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Cryogenic Propellant Storage and Transfer Technology Demonstration: Prephase A Government Point-of-Departure Concept Study

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September 2012

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LIST OF ACRONYMS, SYMBOLS, AND DESIGNATORS

ACS	altitude control system
AFC	automated fluid coupling
APIC	advanced programmable interrupt controller
ARC	Ames Research Center
BAA	Broad Area Announcement
BAC	broad-area cooling
CAD	computer automated design
CCAFS	Cape Canaveral Air Force Station
C&DH	communications and data handling
CDR	Critical Design Review
CFM	cryogenic fluid management
CFMS	cryogenic fluid management system
CFS	cryogenic fluid system
COPV	composite overwrap pressurant vessel
CPS	cryogenic propulsion stage
CPST	cryogenic propellant storage and transfer
CV	check valve
CY	calendar year
DAS	Debris Assessment Software
DDT&E	design, development, test, and engineering
ECDU	experiment computer/data unit
ESEOD	equivalent sharp edge orifice diameter
FEM	finite element model
GN_2	gaseous nitrogen

LIST OF ACRONYMS, SYMBOLS, AND DESIGNATORS (Continued)

GN&C	guidance, navigation, and control
GR&A	ground rules and assumptions
GRC	Glenn Research Center
GSFC	Goddard Space Flight Center
HEFT	Human Exploration Framework Team
HEX	heat exchanger
JSC	Johnson Space Center
J-T	Joule-Thomson
KSC	Kennedy Space Center
LAD	liquid acquisition device
LEO	low-Earth orbit
LH_2	liquid hydrogen
LO_2	liquid oxygen
LTIME	Life TIME (an orbit lifetime tool)
MCR	mission concept review
MDP	maximum design pressure
MEL	master equipment list
MEOP	maximum expected operating pressure
MGA	mass growth allowance
MLI	multilayer insulation
MOC	Mission Operations Center
MSFC	Marshall Space Flight Center
NEA	near Earth asteroid
NGO	needs, goals, and objectives
OCT	Office of the Chief Technologist
OD	outer diameter

LIST OF ACRONYMS, SYMBOLS, AND DESIGNATORS (Continued)

PCSA	passive cryogenic storage analyzer
PDM	power distribution module
PDR	preliminary design review
POD	point of departure
QD	quick disconnect
QPSK	quadrature phase shift keying
RBO	reduced boiloff
RCS	reaction control system
RFMG	radio frequency mass gauging
RV	relief valve
RW	reaction wheel
SOFI	spray-on foam insulation
STK	Satellite Tool Kit
TDRS	tracking and data relay satellite
TDRSS	tracking and data relay satellite system
TOC	Technology Operations Center
TRL	Technology Readiness Level
TVS	thermodynamic vent system
WBS	Work Breakdown Structure
WFF	Wallops Flight Facility
V	vent valve
VME	Versa Module Eurocard (computer bus standard)
ZBO	zero boiloff

NOMENCLATURE

A	area (m^2)
I_{sp}	specific impulse (s)
T	temperature
V	velocity
V_∞	hyperbolic excess speed
ΔV	delta velocity

TECHNICAL MEMORANDUM

CRYOGENIC PROPELLANT STORAGE AND TRANSFER TECHNOLOGY DEMONSTRATION: PREPHASE A GOVERNMENT POINT-OF-DEPARTURE CONCEPT STUDY

1. STUDY OVERVIEW

1.1 Study Purpose

The primary purpose of this study was to define a point-of-departure (POD) prephase A mission concept for the cryogenic propellant storage and transfer (CPST) technology demonstration mission to be conducted by the NASA Office of the Chief Technologist (OCT). The mission concept includes identification of the cryogenic propellant management technologies to be demonstrated, definition of a representative mission timeline, and definition of a viable flight system design concept. The resulting mission concept will serve as a POD for evaluating alternative mission concepts and synthesizing the results of industry-defined mission concepts developed under the OCT contracted studies. This mission concept provides a balanced approach between meeting mission flight test objectives and project funding constraints. The study objectives are summarized as follows:

- Develop a viable mission concept that provides a balanced approach to meeting mission objectives within project constraints.

- Satisfy CPST mission needs, goals, and objectives.

- Provide a framework for evaluating alternative CPST mission concepts.

- Provide a framework for technology development and ground test planning.

- Provide a foundation for a synthesized reference concept based on the OCT contracted studies timeline.

1.2 Study Timeline

The 6-mo Government POD study was conducted from March to October 2011, building on the results of several other prephase studies conducted in 2010 as part of the CPST project formulation. The mission concept developed in this study was based on the same study guidelines as the OCT contracted studies to ensure the results of this study would be of the same content and

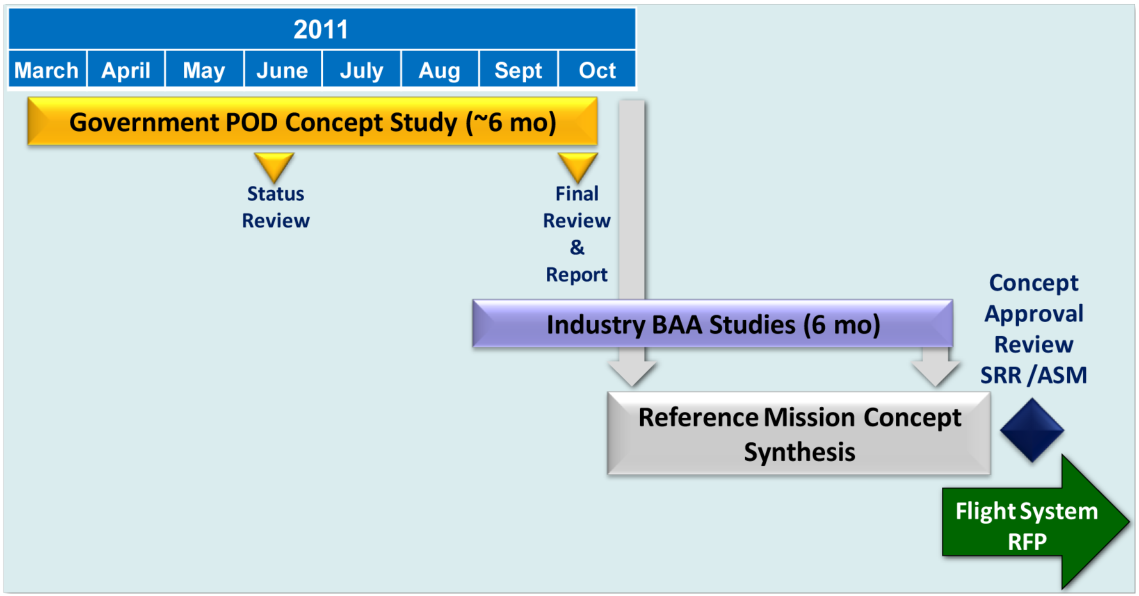


Figure 1. Study timeline.

format as the industry studies. Figure 1 shows the relationship of the Government POD study and the industry Broad Area Announcement (BAA) contracted studies. The results of the Government POD study will be synthesized with the results of the contracted studies to define a reference mission concept that will serve as the basis for defining the actual flight system in accordance to the CPST project acquisition process.

1.3 Study Topics

The study addressed eight study topics which are listed in table 1. Responsibilities for the study topics were spread across four teams from five NASA Centers, totaling over 60 study participants. The responsibilities of each team are shown in figure 2. The first three study topics addressed the basis for the technology demonstration mission. The CPST project management and technology teams provided key inputs to describe the need and justification for the proposed on-orbit technology demonstration. The technology team identified the demonstration technologies and developed the technology maturation plan. The mission concept, including mission orbit analysis and mission timeline, was defined for the study design team. The fourth study topic included the technical design analysis of the demonstration flight system, including the definition of technology components of the cryogenic fluid system (CFS) and the supporting spacecraft subsystem bus. The final four topics addressed programmatic considerations for the POD mission concept. The preliminary mission cost estimate included estimates of the design development and testing of the flight system and mission launch, and operations costs. The schedule assessment included development, testing, and integration of the technology components and flight system. Study topic seven identified potential partnerships and partnership strategies with other government agencies and industry. Study topic eight included identification of top-level project risks and an initial assessment of risk mitigation strategies.

Table 1. Study topics.

Study Topic/Product Area	Responsible Team
(1) Mission justification	Project analysis
(2) Technology maturation	Technology analysis
(3) Mission concept	Design analysis
(4) Demonstration flight system	Design analysis
(5) Mission cost estimate	Project analysis
(6) Project schedule	Project analysis
(7) Government and industry partnerships	Project analysis
(8) Project risk identification	Safety and mission assurance

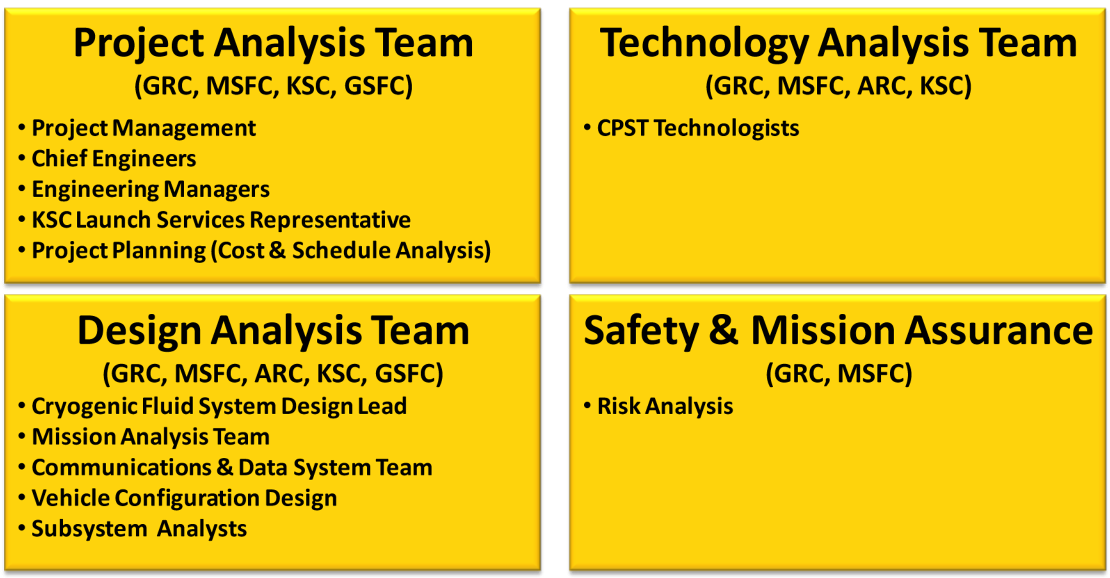


Figure 2. Study teams.

1.4 Needs, Goals, and Objectives

The POD mission concept defined in this study is based on the CPST project needs and goals listed in sections 1.4.1 and 1.4.2.

1.4.1 Project Needs

There is a need for a cost-effective flight test in order to raise the Technology Readiness Levels (TRLs) of cryogenic fluid management (CFM) technologies by demonstrating on-orbit capability and processes used for storing and transferring cryogenic fluids (e.g., liquid oxygen (LO_2) and liquid hydrogen (LH_2) in a microgravity environment.

1.4.2 Project Goals

Specifically, these technologies will help NASA satisfy its strategic goals and objectives in space exploration and scientific discovery by increasing the performance of current and future spacecraft systems. Technology maturity will be raised to a minimum TRL of 6 and a goal of TRL 7. The CPST project has defined two primary goals:

(1) Demonstrate long-duration, in-space storage of cryogenic propellants.
(2) Demonstrate in-space transfer of cryogenic propellants.

To achieve the CPST project goals, sets of flight test objectives were defined. The prioritization system of these objectives is based on the perceived values of future technology system stakeholders. Anticipated stakeholders include the most likely applications for the CPST technology such as in-space cryogenic propulsion stages (CPSs) and in-space cryogenic propellant depots.

Table 2 is a listing of prioritized flight test objectives.

Table 2. Primary flight objectives.

Primary Flight Test Objectives		
P1	LH_2–Cryogenic fluid storage	Minimize storage boiloff LH_2 in microgravity
P2	LH_2–Cryogenic fluid acquisition	Demonstrate acquisition and bubble-free flow of LH_2 in microgravity
P3	LH_2–Cryogenic fluid quantity gauging	Demonstrate mass gauging of LH_2 in microgravity
P4	LH_2–Cryogenic fluid transfer	Demonstrate transfer of LH_2 in microgravity (settled and unsettled conditons)
P5	LO_2–Cryogenic fluid storage	Demonstrate zero boiloff storage of LO_2 in microgravity
P6	LO_2–Cryogenic fluid acquisition	Demonstrate acquisition and bubble-free flow of LO_2 in microgravity
P7	LO_2–Cryogenic fluid quantity gauging	Demonstrate mass gauging of LO_2 in microgravity
P8	LO_2–Cryogenic fluid transfer	Demonstrate transfer of LO_2 in microgravity (settled and unsettled conditions)
Secondary Flight Test Objectives		
S1	Instrumentation–leak detection	Demonstrate leak detection capability for LO_2 and LH_2 in microgravity
S2	Instrumentation–flow measurement	Demonstrate flow measurement of LO_2 and LH_2 in microgavity
S3	Tank pressurization methods	Demonstrate tank pressurization and pressure control approached for LH_2 and LO_2 in microgravity

1.5 Study Results

The study approach followed an iterative process in which the study topics were divided into subtasks to evaluate mission options and perform design trade studies. Many design iterations were required due to changes in mission ground rules and constraints during the course of the study. Since many of the programmatic constraints were not defined at the beginning of the study, the initial approach was to define an upper and lower cost bound for the mission concept. The study task flow is shown in figure 3.

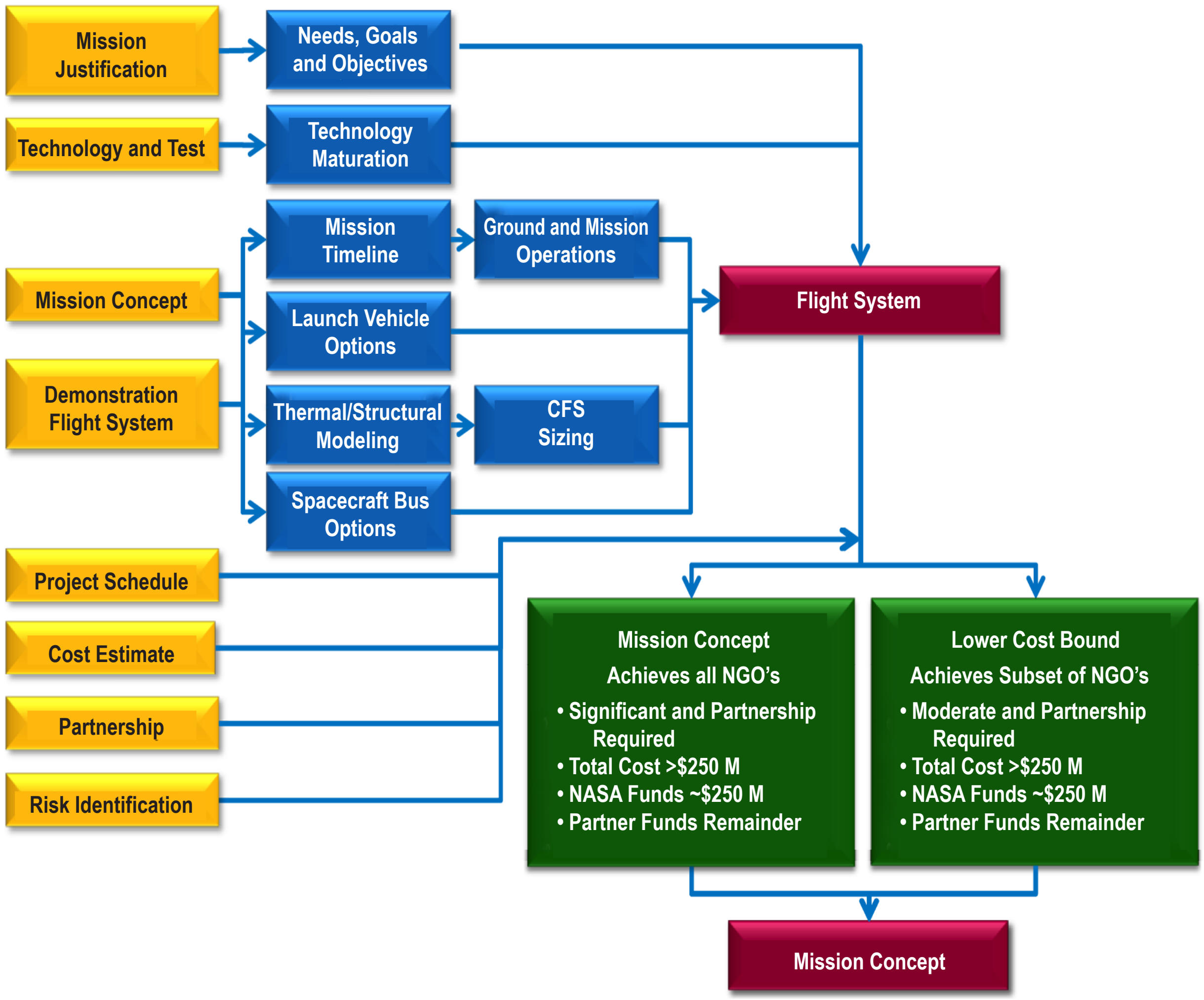


Figure 3. Study task flow.

