ~3 weeks in the SES 290 thermal vacuum chamber. The authors would like to thank all of the various support individuals at GSFC who helped to make this test smooth and very successful. The authors would also like to thank Neal Bachtell, Tony Ahan, Andy Webb, Tony Scarpati, Rick White, Rick Campbell and Carl Clayton from APL and Gary Stewart, Christopher Miller, Eric Bechard, Jonathan O'Neill, Tom Davis and Kevin Aceves from United Technologies for test setup and MLI support and SACS operations.



With the TPS simulator already removed the SACS assembly is being prepared for lifting

SACS assembly is being lifted out of the SES-290

Figure 30. The SACS assembly is lifted out of the SES 290 on 20 March 2017.

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