The design iterations also included subsystem design trades and consideration of mission descope options to reduce the development and operational costs of the technology demonstrations. The options were prioritized to minimize the impact on the established flight test objectives. The evaluation of descope options considered four factors:

(1) Potential reduction in project cost.
(2) Potential reduction in project schedule.
(3) Potential reduction in technology infusion to future programs.
(4) Potential reduction in probability of project success.

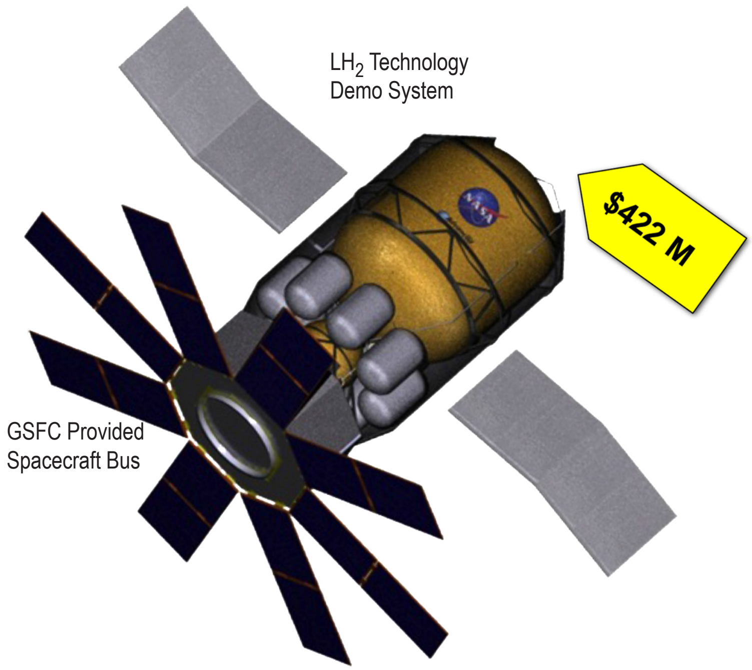
The descope options are summarized in table 3. The descope options that were incorporated into the final POD mission concept are shown in blue text in the bottom rows in the table. The most significant descope that was implemented was the decision to scale back the demonstration to LH_2 only. This eliminated the four primary flight test objectives associated with LO_2. It was determined that by reducing the demonstration to a single fluid, the overall value of the demonstration would be maintained since the LH_2 demonstrations represent the more challenging technologies.

Table 3. CPST mission concept descope options (prioritization based on minimizing technical on flight test objectives).

Descope Factor	Affected Flight Test Objectives NGO Flight Test Objectives Affected by Descope	Project Cost Reduction Potential Reduction of DDT&E, Integration and Operations Cost	Project Life Cycle Schedule Potential Reduction of DDT&E, Integration and Operations	Technology Infusion Potential Reduction of Technology Infusion into Future Missions	Programmatic Risk Potential Reduction of Project Success Probability	Comments
		>$20 M (Green)	>6 mo (Green)	High (Green)	Low (Green)	
		$4-20 M (Yellow)	2-5 mo (Yellow)	Medium (Yellow)	Medium (Yellow)	
Ranking Critieria		<$4M (Red)	<1 mo (Red)	Low (Red)	High (Red)	
Reduce pre-CDR ground test	None	Yellow	Red	Green	Red	Increases technology development risk
Reduce post-CDR ground test	None	Red	Yellow	Green	Red	Increases flight system risk
Eliminate active cooling	P 1, 5	Green	Yellow	Red	Red	
Demonstrate LO_2 only	P 1–4	Green	Yellow	Red	Green	
Reduce mission duration	P 1, 5	Green	Yellow	Red	Yellow	Reduce from 6 mo to 3 mo
Eliminate propellant transfers	P 4, 8; S 2, 3	Green	Yellow	Red	Green	
Reduce pressure-fed settled transfers	P 4, 8; S 2, 3	Yellow	Red	Yellow	Yellow	
Reduce pressure-fed unsettled transfers	P 4, 8; S 2, 3	Yellow	Red	Yellow	Yellow	
Reduce pump-fed settled transfers	P 4, 8; S 2, 3	Yellow	Red	Red	Yellow	
Reduce pump-fed unsettled transfers	P 4, 8; S 2, 3	Yellow	Red	Red	Yellow	
Eliminate low-g automated leak detection	S 1	Red	Red	Green	Red	Leak detection, not leak rate
Reduce mass gauging experiment	P 1-8	Red	Red	Red	Yellow	Linked to all flight test objectives
Reduce video requirements	P2, 4, 6, 8	Red	Red	Green	Green	Use nonvideo flow-bubble sensor
Reduce number of transfer cycles	P 4, 8; S 2, 3	Red	Red	Yellow	Green	Current POD has two transfer demos
Demonstrate LH_2 only*	P 5–8	Green	Yellow	Yellow	Yellow	LH_2 is the preferred single fluid demo
Eliminate automated fluid coupling*	P 4, 8	Yellow	Red	Green	Green	Simplifies system
Eliminate CFS redundancy*	None	Yellow	Yellow	Green	Yellow	Increases flight system risk
Eliminate autogenous tank pressurization*	S 3	Yellow	Yellow	Yellow	Green	Reduced number of components

*Implemented on CPS-Pathfinder POD concept.

Figure 4 shows the final POD mission concept. The flight system consists of an LH_2 technology demonstration system and a spacecraft subsystem bus. The technology demonstration system consists of the CFM payload and propellant. The spacecraft bus consists of all the subsystem components necessary to control and operate the spacecraft for a 6-mo, on-orbit demonstration mission. The design of the spacecraft bus was provided by the Goddard Space Flight Center (GSSF) team. The CPST spacecraft was sized so that it could be launched on a Taurus II launch vehicle departing from the NASA Wallops Flight Facility (WFF). The preliminary mission cost estimate, which includes development, integration, launch, and mission operations for the demonstration mission, was $422 million.



Mission Duration: 6 Months

Spacecraft Size

Length: 4.4 m

Diameter: 2 m

Element	Mass (kg)
CFM payload mass	1,207
CFM propellant (LH_2)	260
Spacecraft bus	831
Launch mass	**2,298**

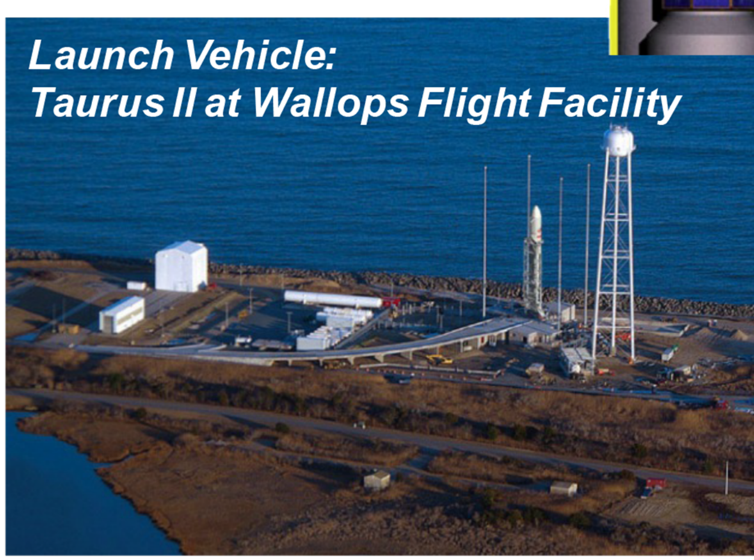


Figure 4. CPST Government POD mission concept.

2. STUDY TOPIC REVIEW

2.1 Mission Justification

2.1.1 Mission Justification and Rationale

During the course of this study, team members assessed CFM technology gaps in light of the full range of mission architecture scenarios. In the early phases of the study, these assessments were based on expert knowledge of previously envisioned mission architectures available in the open literature. As the study progressed, the assessments expanded to include all available versions of the OCT technology roadmaps as well as presentations generated by the Human Exploration Framework Team (HEFT) and its successor, the Human Architecture Team. As shown in table 4, cryogenic storage technologies are enabling for at least seven and as many as ten of the OCT roadmaps while expulsion and transfer technologies are enabling for at least four and as many as nine roadmaps. Also, table 5 highlights the mapping of these cryogenic technologies to the 'top 10' lists in the OCT roadmaps with direct or significant relevance to nine challenges in three different roadmaps. Assessment of the range of HEFT architectures indicated that long-duration cryogenic storage (i.e., for storage durations beyond current state-of-the-art, about 9 hr) is a key enabler for missions beyond low-Earth orbit (LEO), as illustrated in table 6.

Table 4. Mission relevance to OCT roadmaps.

		OCT Roadmaps That Benefit From CRYOSTAT Technologies*									
		TA01	TA02	TA03	TA06	TA07	TA08	TA11	TA12	TA13	TA14
CRYOSTAT Technologies		Launch Propulsion Systems	In-Space Propulsion Technologies	Space Power and Energy Systems	Human Health, Life Support, and Habitation Systems	Human Exploration Destination Systems	Science Instrumentation, Observation, and Sensor Systems	Modeling, Simulation, IT and Processing	Materials, Structures, Mechanical Systems, and Manufacturing	Ground and Launch Systems Processing	Thermal Management Systems
Cryo Strorage Technologies	Active cooling	Probably required	Required	N/A	Probably required	Required	Probably required	May be required	May be required	Required	Required
	Passive cooling	Required	Required	Required	Probably required	Required	Probably required	May be required	Required	Required	Required
	Tank gauging	Required	Required	Required	Probably required	Required	N/A	May be required	N/A	Required	Required
Cryogenic Expulsion and Transfer Technologies	Liquid acquisition	Required	Required	Required	N/A	Required	Probably required	May be required	May be required	N/A	N/A
	Feeding conditioning	Required	Required	N/A	Probably required	Required	N/A	May be required	Probably required	May be required	N/A
	Auto couplings	N/A	Required	N/A	Probably required	Required	N/A	N/A	May be required	Probably required	N/A

* Based on paragraph-by-paragraph assessment of OCT roadmaps. This table is a summary of detailed evaluation.

Table 5. Mission relevance to top challenges in OCT roadmaps.

TA02 (In-Space Propulsion)	TA14 (Thermal Management Systems)	TA01 (Launch Propulsion)
#2 Long-term, in-space cryogenic propellant storage and transfer*	#2 Innovative thermal components and loop architectures**	#4 Nontoxic RCS**
#4 Advance in-space cryogenic engines and supporting components**	#3 20K cryocoolers and propellant tank integration*	#5 Advanced main propulsion system components**
#7 Nuclear thermal components and systems**	#4 Low-conductivity structures and supports*	
#8 Advanced space-storable propellants**		

* CPST demo directly addresses challenge.

** CPST demo addresses significant part of challenge.

Table 6. Mission relevance to human architecture destinations.

Technology Applicability for Future Destinations*		Destination									
		LEO		Beyond LEO	Moon		NEA		Mars		
		LEO	Advanced LEO	Cis-Lunar	Lunar Surface Sortie	Lunar Surface Outpost	Minimum NEA	Full Near Earth Asteroid	Mars Orbit	Mars Moons	Mars Surface
Technology Need	LO_2/LH_2 reduced boiloff flight demo	N/A	N/A	May be required	Required	Required	Required	Required	Required	Required	Required
Technology Need	LO_2/LH_2 reduced boiloff and other CPS tech development	N/A	N/A	May be required	Required	Required	Required	Required	Required	Required	Required
Technology Need	LO_2/LH_2 zero boiloff tech development	N/A	N/A	N/A	N/A	May be required	May be required	May be required	Probably required	Probably required	Probably required

*Adapted from chart 26 of HEFT Final Briefing dated 1/18/2011, posted at: <http://www.nasa.gov/pdf/511089main_HEFT_Final_Brief_508_20110111.pdf>

Analysis of these multiple viewpoints led to the conclusion that long-duration cryogenic propellant storage and transfer is not only a suite of technologies that is broadly beneficial for a range of spacecraft systems, but is most notably an enabler for key architecture elements such as long-duration CPSs, in-space cryogenic propellant storage facilities (e.g., tankers and depots), and nuclear thermal propulsion systems.

The fact that CFM technology advancement has stalled at TRLs 5 and 6 due to lack of a flight demonstration is well documented. In fact, the technology gaps identified in this present study bear remarkable similarity to the gaps identified in the COLDSAT studies conducted by three independent contractor teams in the early 1990's and summarized in 1997 by Glenn Research Center (GRC). The intervening years have witnessed cryocooler technologies development, vastly improved analytical tools, significantly expanded 1-g test databases, and improved gauging technologies; however, key technologies that are dependent on the acceleration environment (i.e., 1-g versus microgravity) are still awaiting advancement to TRLs 6 and 7. Whether NASA selects an architecture based on a heavy-lift vehicle or an on-orbit cryogenic storage capability, the need for a flight demonstration of cryogenic propellant storage is required in the near term to bring full-scale development risks to acceptable levels. The study team has identified the CPS preliminary design review (PDR) as the most likely driving constraint for the CPST flight date. A flight program such as the CPS needs TRLs 6 and 7 in the PDR timeframe to avoid unacceptable levels of development risks. The team believes the CPST flight should be complete and the flight data correlated in time for CPST conclusions to be considered with the CPS preliminary design package development. Assuming the CPS PDR occurs in the fourth quarter of CY 2017, and allowing time for PDR package development, this line of reasoning leads to a CPST flight that ends no later than the second quarter of CY 2017, indicating a launch date in the early fourth quarter of CY 2016.

2.1.2 Value and Benefits for Future Applications

The ability to store and expel (or transfer) subcritical cryogenic liquids in space over long periods of time has significant benefits for NASA missions. Since cryogenic oxygen and hydrogen have much higher performance characteristics than conventional storable propellants, a capability to store and use LO_2 and LH_2 in the propulsion systems of long-duration spacecraft opens a path to shorter transit times and/or increased payload masses for long-duration missions to the Moon, Mars, and other destinations. Considering that many propellants compatible with in situ propellant production must be stored as cryogenic liquids to meet vehicle mass and volume requirements, these new cryogenic capabilities would also help bridge the gap to 'live-off-the-land' architectures, such as the envisioned production of methane, hydrogen, or oxygen by carbon dioxide reduction or water electrolysis. These same storage, acquisition, and gauging technologies are also useful for accelerated missions to Mars and outer planets, since nuclear thermal propulsion systems require the use of these CFM technologies to assure that their propellants (such as xenon) are stored within acceptable volume and dry-mass constraints.

The ability for long-term, low-boiloff cryogenic storage also enables extended duration upper stages that are a confirmed interest to the commercial launch providers and their customers, including the Department of Defense. These longer duration, upper-stage missions enable a wider range of orbit selections for Earth-orbiting satellites.

2.1.3 Mission Goal Prioritization

During this study, a team of subject matter experts ranked the mission objectives in light of perceived stakeholder needs. This team represented a cross section of NASA's CFM technologists as well as spacecraft fluid system and propulsion system development experts. This community of experts represented experience in CFM ground testing, the design and operation of both storable and cryogenic propulsion systems, and the design of propellant transfer systems. Viewing the most likely and most immediate stakeholders to be the CPS or an in-space propellant storage function (i.e., a tanker or depot), the group ranked the CPST mission objectives to ensure that the most daunting development risks and the technology gaps without feasible workarounds were ranked highest. The group was careful to assure that the ranking reflected 'mission pull' rather than 'technology push.' The results of this ranking are shown in table 2.

Hydrogen technologies ranked higher than oxygen technologies. Closing the hydrogen technology gaps significantly reduces the risk of failing to conquer the oxygen gaps, due to hydrogen's significantly lower boiling point and decreased surface tension properties. Storage ranked highest, since failure to achieve significantly reduced boiloff (RBO) rates would be a first-order driver in sizing (and paying for) future architecture elements.

Acquisition and expulsion of bubble-free cryogenic liquids ranked below storage due to the catastrophic hazard associated with feeding entrained bubbles to a rocket engine and due to the complex interaction of surface tension and thermodynamics in the design and operation of the acquisition device(s) in a long-duration, microgravity environment. It is also worth mentioning that acquisition is an enabler for systems that purport to use cryogenic liquids as reaction control system (RCS) propellants (not merely as main engine propellants), since the ability to settle main engine propellants would require a robust acquisition capability within the RCS.

Gauging in microgravity ranked third. The gauging function is critical for the efficient operation of a fully capable storage system or propulsion system. The risk is initially offset by the somewhat inefficient workaround of carrying additional RCS propellant to settle the cryogenic liquids, enabling the use of somewhat lower risk, settled gauging methods. There are scenarios (such as the need for multiple gauging operations near the end of a depot/tanker transfer operation) where settled gauging would be an unwieldy, if not completely impractical, solution. Hence, this third-place ranking still represents a very significant need.

Finally, the overall transfer technology suite ranked fourth—not because this technology is unimportant, but because the others represented more significant development risks in the collective opinion of the expert team. It is noteworthy that the team concluded that the demonstration of an automated fluid coupling in flight did not rise to the level of a flight objective. Close examination revealed that this coupling can be effectively matured through ground tests (including cryogenic flow testing and 6-degree-of-freedom mate/demate simulations) and does not require the microgravity environment to achieve TRL 6, since the microgravity environment does not significantly affect the design or operation of the coupling.

2.1.4 Satisfaction of Mission Objectives

The POD architecture settled on a hydrogen-only technology demonstration based on the higher ranking hydrogen technology objectives as well as the challenging budget. As explained above, the hydrogen-only approach does not close the oxygen technology gaps, but it does significantly reduce the risks associated with the development of cryogenic oxygen capabilities. Within the fiscal realities of the technology demonstration mission, the hydrogen technologies demonstration is considered the very best return on investment for the eventual stakeholders in these technologies.

2.2 Mission Concept

2.2.1 Mission Analysis

2.2.1.1 Ground Rules and Assumptions. The guiding ground rules for the mission analysis work are listed in table 7. Since the target users of this technology are space missions that will require storage of cryogenic propellants on both LEO, translunar, and interplanetary trajectories, the thermal team decided that the best relevant thermal environment for the demonstration would be a LEO with an inclination that included both sun and shadow. As a result, the mission analysis team ground ruled a low-Earth circular orbit with a target inclination that would allow time in both sun and shadow, and decided to let the lifetime requirement determine the initial orbit altitude. Experience suggested that the target altitude for a circular orbit would most likely be in the range of 300 to 600 km.

Table 7. Mission analysis ground rules and assumptions.

Target orbit	LEO circular, ≈ 500-km altitude (determined by lifetime requirement), with inclination relevant to future space missions that will require cryogenic propellant storage
Nominal mission duration	6 mo
Target orbital lifetime	1 yr
Controlled reentry	Yes; the team assumes that a controlled reentry is required due to the anticipated mass of the spacecraft
Launch year	2016

For this study, the target orbital lifetime is defined as the estimated time that the spacecraft will orbit before atmospheric reentry, assuming no stationkeeping. Based on data from previous studies and on the advice of the Orbital Debris Program Office at Kennedy Space Center (KSC), analysts set the target orbit lifetime to be twice the target spacecraft lifetime or 1 yr beyond the end of the mission, whichever was less. This resulted in a target orbit lifetime of 1 yr. Based on the anticipated mass of the spacecraft, and in conjunction with data from previous studies, the team decided to assume that a controlled reentry would be required to keep the risk of human casualty below the required probability of 1:10,000. Since the design was not complete until the end of the study, the team recommends that the assumption of controlled reentry be reexamined in follow-on work. A preliminary reentry analysis completed by the Mission Design Lab at GSFC supports the controlled reentry assumption, but not all spacecraft components were included.

The assumed launch year of 2016 results in conservative lifetime estimates since the solar flux values are expected to be decreasing from 2016 (which is the earliest that CPST is expected to launch) through 2020. Therefore, if the lifetime requirements are met for the year 2016, then they should be met or exceeded for all subsequent years up to 2020.

2.2.1.2 Mission Profile. Following launch, the spacecraft will operate for 2 wk in checkout mode. Assuming satisfactory completion of the checkout operations, the next month on orbit will involve the passive CFM storage demonstration. Afterwards, the spacecraft will perform the active portion of the CFM demonstrations. Following an optional extended mission phase of up to 6 mo, the spacecraft will perform a controlled reentry into the Earth's atmosphere.

2.2.1.3 Orbit Lifetime Analysis and Target Orbit Altitude. Orbit lifetime analysis ensures that the spacecraft will remain in the required orbit and not be overcome by aerodynamic drag or other perturbations over the required duration of the mission. Given a 6-mo mission, a 1-yr orbit lifetime requirement conservatively results in an initial orbit altitude that will easily meet the mission requirements. The mission analysis team used NASA's Debris Assessment Software (DAS) version 2.0, developed at Johnson Space Center (JSC), and Life TIME (LTIME), an orbit lifetime tool developed at Marshall Space Flight Center (MSFC), to estimate the orbit lifetime and decay values for CPST. Stationkeeping requirements came from a simple custom tool developed within the MSFC Advanced Concepts Office.

The aerodynamic area tool in DAS provided a convenient way to estimate the area/mass ratio for CPST. The user inputs the basic geometry of the spacecraft, specifies an appropriate aerodynamic attitude, and lets DAS determine the aerodynamic area. The configuration used in DAS to determine the aerodynamic area is shown in figure 5. Assuming an aerodynamic attitude of 'random tumbling' resulted in an aerodynamic area of 14.16 m^2. This attitude represents the best match among the options of random tumbling, gravity gradient, and fixed attitude, since the actual spacecraft attitude will be solar inertial with the aft end of the spacecraft facing the Sun. This solar inertial attitude results in the spacecraft slowly tumbling relative to the oncoming atmosphere during flight.

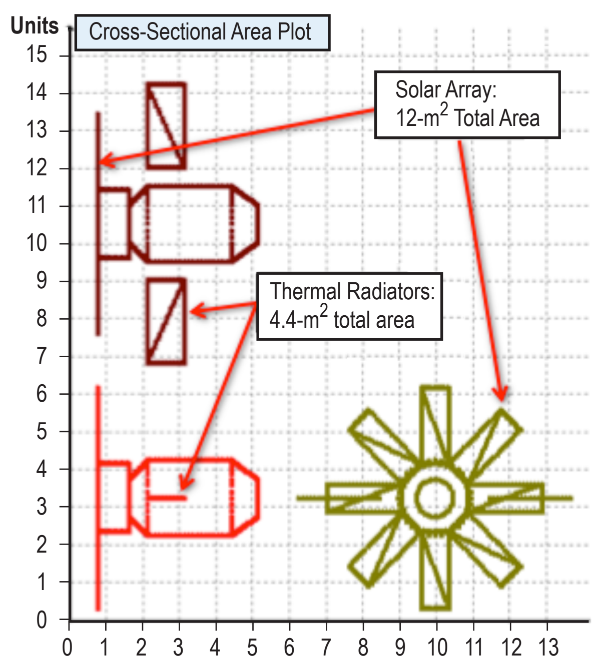


Figure 5. Configuration used in DAS to determine the area/mass ratio.

The resulting DAS output for the aerodynamic area is an input into LTIME, along with the spacecraft mass, orbit, aerodynamic parameters, and other values as listed in table 8. Analysts compared a few test cases of LTIME with two other computer applications that can also estimate lifetime: DAS and the Satellite Tool Kit (STK). Since the results compared well, the team used LTIME due to its speed and more user-friendly output. A comparison of the tool outputs indicated that a nominal solar flux along with a drag coefficient of 1.8 would result in LTIME output closely matching the other tools.

Table 8. Input parameters for the LTIME satellite lifetime program.

Epoch	1/1/2016
Orbit altitude	300–600 km, circular
Inclination	28.5 deg (see text for explanation)
Longitude of ascending node, argument of perigee, and mean anomaly	0
Area	14.16 m^2
Mass	1,500–2,750 kg
Drag coefficient	1.8 (see text for explanation)
Predicted solar flux	Nominal (see text for explanation)

Initially, the spacecraft was to be launched from Cape Canaveral Air Force Station (CCAFS) to a 28.5-deg inclination. However, midway through the study, launching from WFF became a possibility, so the team investigated the effect of orbit inclination on orbit lifetime. Based on the data listed in table 9, the team concluded that orbit inclination has a minimal effect on lifetime (up to 40 deg), with launch year having a much larger influence. Thus, all orbit lifetime runs in LTIME were with an inclination of 28.5 deg.

Table 9. Effect of launch and orbit inclination on orbit lifetime.

	Altitude: 500 km Area/Mass: 0.0050		
	Inclination (deg)		
Launch Year	0	28.5	40
2017	7	7.1	7.1
2017.5	6.6	6.6	6.7
2018	6.1	6.2	6.2
2018.5	5.6	5.7	5.7
2019	5.1	5.2	5.3
2019.5	4.7	4.7	4.8
2020	4.2	4.2	4.3
2021	3.3	3.3	3.4
2022	2.9	3	3.1
2023	3.4	3.5	3.7

The range of masses used in the analysis was an attempt to cover a wide range of area/mass ratios in anticipation of the mass and/or configuration changing. Even if the configuration changes considerably, as long as the resulting area/mass ratio is covered in previous LTIME runs, no additional runs are required. The resulting area/mass ratios are listed in table 10. The value in **bold** represents the estimated values for the final configuration of this study.

Table 10. Resulting area/mass ratios used in the orbit lifetime analysis.

Mass (kg)	Area/Mass (kg/m^2)
1,500	0.0094
1,750	0.0081
2,000	0.0071
2,200	**0.0064**
2,500	0.0057
2,750	0.0051

Many runs of LTIME resulted in a large table of data listing the expected orbit lifetimes for various area/mass ratios. An abbreviated table is provided in table 11. One can see that for the current configuration's area/mass ratio of 0.0064, an initial circular orbit altitude of 400 km provides plenty of margin on the orbit lifetime since area/mass ratios of 0.0060 and 0.0080 result in lifetimes of 1.54 and 1.11 yr, respectively. Note that this analysis so far does not include the effect of launch vehicle insertion errors, which are discussed below. Also, since the predicted solar flux values are decreasing throughout 2016–2020, these tabulated values will increase for launches beyond 2016.

Table 11 also lists estimated stationkeeping delta-velocity (ΔV) requirements. These values are rough estimates given by taking the orbit decay rates from LTIME and putting these into a custom tool that determines the required ΔV based on a user-specified maximum amount of altitude decay before boosting the orbit back to its original value. (Only corrections to altitude are considered here.) In this study, analysts set the maximum altitude decay value to 0.2 km before reboost was necessary. However, while the values were calculated and included in the results, the team's opinion is that 'stationkeeping is not required for CPST' since specific altitude and inclination requirements are not part of the requirements, the only requirement being lifetime.

The team's final step in determining the recommended target initial orbit altitude was to consider launch vehicle insertion errors. Since at least one of the candidate launch vehicles contains a solid stage, the insertion errors may be large enough to decrease the orbit lifetime substantially. Insertion errors from two candidate vehicles, the Minotaur IV and the Taurus II, are listed in table 12. While the current CPST design is too heavy for the Minotaur IV, it is included here for comparison purposes.

Table 11. Orbit lifetime and stationkeeping estimates.

Area/Mass Ratio	Initial Circular Orbit Altitude (km)	ΔV Required for Stationkeeping (m/s)	Altitude After 0.5 yr With No Stationkeeping (km)	Orbit Lifetime With No Stationkeeping (yr)
0.0050	300	71	Reentry	0.16
	350	24	168	0.55
	400	8.6	382	1.90
	450	3.4	444	6.37
	500	1.6	497	9.88
0.0060	300	84	Reentry	0.13
	350	28	Reentry	0.44
	400	10	377	1.54
	450	4	442	5.65
	500	1.8	497	8.96
0.0070	300	99	Reentry	0.11
	350	33	Reentry	0.33
	400	12	377	1.11
	450	4.7	442	3.97
	500	2	497	7.90
0.0080	300	113	Reentry	0.10
	350	38	Reentry	0.33
	400	14	366	1.11
	450	5.4	439	3.97
	500	2.2	496	7.90

Table 12. Orbit insertion errors for two example launch vehicles.

	Insertion Errors (km)	
	Minotaur IV	Taurus II
Insertion apse	18.5	15
Noninsertion apse	92.6	80

These insertion errors can greatly affect the orbit lifetime. To estimate the magnitude of the effect, analysts ran several additional LTIME runs with the estimated insertion errors included. The results are listed in table 13. For the purposes of this exercise, the team used an area/mass ratio of 0.0070; time constraints did not allow running a variety of area/mass ratios. Launching the spacecraft on a Taurus II to a target circular orbit of 400 km could result in a worst-case orbit of 320×385 km, and a decrease in orbit lifetime from 470 to 135 days, which no longer meets the lifetime requirement. One solution is to target a 450-km circular orbit, so that even the worst-case resulting orbit will have sufficient lifetime (466 days in this case) to satisfy the mission requirements.

Table 13. The effect of launch vehicle insertion errors on orbit lifetime.

	Target Orbit (and lifetime)	Actual Orbit	
		Minotaur IV	Taurus II
Apogee	400	381.5 km	385 km
Perigee	400	307.4 km	320 km
Lifetime*	470	110 days	135 days
Apogee	450	431.5 km	435 km
Perigee	450	357.4 km	370 km
Lifetime*	1,760	376 days	466 days
Apogee	500	481.5 km	485 km
Perigee	500	407.4 km	420 km
Lifetime*	3,040	1,350 days	1,750 days

* Lifetime assumes 2016 launch, area/mass of 0.007.

Therefore, taking into consideration launch vehicle insertion errors, launch date, solar flux, and the possible area/mass ratios of the CPST configuration, 'the mission analysis team recommends a target circular orbit altitude of 450 km.' Additional options that could be investigated include, but are not limited to, using high-accuracy orbit insertion options for the Taurus II, having the spacecraft perform propulsive maneuvers to correct for insertion errors, and having the spacecraft perform stationkeeping.

2.2.1.4 End-of-Life Disposal and Risk of Human Casualty. The team completed a preliminary assessment of the Risk of Human Casualty as specified in NASA-STD-8719.14, Process for Limiting Orbital Debris, NASA's DAS, version 2.0.1, was used in the analysis.[1] While not all spacecraft components were included in the assessment, those that were resulted in a risk of human casualty of 1:2,100, which exceeds the NASA standard of 1:10,000. Therefore, the analysis shows that a controlled reentry will probably be required.

2.2.1.5 Preliminary Propellant Requirements. The propellant was estimated to accomplish three major tasks: propellant settling during the fluid transfer and mass gauging experiments, collision avoidance, and the end-of-life disposal of the spacecraft. For the propellant settling, the assumed initial mass was 2,210 kg. The fluid transfer experiments required an average of 3×10^{-5} g for a total of 312 min, while the mass gaugings required 1×10^{-3} g for 48 min. This resulted in a propellant settling ΔV of 34 m/s. A collision avoidance maneuver was included with a ΔV of 3 m/s and an additional 0.2 m/s for attitude control.

Before the controlled reentry, most of the cryogenic propellant must be dumped overboard. Three percent of the initial load is assumed to remain in the tanks and lines. The deorbit (from the 450-km circular orbit) is separated into three maneuvers. The first two burns, both at 450 km, reduce the perigee altitude to 150 km. The final maneuver is performed such that the flight path angle is –1.2° at an altitude of 60 km. A preliminary finite-burn analysis resulted in a total deorbit ΔV of 144.3 m/s with an additional 14.3 m/s for attitude control.

The mission events and corresponding propellant requirements are listed in table 14. Margins of 30% were added to the settling and avoidance ΔVs. More fidelity went into the deorbit analysis, justifying a smaller 5% margin. The required maneuver propellant was 192 kg. A residual amount of 5% was added to account for the propellant trapped in the tanks and lines, resulting in a total loaded propellant requirement of 202 kg. For these propellant computations, a constant specific impulse (I_{sp}) of 220 s was assumed, which is typical for monopropellant propulsion systems. A more detailed analysis is required to track the variation of the I_{sp} and thrust during the various events, especially the controlled reentry.

Table 14. Preliminary mission propellant estimate.

Event	Mass (kg)	MPS ΔV (m/s)	ACS ΔV (m/s)	Margin (%)	Total ΔV (m/s)	Propellant (k/g)
Propellant settling	2,210	–	34	30	44.2	44.8
Collision avoidance	2,175	3	0.2	30	4.1	4.2
Deorbit	1,923	144.3	14.3	5	166.5	142.9
					Maneuver propellant:	192
					Residuals:	10
					Total propellant requirement:	202

2.2.2 Technology Demonstration Timeline

The CPST mission timeline was developed to satisfy the identified needs, goals, and objectives as well as the requirement for a mission duration of 6 mo.

2.2.2.1 Storage Demonstration Timeline. The top-level mission timeline is shown in figure 6 and begins with on-orbit arrival. There will be an initial checkout period for the vehicle and the CFS systems lasting an estimated 10 days. At or about the same time, the passive storage demonstration will begin with an initial settled mass gauging event and activation of the CFS data collection system. It is vital that thermal performance data collection begin as soon as possible in order to characterize the post-ascent thermal transient due to venting and offgassing of the storage tank insulation system. The passive storage demonstration is planned to last 1 mo. This should allow adequate time for data collection through the thermal transient period and subsequent steady-state period.

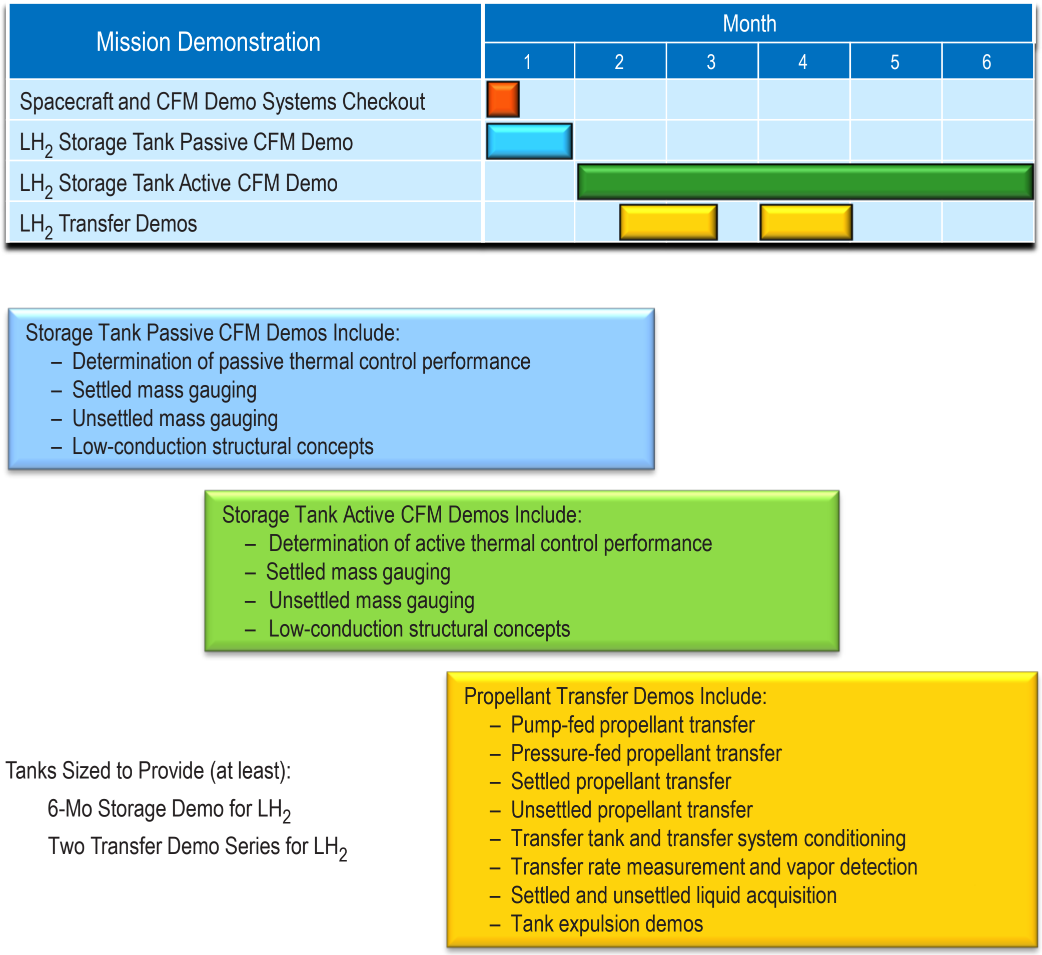


Figure 6. CPST Government POD demonstration timeline.

Upon completion of the passive storage demonstration, the active storage demonstration will begin with another settled mass gauging event and startup of the storage tank's active cooling system. The remainder of the mission will be conducted with active cooling of the storage tank. The active storage demonstration is expected to last 5 mo, with adequate opportunities for collection of steady-state thermal performance data under active cooling.

2.2.2.2 Transfer Demonstration Timeline. During the active storage demonstration, two 30-day transfer demonstration series will be completed. Each transfer demonstration series will include four, two-way transfers of LH_2 between the storage and transfer tanks (i.e., transfer from storage tank to transfer tank followed by return transfer from transfer tank to storage tank). Two transfers will be performed under settled conditions—one pressure fed and the other pump fed. Similarly, two transfers will be performed under unsettled conditions—one pressure fed and the other pump fed. In addition, the transfer tank and transfer system will be chilled from the spacecraft internal ambient temperature to an initial transfer hardware target temperature prior to each two-way transfer. After each two-way transfer, the transfer tank and transfer system will be warmed to the spacecraft internal ambient temperature or other warm target temperature in preparation for the next chilldown and two-way transfer.

The mission propellant inventory profile is shown in figure 7. The corresponding propellant loss and heat load breakdown by location are given in table 15. These show that the propellant loss at the end of mission is less than the initial propellant load of 260 kg. Predicted tank heat loads are based on analytical results from the CPST Government POD vehicle flight thermal model (June 2011), constructed using Thermal Desktop® with RadCAD® and SINDA/FLUINT. A margin of 50% is applied to all heat loads.

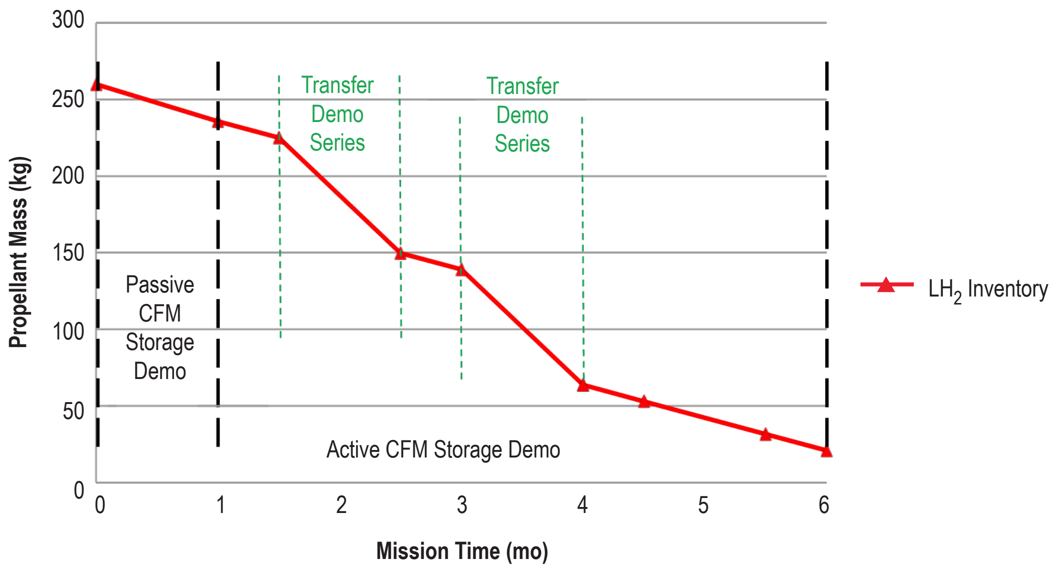


Figure 7. CPST Government POD propellant loss profile.

Table 15. CPST Government POD propellant loss and heating rates.

	Propel-lant Loss (kg)	Steady-State Heating Rates							
		Tot Q Tank (W)	Str Q Tank (W)	Pen Q Tank (W)	Rad Q Tank (W)	Tot Q BAC (W)	Str Q BAC (W)	PenQ BAC (W)	Rad Q BAC (W)
LH_2									
Storage tank									
Passive storage	24.2	4.13	2.50	0.61	1.02	–	–	–	–
Active storage (reduced boiloff)	107.2	3.66	2.50	0.61	0.55	2.10	–	–	2.10
Transfer tank (includes chilldown and residuals)	50.4	2.02	1.31	0.53	0.18	–	–	–	–
Transfer system (includes chilldown and residuals	57.5	10.01	–	–	10.01	–	–	–	–
Storage tank residual	7.8	–	–	–	–	–	–	–	–
Total propellant loss	247	–	–	–	–	–	–	–	–

Interception of conductive heat loads at the tank supports and penetrations with active cooling was not included in this analysis, due to the lack of a vetted heat exchanger concept. It is hoped that one or more viable heat exchanger concepts will be proven in the CPST WBS 4.0

Technology Maturation activities. The increased conservation of propellant resulting from the interception of conductive heat loads could enable a third transfer demonstration series in the CPST mission timeline. This would be a valuable thermal design feature and should be addressed in future efforts.

Figure 8 depicts the content of each transfer demonstration series in greater detail. The four, two-way transfers with associated chilldown and settled mass gauging activities are shown notionally. Unsettled mass gauging is shown as an ongoing activity during all settled and unsettled periods, since the associated cost in terms of power and data handling is relatively low. The two-way transfers will be scheduled approximately a week apart to allow for downlink and review of CFS data and possible preparation of new instruction for uplink before the next two-way transfer.

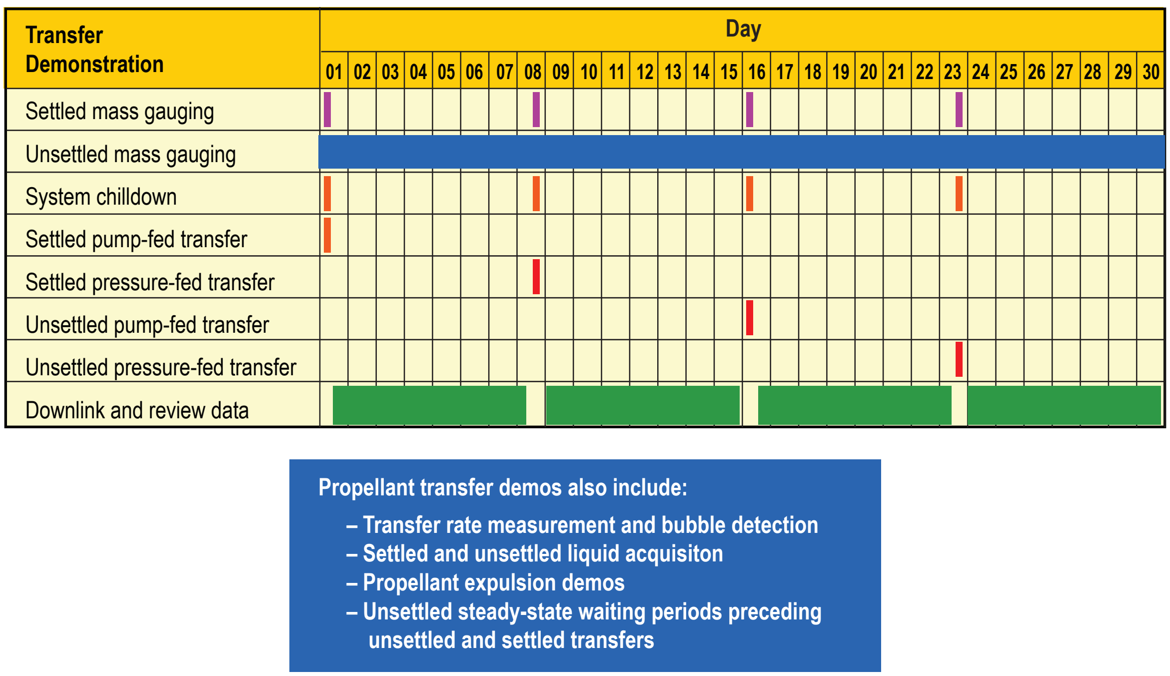


Figure 8. Thirty-day transfer demonstration series timeline.

Tables 16 and 17 provide even more detailed breakdowns of events for individual settled and unsettled two-way transfers, respectively. Approximately 5 hr are required to complete each two-way LH_2 transfer. The sequence of events, the duration of individual events (LH_2 transfers, pretransfer chilldowns, unsettled waiting periods, and mass gauging events) and target acceleration levels for the CFM Technology team provided settled operations.

Due to the mass impact associated with providing settled conditions, the allowable duration of settled conditions will be limited. For purposes of this study it has been assumed that 30 min will be required for each two-way settled transfer and 2 min will be required for each settled mass gauging event. Axial thrust levels of 1 mg and 30 μg have been selected for settled operations. No axial thrust will be provided for unsettled operations. Total durations for each axial thrust level are shown at the bottom of tables 16 and 17.

Table 16. Settled transfer demonstration timeline.

Elapsed Time (min)	Event Duration (min)	Event	Thrust Acceleration (g's)
2	2	Settled mass gauging	0.00003
4	2	Settled mass gauging	0.001
19	15	Unsettled steady state waiting period	0
31	2 × 6	Chilldown prep	0.00003
139	18 × 6	Chilldown	0
141	2	Settled mass gauging	0
156	15	Settled propellant transfer from storage tank	0.00003
158	2	Settled mass gauging	0.00003
160	2	Settled mass gauging	0.001
175	15	Unsettled steady state waiting period	0
177	2	Chilldown prep	0.00003
285	18 × 6	Chilldown	0
287	2	Settled mass gauging	0.00003
302	15	Settled propellant transfer to storage tank	0.00003
304	2	Settled mass gauging	0.00003
306	2	Settled mass gauging	0.001

Total Time at 0.001 g's: 24 min for CPST Government POD (total associated with all settled transfers).
Total Time at 0.00003 g's: 216 min for CPST Government POD (total associated with all settled transfers).

Table 17. Unsettled transfer demonstration timeline.

Elapsed Time (min)	Event Duration (min)	Event	Thrust Acceleration (g's)
2	2	Settled mass gauging	0.00003
4	2	Settled mass gauging	0.001
16	2 × 6	Chilldown prep	0.00003
124	18 × 6	Chilldown	0
126	2	Settled mass gauging	0.00003
141	15	Unsettled steady-state waiting period	0
156	15	Unsettled propellant transfer from storage tank	0
158	2	Settled mass gauging	0.00003
160	2	Settled mass gauging	0.001
162	2	Chilldown prep	0.00003
270	18 × 6	Chilldown	0
272	2	Settled mass gauging	0.00003
287	15	Unsettled steady-state waiting period	0
302	15	Unsettled propellant transfer to storage tank	0
304	2	Settled mass gauging	0.00003
306	2	Settled mass gauging	0.001

Total Time at 0.001 g's: 24 min for CPST Government POD (total associated with all unsettled transfers).
Total Time at 0.00003 g's: 96 min for CPST Government POD (total associated with all unsettled transfers).

2.2.2.3 Minimum Mission Timeline. During the course of the study, it was determined that by starting the first transfer demonstration series 2 wk earlier, contingency storage and transfer demonstrations could be accomplished within the first 2 mo of the mission timeline, if necessary.

2.2.3 Mission Communications Architecture

The communications architecture covers the hardware, software, spacecraft and CFM command and control terminals, and networks required to operate CPST. The major components and their functions are as follows:

- Ground stations (White Sands (WS1), Wallops, and Santiago (AGO) with Dongara (AUWA) and Hawaii (USHI) as backup):
  - Receive spacecraft and CFM data from CPST.
  - Send commands to CPST (minimal command processing).
  - Send the spacecraft data to the Missions Operations Center (MOC).
  - Retransmit data upon request.

- Data distribution networks—combination of NASA integrated services network and the local area network:
  - Provide connectivity between ground station(s) and the MOC and Technology Operations Center (TOC).
  - Meet any needs for dedicated voice communication between MOC and TOC.

- MOC—overall command and control:
  - Convert CFM objectives into spacecraft and CFM commands in order to perform mission operations (command fabrication).
  - Perform command stream validation.
  - Analyze and monitor real-time telemetry from the spacecraft for passes that are scheduled.
  - Analyze real-time and stored health and status to generate trending data.
  - Ensure healthy operation of the spacecraft bus.
  - Manage the spacecraft onboard data recorder.
  - Detect, report, and troubleshoot anomalous conditions.
  - Transmit scripts, commands, data tables, and software updates to CPST via the ground station(s).
  - Receive spacecraft data from CPST via the ground station(s).
  - Station scheduling.
  - Maintain data dictionaries.
  - Archive scripts, commands, data tables, and software updates that were sent to CPST data storage and archive.

- Flight dynamics function:
  - Perform targeting for controlled reentry.
  - Provide acquisition data to the tracking networks.
  - Provide time history of orbit and attitude based on downlinked spacecraft telemetry.

• TOC—demonstration objectives and analysis:
  – Provide demonstration objectives and onboard data recorder recommendations to MOC.
  – Receive CFM data from MOC.
  – Analyze CFM health and status data post pass to ensure healthy operation of the CFM and recommend modification to operational processes.
  – Detect, report, and support trouble shooting of anomalous conditions in the CFM.
  – Support CFM sensor calibrations, algorithms, trending, and built-in tests.

2.2.4 Prelaunch to Spacecraft Separation

Figure 9 shows a simplified view of the communications architecture during this phase. CPST will undergo cryogenic loading and prelaunch checkout at the launch site prior to launch. This will require MOC, TOC, and/or an engineering support room at the launch site. CPST will not have telemetry interleaved with the vehicle data, but due to the uniqueness of cryogenic loading at the pad, there may be umbilical disconnects that are controlled through the vehicle launch sequencer. Also, the cryogenic loading information may need to be routed to the launch vehicle control room for the launch vehicle team to review.

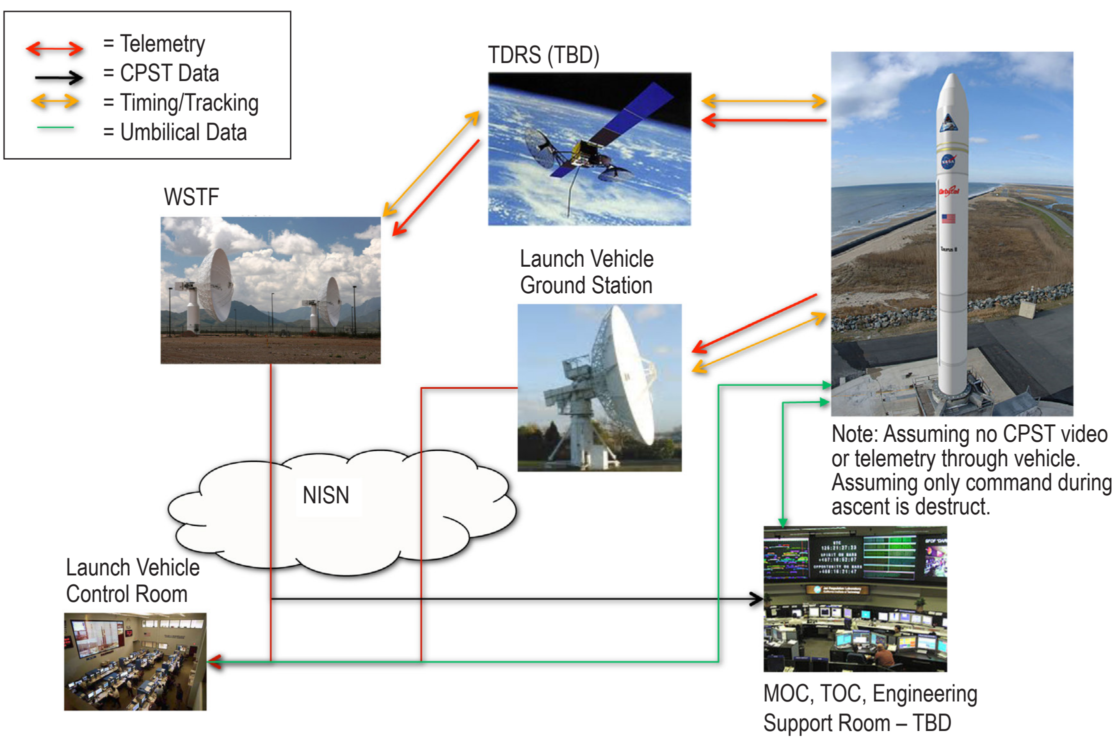


Figure 9. Prelaunch to spacecraft separation communications architecture.

During ascent, the use of a tracking and data relay satellite (TDRS) is to be determined. The launch vehicle will send launch vehicle telemetry to its ground station. At this point it is assumed that CPST will record all data during ascent for later transmission.

2.2.5 Spacecraft Separation to Controlled Deorbit

Figure 10 shows a simplified view of the communications architecture in this phase.

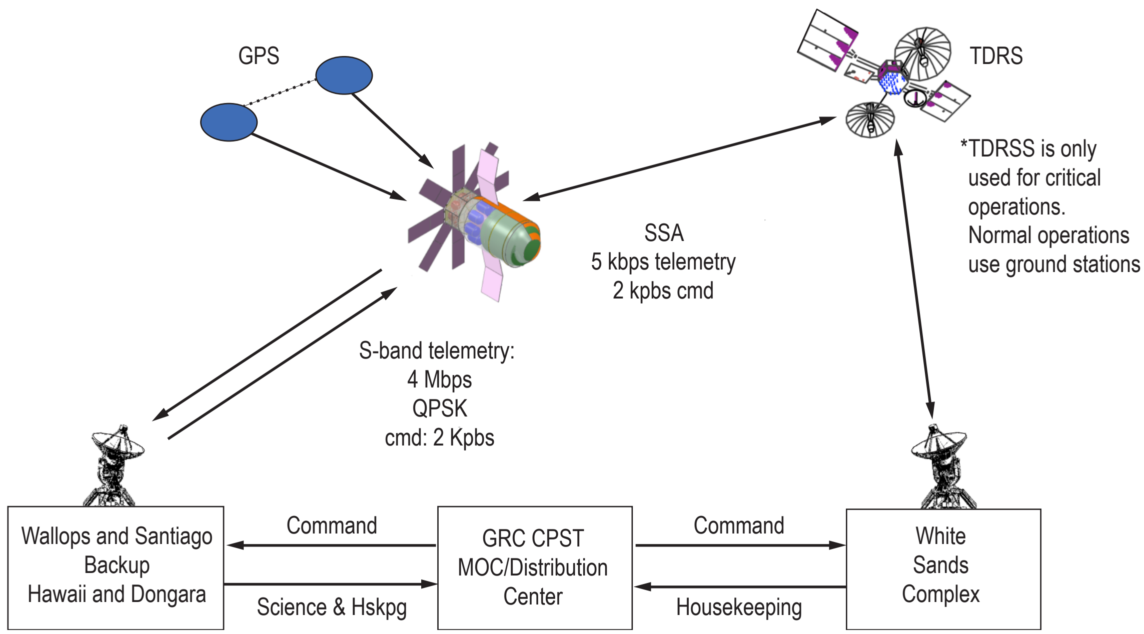


Figure 10. Spacecraft-controlled deorbit communications architecture.

The first 2 wk after spacecraft separation, during on-orbit checkout, a tracking and data relay satellite system (TDRSS) would be used in addition to the ground stations, but after that period, normal operation would ensue and ground stations would be used primarily.

The MOC distributes all CFM data to the TOC for their review and analysis.

2.2.6 S-Band

S-band is required for uplinking commands to CPST. Uplink of commands can be done through TDRSS or one of the ground stations. The S-band frequency offers 4 Mbps of telemetry downlink to the ground stations and 5 kbps telemetry downlink through TDRSS. After the fluid transfers, all the CFM data and video can be downlinked within 24 hr; therefore, a higher frequency band (such as X-band) is not required.

The analysis in figure 10 shows the ground coverage for White Sands (WS1), Wallops, and Santiago (AGO) with Dongara (AUWA) and Hawaii (USHI) as backup. This analysis used a 500-km altitude, 40-deg inclination, and 5-deg minimum station elevation. The S-band coverage summary is shown in table 18.

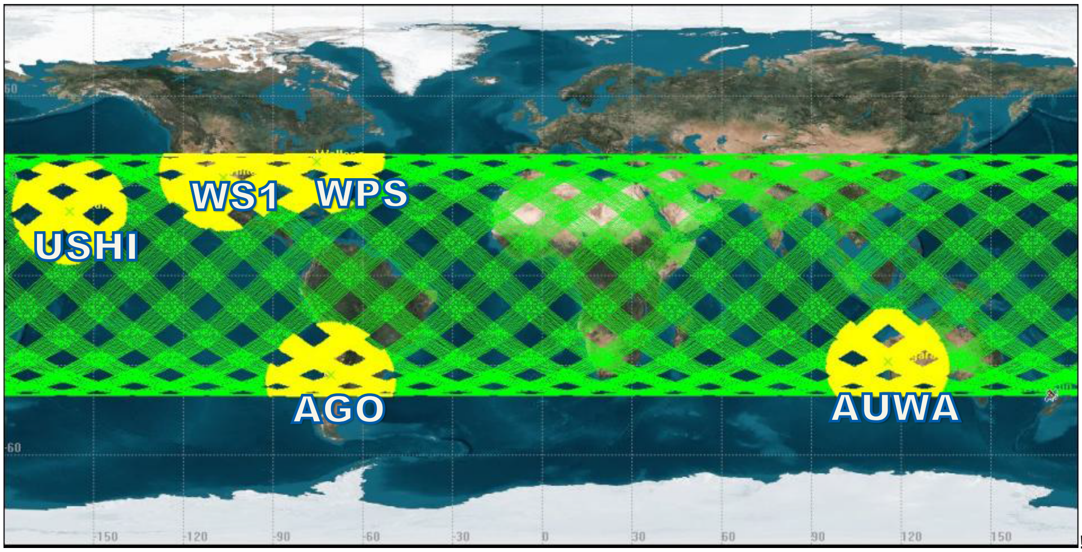


Figure 11. Ground station S-band coverage.

Table 18. S-band coverage summary.

Ground Station	Passes per Day*	Average Pass Duration** (min)	Maximum Pass Duration (min)	Total Minutes per Day
White Sands	6	8.95	9.74.	53.68
Wallops	5.21	9.26	9.78	48.26
Santiago	5.71	9	9.76	51.42
Dongara	6.71	8.43	9.74	56.63
Hawaii	5	8.24	9.71	41.22
Station coverage for the S-band assumption of >5° elevation of the GRC CPST satellite in a 500-km circular orbit inclined 40°				

* Numbers based on a random 14-day coverage; only passes >5 min are considered.
** Under favorable conditions, passes may be longer but cannot always be assured.

The peak data volume per day is 18 Gbits, which required about nine passes to downlink to the ground. The peak occurs after a fluid transfer experiment. At other times, the data volume is about 6 Gbits per day, which requires three passes to downlink.

2.3 Demonstration Flight System

2.3.1 Configuration Overview

The layout of the CPST configuration can be divided into two main sections: a CFS and the spacecraft bus (fig. 12). The CFS is the experiment part of the spacecraft. The spacecraft bus provides the platform for the CFS to conduct its required mission and test objectives.

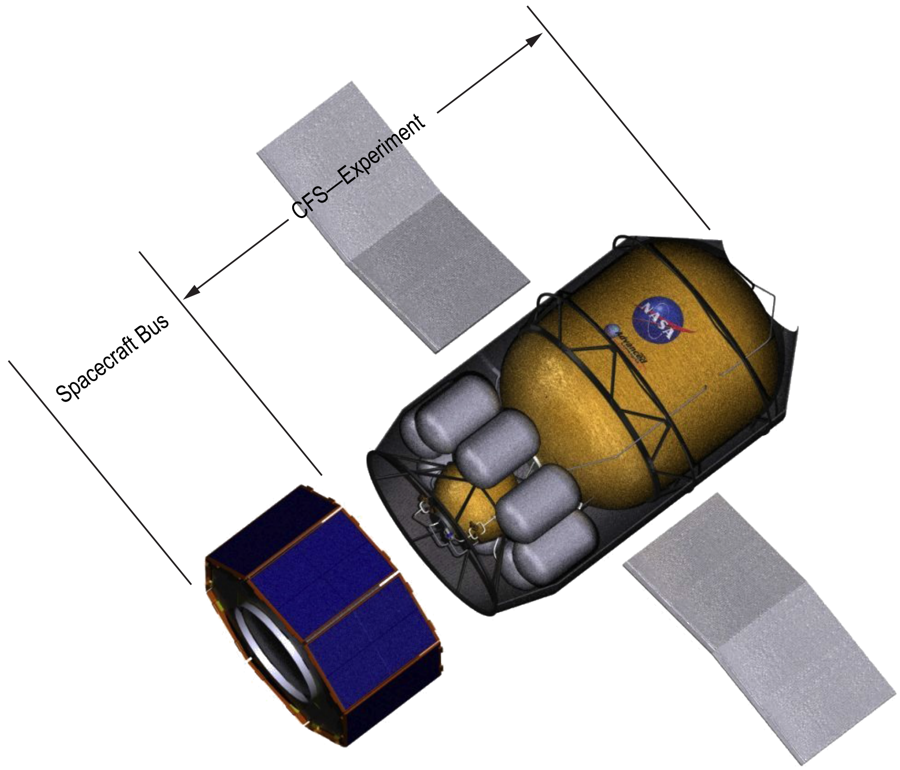


Figure 12. CPST primary sections.

2.3.1.1 Design Drivers. The main design constraint was to meet the LH_2 volume requirements of the storage and transfer tanks. In addition to the volume requirements, the other design driver was the maximum allowable payload size of the expected launch vehicle (fig. 13). Early in the conceptual phase a Minotaur IV launch vehicle was chosen. The storage tank was reduced in diameter to allow for the thermal isolating support structure and anticipated thermal insulation thicknesses. A final diameter of 1.7 m was selected. After examining the transfer tank sizes and dimensions of the pressurant tanks, it was decided to place the storage tank at the top of the spacecraft. The transfer tank is located directly below the storage tank in a diameter transition zone. The spacecraft bus mounting diameter and the outside diameter of the CFS creates the zone. This area was also selected to mount the eight pressurant tanks. This was done to keep the center gravity as low as possible and to group the various CFS components together.

2.3.1.2 Final Launch Configuration. When the final preliminary design was completed, the Minotaur IV launch vehicle did not have sufficient lift capability for spacecraft. The Taurus II was then chosen for the baseline design. This shroud has a larger diameter of 3.4 m (fig. 14). The design could possibly be resized to this new shroud size, which would make the tank dimensions and overall height different. It was decided to stay with the 2-m-diameter size since it would still fit and with the time constraints involved.

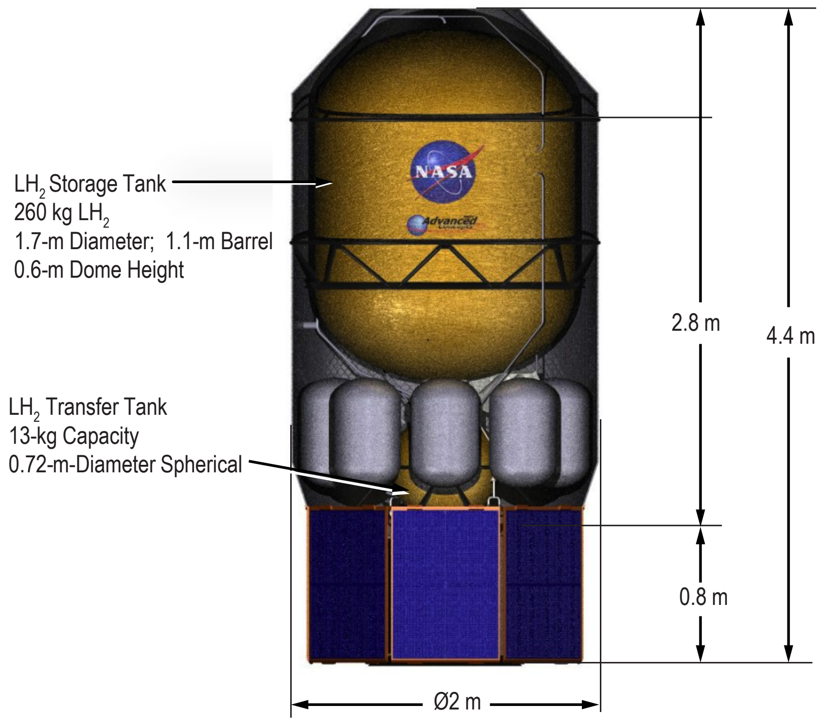


Figure 13. Size and dimensions.

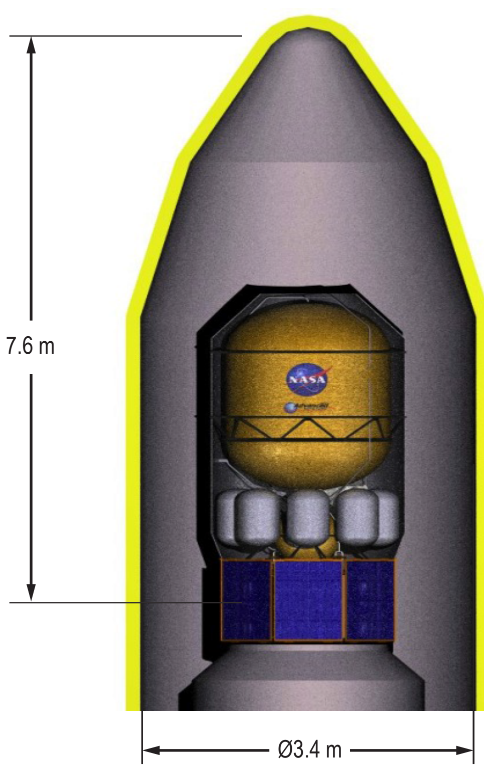


Figure 14. Taurus II launch configuration.

2.3.1.3 On-Orbit Configuration. Once the spacecraft reaches its desired altitude and check-outs are completed, the solar arrays and radiators are deployed. The solar arrays consist of eight separate two-panel-folding arrays, which are stowed against each side of the spacecraft bus. For this level of study, the two double-sided radiators are assumed to be located in the CFM experiment section in close proximity to the thermal subsystem to minimize plumbing (structural shell removed in all figures) (fig. 15).

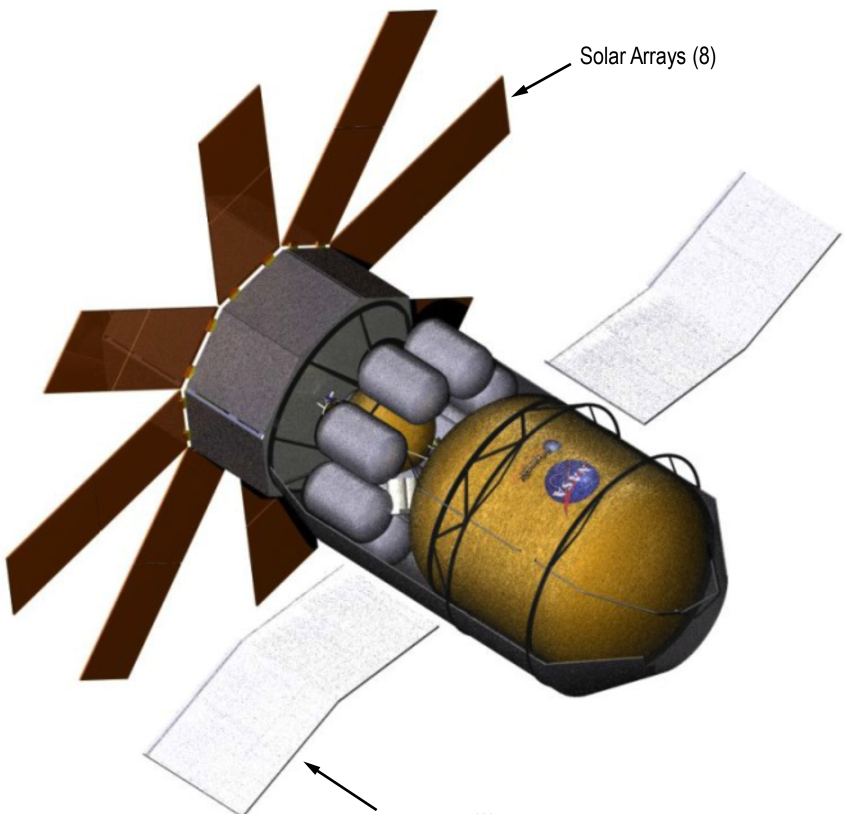


Figure 15. On-orbit configuration.

2.3.2 Payload Fluid System Overview

2.3.2.1 Description. The CPST payload fluid system includes all of the necessary functions and features to allow demonstration of LH_2 cryogenic fluid storage at RBO levels under microgravity conditions, as well as propellant transfer and propellant quantity gauging functions under microgravity and settled conditions. To affect reduced propellant boiloff demonstrations, passive and active thermal control design features are incorporated, including a thermodynamic vent system (TVS), active propellant mixing, multilayer insulation (MLI), low heat leakage structural support struts, and reduced fluid penetrations for the main storage propellant tank. The CPST payload fluid system also employs a cryocooler broad-area cooling (BAC) architecture to actively refrigerate the external surface area of the storage propellant tank. This intercepts heat entering into this tank through the MLI over the surface area of the tank, reducing hydrogen propellant boiloff.

Helium pressurization is provided to support pressurized propellant transfer and pump-fed transfer functions between the storage and transfer propellant tanks. For the pressure-fed transfer function, adequate driving pressure is provided from the source propellant tank to affect the required propellant flow rate to the destination propellant tank. Pressurization control is provided through closed-loop control of fast response pressure control valves ('bang-bang' valves). It is assumed that the helium high-pressure storage and pressure control valves will be thermally conditioned on orbit (heated to approximately –20 to –50 °F) to stay within the temperature capability of existing heritage components and associated soft goods. For the pump-fed transfer function, active pumping using a motor-driven, impeller-type pump generates propellant flow rate. Cryogenic transfer is based on unidirectional flow in transfer lines due to the constraint to use separate connections for liquid entry into and exit from the propellant tanks. Bi-directional flow configurations are possible, but with additional valve arrangement complexity. As a result, the unidirectional flow configuration is the simplest approach and has been incorporated into the payload fluid system design.

The propellant transfer function includes line and transfer tank chill-in conditioning operations for both the settled and unsettled acceleration environments using pulses of propellant flow. Venting once the injected propellant has vaporized and absorbed energy from the line/tank follows each pulse. Cooling of lines and the transfer tank is achieved through automated iterative pulse/vent cycles until temperatures are reached which support no-vent fill operations. The vent system is designed to use a common vent line for both the TVS venting and venting of the two propellant tanks. The vent exhaust is nonpropulsive to minimize vehicle disturbances during these operations. Valving for transfer and venting subsystems utilizes common design to the greatest extent possible to reduce cost and technical risk. Also, many of these valves employ a mechanical latching capability to reduce valve electrically-induced heat loads into the propellant systems. Common or derivative pump designs are also assumed for the TVS mixing and propellant transfer functions. These pump designs need to be tolerant of short-duration, two-phase flow and low margin on cavitation (net positive suction pressure). Derivatives of off-the-shelf laboratory pedigree pump designs are assumed. Key CPST payload fluid design parameters are shown in table 19.

Table 19. CPST payload key design parameters.

CFS design pressure levels:		Comment
Helium pressurization	4,500–250 psi	
Storage and transfer tank subsystems and transfer subsystem	15–40 psia	
Vent subsystem:	0–40 psia	
Cryocooler/BAC circuit	Up to 250 psia	At operating temperature of ~90 K
Propellant flow rates:		
TVS mixing	244 lbm/hr LH_2	
Propellant transfer	78 lbm/hr LH_2	With capability to increase up to TVS mixing flow rates
Propellant storage tank sizing basis:		
Propellant condition	Saturated LH_2 at 15 psia	
Loading ullage volume	10% of useable propellant volume	
Combined tank internal component Volume and loading uncertainty	5% of useable propellant volume	
Storage tank volume	148.6 ft^3	
Transfer tank volume	7.1 ft^3	5% of storage tank volume

2.3.2.2 Payload Fluid System Ground Rules and Assumptions. The CPST payload fluid system ground rules and assumptions are provided in table 20.

Table 20. CPST fluid system ground rules and assumptions

CFM GR&A	
Ground replenishment of cryogens	Ground replenishment until *T*–0
Venting of cryogens	Active venting through umbilical until *T*–0
Ground purge	Dry GN_2 purge provided for LH_2 and LO_2 tank MLI until *T*–0
Tank venting during ascent	Active venting to relieve pressure
Tank venting on orbit	Active and passive venting through a nonpropulsive vent
Valve power control	Fixed current supplied to open valves, holding (reduced level) current used when valve needs to be powered open for extended duration
Ground actuation of valves	Capability to actuate all fill and drain valves and vent valves from ground
Use of latching valves	Latching valves used wherever media is a cryogen
Vehicle structural/thermal design	Similar to CPS nonstructural tank concept
Cryogenic fluid management design	Similar to COLDSAT plus active system
Target storage conditions	Normal boiling point
CFM performance demos	Active CFM and passive CFM
Transfer modes for transfer demos	Pressure-fed and pump-fed
Thrust accelerations for transfer demos	Zero thrust, 0.00003 g axial, 0.001 g axial
Other demos	Mass gauging, liquid acquisition
Post-ascent thermal transient	Passive CFM initiated ASAP to assess thermal transient
Margin on all heat loads	50%

Key CPST payload ground rules and assumptions include the use of *T*–0 umbilicals for cryogenic servicing activities and storage of the LH_2 at normal boiling point (saturated) conditions.

2.3.2.3 Payload Fluid System Hardware Selection Methodology. To minimize costs, strong preference was given to selecting off-the-shelf components or components with flight heritage or qualification. Additionally, minimizing the total number of different components was also preferred to reduce potential design, development, test, and engineering (DDT&E) expenses. An individual component design therefore would be used in as many locations within the system as practical.

2.3.2.4 Payload Fluid System Redundancy Rationale. The CPST payload is designated as a class D mission per NPR 8705.4, Risk Classification for NASA Payloads.[2] As such, the payload is allowed to design for single-point failure (zero failure tolerance) with mitigation of 'use of high reliability parts, additional testing, or by other means.' Additionally, 'single string and selectively redundant design approaches may be used,' but all required 'applicable NASA safety directives and standards' must be met. Given these design requirements, the CPST payload employs a zero failure tolerant architecture as its baseline, in order to reduce cost, although selective areas were addressed for failure tolerance redundancy. Specifically, two failure modes were evaluated for fault tolerance: internal leakage/failed open position and failed closed position.

2.3.2.4.1 Internal Leakage or Failed Open Position Failure Modes. Components were placed in a series configuration to mitigate the risk of excessive leakage or a failed open valve that would result in a loss of mission or pose a safety concern. Examples of components in this arrangement are helium pressurization control valves, ullage transfer valves, vent valves, the helium fill and drain valve and helium quick-disconnect configuration, the LH_2 fill and drain valve and LH_2 quick-disconnect configuration, and the BAC fill and drain valve and sealed plug configuration.

2.3.2.4.2 Failed Closed Position Failure Modes. Components were placed in a parallel configuration if a failed closed condition would result in a safety concern, such as overpressurizing the fluid system. Examples of components in this arrangement include the burst disk/relief valve and vent valve configuration. Due to budget and mass constraints, parallel component arrangements could not be made to prevent against loss of the mission if a component were to inadvertently fail closed. Conducting protoflight ground testing prior to launch plans mitigation of this risk. This test methodology would allow any latent defects in the component design or control system to be discovered and rectified prior to on-orbit use.

2.3.2.4.3 Redundancy Not Required. Several of the components in the fluid system do not require series or parallel redundancy. One example of this is the transfer pump, whereas the loss of the pump would not prevent the transfer of fluids from one tank to the other. Transfer can still be accomplished via the use of the helium pressurization system. The other example of no redundancy required is with the backpressure relief valves, which is a check valve used in a relief function. These valves are used to prevent overpressurization of liquid trapped in line segments. Since these valves do not have a credible failure mode of failing closed, there is no need to consider this mode for redundancy.

2.3.2.5 CPST Payload Fluid System Functions. As identified above, key CPST payload fluid system functions include passive thermal conditioning during storage (including TVS and propellant mixing functions), active thermal conditioning during storage (employing a refrigerated cycle using a cryocooler and BAC circuit for heat intercept), propellant transfer (between the storage and transfer propellant tanks, including line and transfer tank conditioning), propellant acquisition (to support propellant transfer and mixing operations), helium pressurization and propellant tank venting operations (as required to support propellant transfer and storage objectives), and propellant quantity gauging (under microgravity and settled acceleration environments).

2.3.2.5.1 Overview. To achieve the mission objection of pressurized transfer, a helium pressurization subsystem was incorporated into the CPST payload design. A diagram showing the baseline design is shown in figure 16.

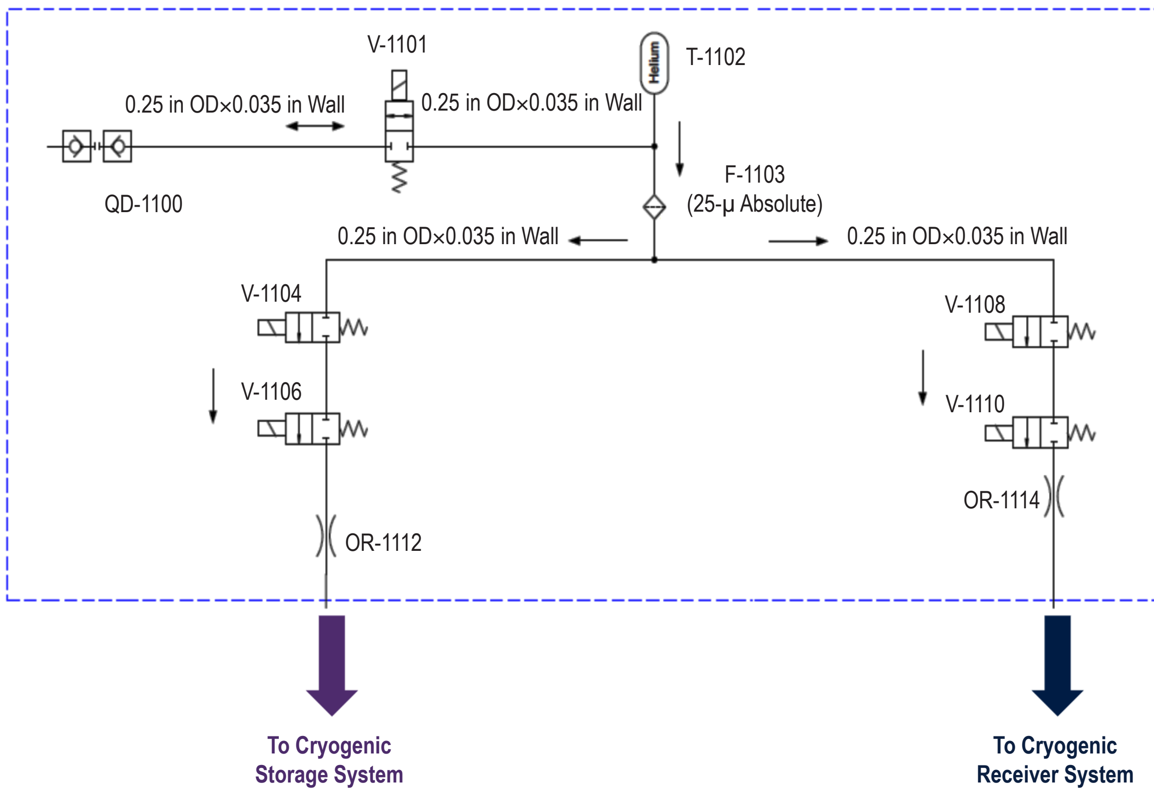


Figure 16. Helium pressurized subsystem schematic.

The helium pressurization subsystem is designed to provide pressurization capabilities to both the storage and transfer tanks via a set of helium tanks manifolded together. Each cryogenic tank is pressurized via a separate line segment containing series redundant pressurization control valves with a downstream flow control orifice. All helium being supplied to the storage and transfer tanks is flowed through a 25-μ absolute filter prior to entering the pressurization control valves. This is to minimize the potential for internal contamination that could result in excessive internal leakage of the control valves.

2.3.2.5.2 Ground Loading Operation. The helium pressurization subsystem is loaded with helium prior to integration into the payload shroud. The ground interface quick-disconnect QD-1100 is engaged with the ground half disconnect and 28 VDC is applied to the helium fill and drain valve V-1101 to allow helium to be loaded into the system until the pressure is ≈4,050 psia. Once the helium is loaded, power is removed from V-1101 and the ground half quick disconnect is demated from QD-1100.

2.3.2.5.3 Design History. There are two possible component types—a mechanical regulator or a normally closed, two-way valve—to control the flow of helium to the cryogenic tanks. Ultimately, a two-way normally closed valve (also called bang-bang valves) was chosen for the CPST pressurization control valves. This decision was based on several factors, such as allowing use of common valves in a fluid system, lower cost component, lower design risk, and more flexibility on cryogenic tank pressure.

The current design uses the same valve for both the helium pressurization valve and helium fill and drain valve. Due to the limited funding available to develop and operate the spacecraft, commonality of components was desired to reduce DDT&E costs associated with each component design.

Regulators compared to two-way valves tend to be more expensive and have more failure modes. Additionally, mechanical regulators have more limited operational limits such as flow rate and regulated outlet pressure. With two-way valves, the commanded open time can be adjusted to get the desired tank pressure. Based on analysis conducted, the time to overpressurize one of the tanks was ≈1 s, long enough for the command logic to open and close the pressurization control valve.

2.3.2.5.4 Alternate Pressurization Methods. Alternative pressurization methods were evaluated as part of the prephase A study. The method evaluated was using autogenously generated pressure to pressurize the storage and transfer tanks. There were two autogenous pressurization techniques evaluated: direct autogenous and stored autogenous:

(1) Direct autogenous—The direct autogenous approach would pump LH_2 through control valves into a heat exchanger to gasify the liquid and send it directly back into the LH_2 tank. A schematic of this concept is shown in figure 17.

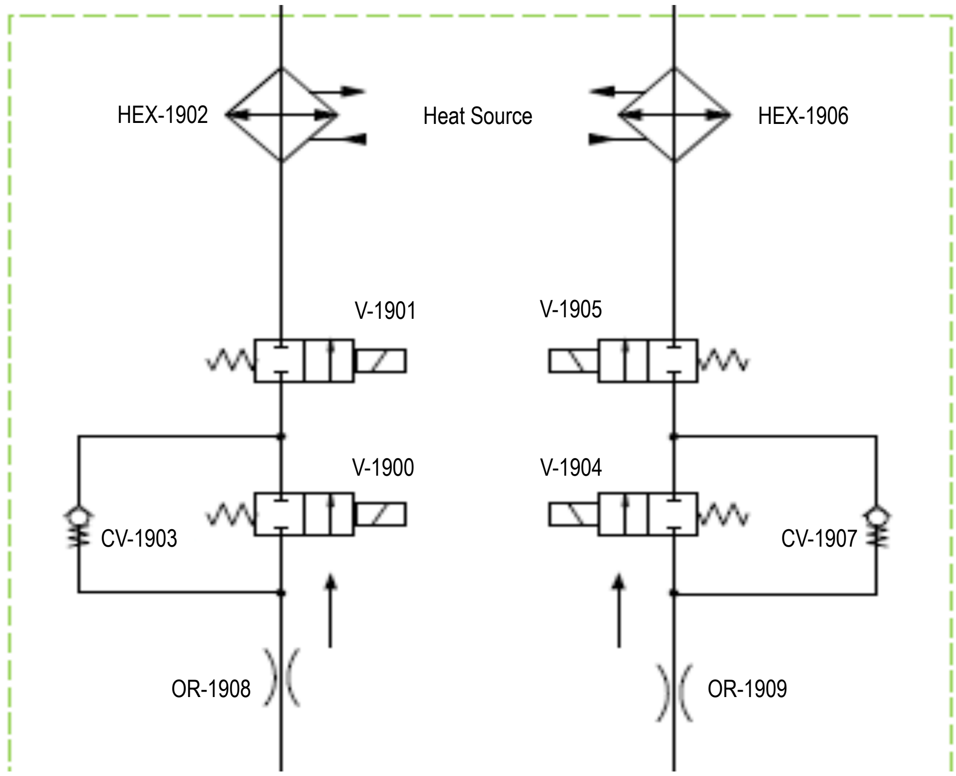


Figure 17. Direct autogenous schematic.

The disadvantages with this type of pressurization system are as follows:

• A high power usage is required to vaporize the liquid, heating, and therefore boiloff of the stored cryogen due to heat energy input used for vaporization.

• Collapse of the vapor back into liquid form, resulting in lower than designed pressurization of the cryogenic tank. Based on these concerns, the direct autogenous pressurization approach was not used for the flight vehicle design.

(2) Stored autogenous—The stored autogenous approach was an enhanced version of the direct autogenous method discussed above. In the stored autogenous method, LH_2 was pumped to a predetermined quantity into an intermediate storage tank. This intermediate tank, which is isolated from the cryogenic tank by control valves, is then heated to vaporize the liquid stored within the tank. Due to the long times between transfers, the heat needed to boil off the liquid is generated by the environment which eliminates the need for additional power capabilities on the spacecraft. After the liquid has vaporized, the control valves would pulse open, just like the helium pressurization valves, to regulate cryogenic tank pressure. An advantage of this approach is that the LH_2 used to pressurize is essentially recycled once it collapses back to liquid temperatures. A schematic of the stored autogenous approach is shown in figure 18. The stored autogenous method was not used due to funding and payload mass limitations for the vehicle.

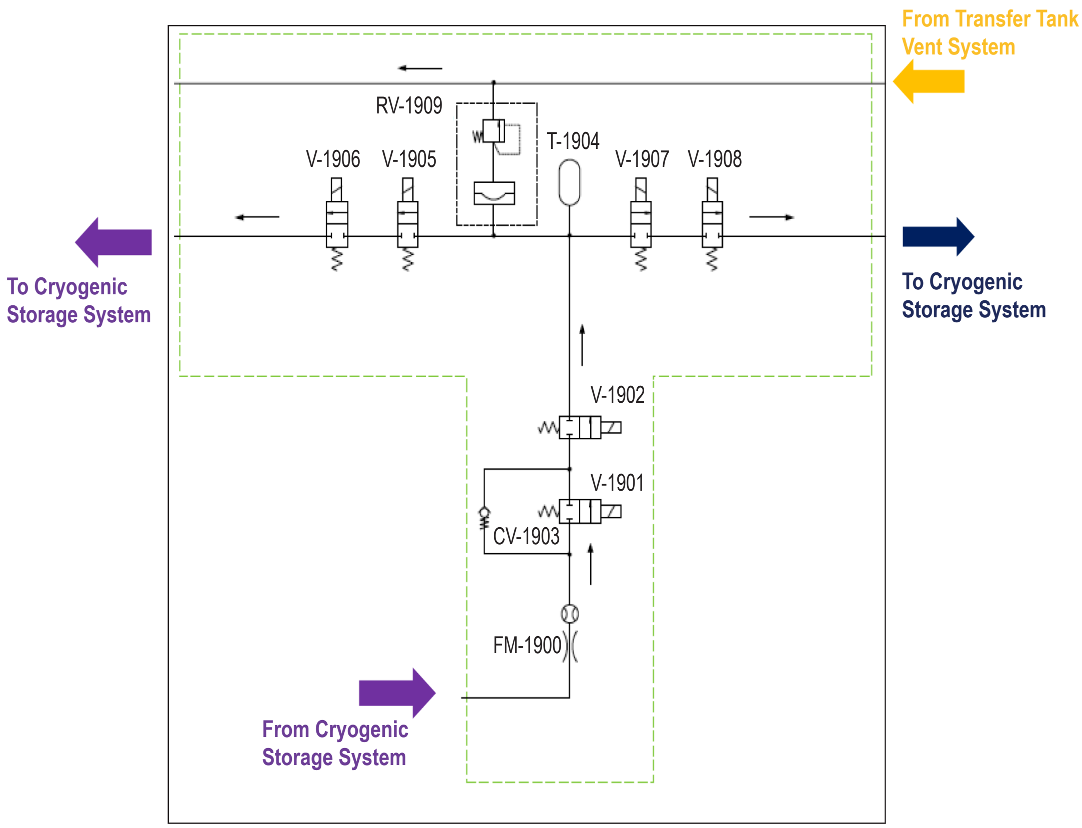


Figure 18. Stored autogenous schematic.

2.3.2.5.5 Venting—Overview. To ensure there is no tank rupture due to overpressurization, a vent subsystem was incorporated into the CPST payload design. A diagram showing the baseline design is shown in figure 19.

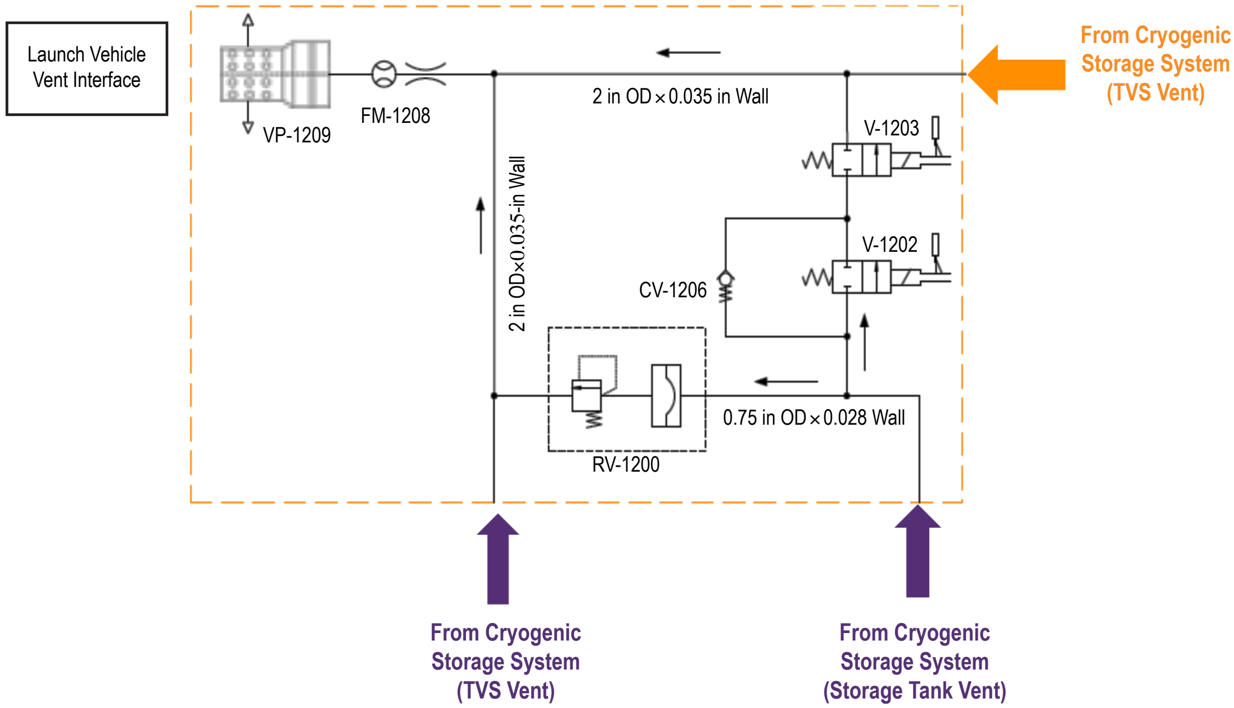


Figure 19. Vent subsystem schematic—storage tank side shown only.

The vent subsystem is designed to manifold vents from both TVS and cryogenic tanks into a single vent out of the vehicle. Controlled venting on the cryogenic tanks can be accomplished by commanding the vent valves V-1202 and V-1203 to the open position (for the storage tank). A check valve, CV-1206, prevents the cavity between the two valves from overpressurizing if there is trapped cryogenic liquid between the two valves when the vent valves are in the closed position.

Passive venting of the cryogenic tanks is accomplished using a burst disc/relief valve combination per cryogenic tank. The burst disc minimizes the internal leakage that can occur through a closed relief valve prior to the disc being ruptured. The relief valve and burst disc are set to open lower than the rated pressure of the cryogenic tanks.

Since the TVS valve vents also are connected with the tank vents, consideration was made to ensure that any backpressure generated in the vent lines was included in the TVS valve design parameters. A pressure of 5 psia was determined to be in the vent lines during a cryogenic tank venting procedure.

2.3.2.5.6 Vent Probe Concept. One of the challenges of venting in space is not generating undesired torques on the spacecraft. The concept selected to minimize these torques was what has been called the vent probe. An external view, looking at the side of the vent probe, is shown in figure 20.

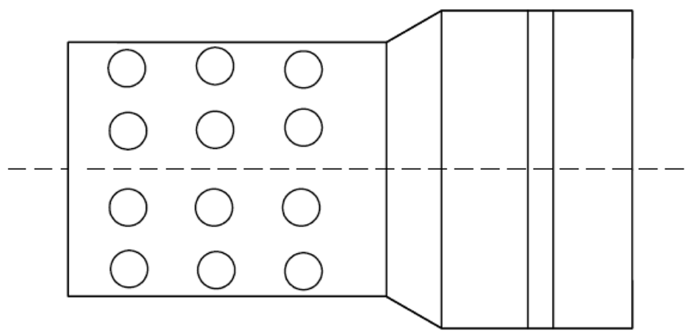


Figure 20. Vent probe—external view.

An additional requirement of the vent probe is to duct the vent fluid through a vent line when the CPST vehicle is attached to the launch vehicle and then be a nonpropulsive vent after the CPST separates from the launch vehicle. To accomplish both of these vent requirements, the vent probe is designed to seal in a sleeve fixed to the launch vehicle to route the vent fluid through the vent line on the launch vehicle. To ensure a nonpropulsive vent path, the flow within the vent probe will be turned 180 deg to eliminate most of the fluid momentum before it is expelled from the CPST vehicle. Additionally, there is a large volume internal to the probe to maintain a low fluid exit velocity. Figure 21 shows the vent probe in the configuration when integrated to the launch vehicle. Figure 22 shows the vent probe configuration when used as a nonpropulsive vent.

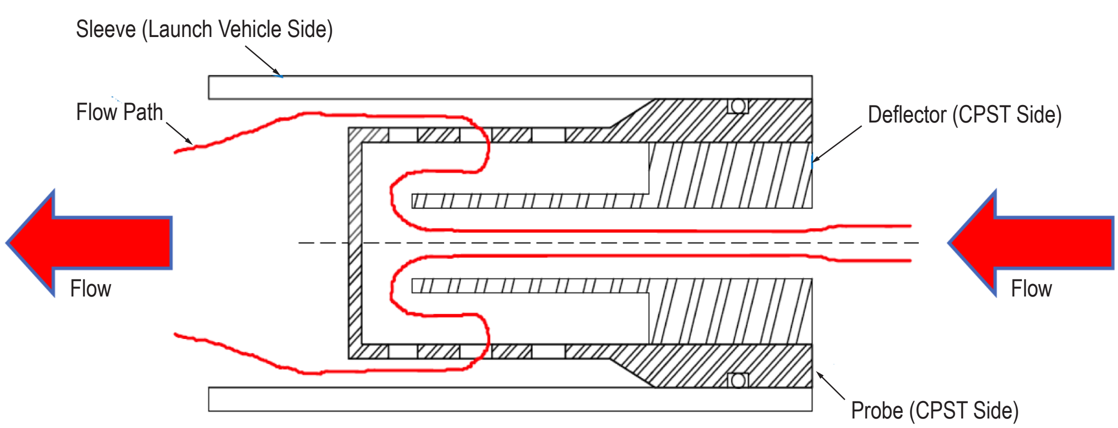


Figure 21. Vent probe (payload on launch vehicle) cross section.

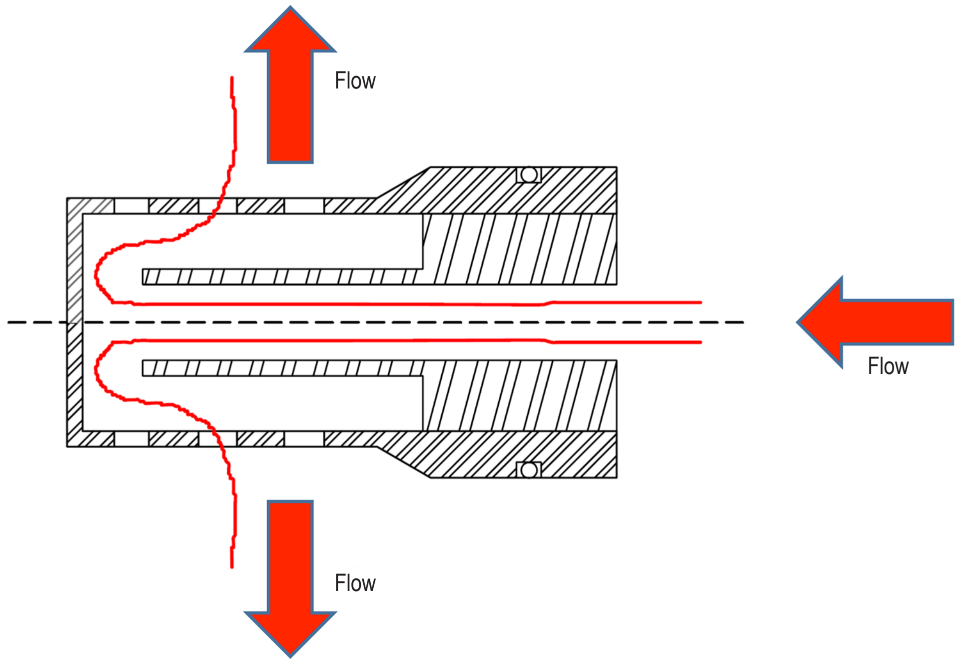


Figure 22. Vent probe in nonpropulsive vent use—shown in disengaged position.

2.3.2.5.7 Payload Fluid System Liquid Hydrogen Storage—Overview. The LH_2 storage subsystem contains both the LH_2 storage tank and ground fill and drain connection. The LH_2 will be maintained in the storage tank during flight to verify minimal boiloff, accomplished by both passive and active storage techniques.

The burst disc/relief valve (RV-1200), vent valves (V-1202 and V-1203), pressurization check valve (CV-1413), and ullage transfer valve (V-1414) are close coupled with the top of the storage tank to minimize propellant migration into the lines.

The TVS valve (V-1404), LH_2 fill and drain valve (V-1402), and circulation isolation valve (V-1411) are close coupled with the bottom of the storage tank to also minimize propellant migration into the lines. To reduce the heat leak into the propellants, the number of storage tank penetrations was minimized. Separate propellant transfer lines were required due to the need to pull liquid from the liquid acquisition device (LAD) and flow liquid back into the storage tank outside the LAD. A schematic of the LH_2 storage subsystem is shown in figure 23.

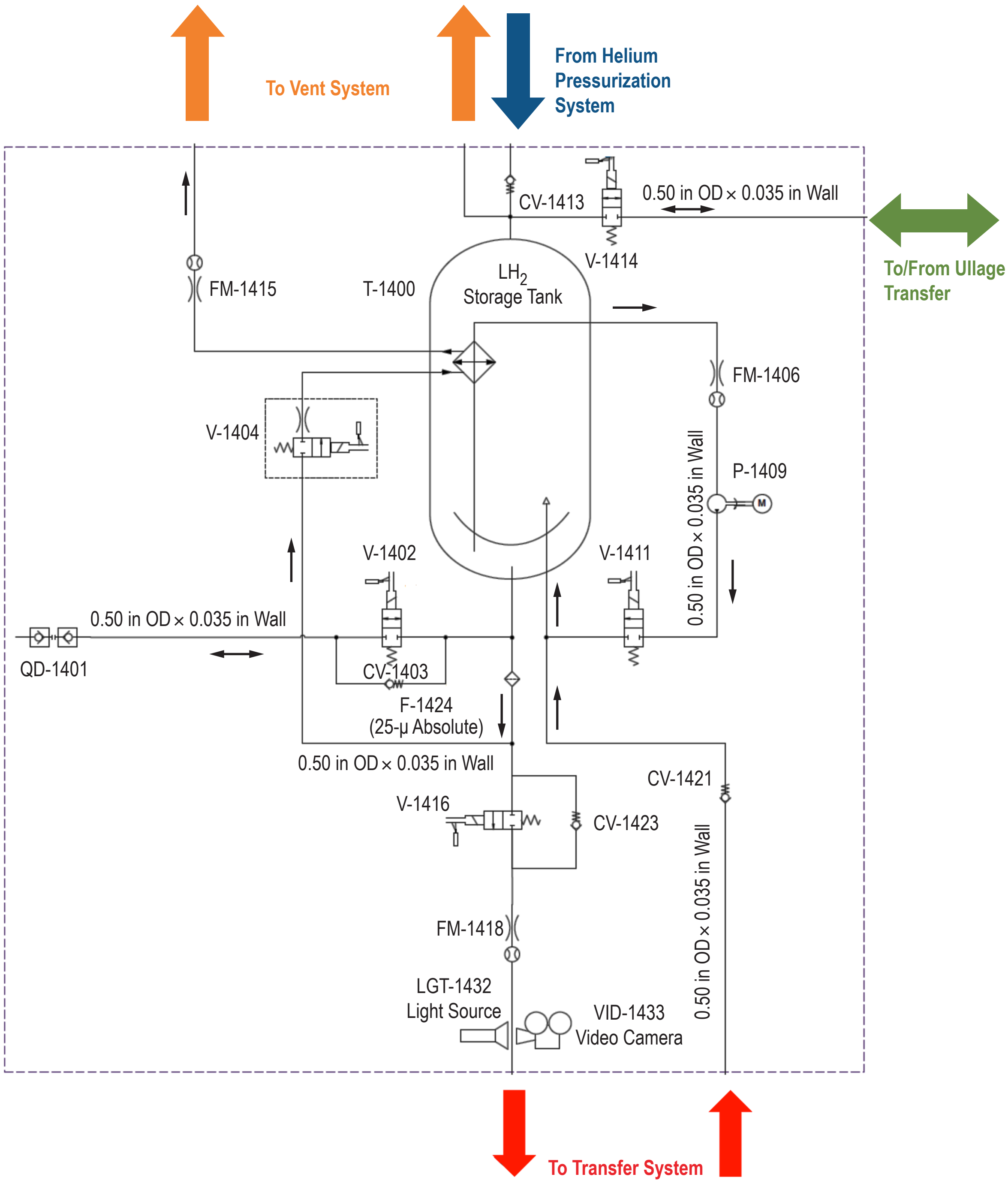


Figure 23. LH_2 storage subsystem schematic.

2.3.2.5.8 Ground Loading Operation. The LH_2 storage subsystem is loaded with LH_2 using a *T*–0 umbilical on the launch pad. The ground interface quick disconnect (QD-1401) is engaged with the ground half disconnect and 28 VDC is applied to the LH_2 fill and drain valve (V-1402) to allow LH_2 to be loaded into the system until the pressure is ≈20 psia. Once the LH_2 is loaded, power is removed from V-1402 and the ground half quick disconnect is demated from QD-1401. LH_2 is not loaded into the LH_2 receiver tank while on the ground.

2.3.2.5.9 Liquid Hydrogen Storage Tank Assembly. Contained within the LH_2 storage system is the storage tank, which is designed to be a modular system containing all tank valving, LAD, helium gas diffuser, internal mass gauging sensors, and thermal controls. A diagram of the components included in the LH_2 tank assembly can be seen in figure 24.

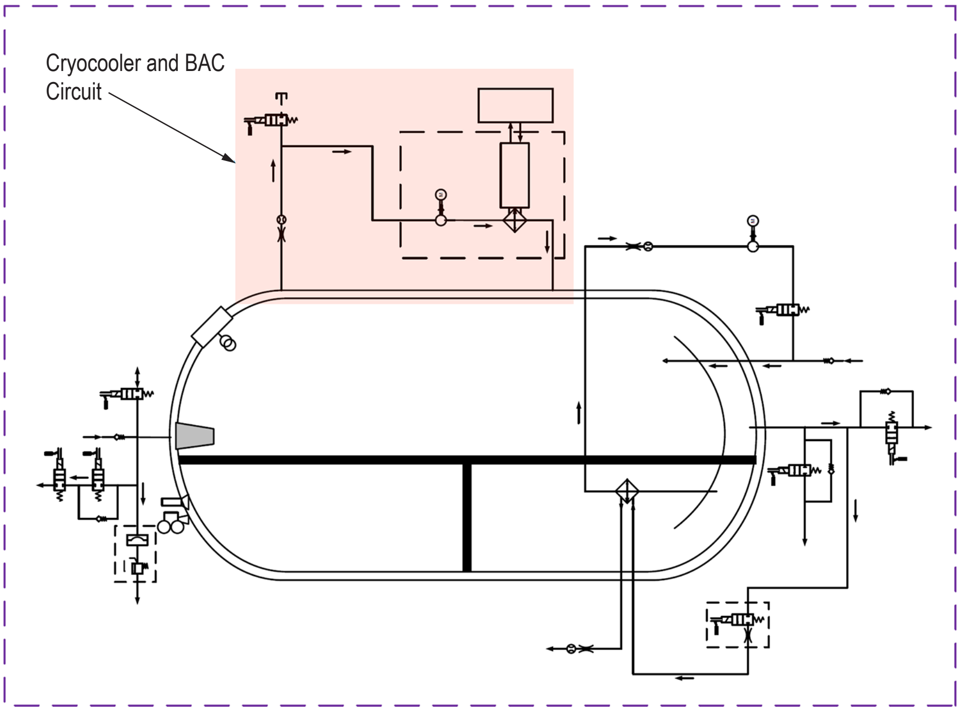


Figure 24. LH_2 storage tank assembly diagram.

2.3.2.5.10 Payload Passive Thermal System. To affect reduced propellant boiloff demonstrations, passive and active thermal control design features are incorporated, including a TVS, active propellant mixing, MLI, low heat leakage structural support struts, and reduced fluid penetrations for the main storage propellant tank. The TVS utilizes a Joule-Thomson (J-T) expansion device for vented gas cooling which, in conjunction with a heat exchanger, provides cooling of the stored LH_2 and pressure control. Propellant is acquired for the TVS function under settled and unsettled acceleration conditions via surface tension LADs installed inside the tanks, which also provide propellant for planned propellant transfer operations. Penetrations into the propellant tanks are isolated with close-coupled components to preclude propellant migrations into the lines. These include the vent system burst disk/relief valve and vent valves, the pressurization check valve, the ullage transfer valve, the TVS control valve, the LH_2 fill and drain valve, the tank isolation valve, the transfer check valve, and the circulation isolation valve. Composite/metallic hybrid struts are used to thermally isolate while structurally supporting both the storage and transfer propellant tanks. To provide acceptable conditioning of the storage tank during the ground and ascent phases, the tank is encased in a layer of spray-on foam insulation (SOFI), and the MLI is actively purged with GN_2 (to prevent moisture condensation within the MLI). Only the storage propellant tank is loaded with LH_2 for the ground and ascent phases of the mission.

2.3.2.5.11 Payload Active Thermal System. The CPST payload fluid system also employs a cryocooler-BAC network architecture to actively refrigerate the external surface area of the storage propellant tank. This intercepts heat entering into this tank through the MLI over the surface area of the tank, reducing hydrogen propellant boiloff. Due to the low technology readiness of a 20 ºF cryocooler system, the CPST payload employs a state-of-the-art 90 K cryocooler architecture. Because of this, the payload realizes an RBO design implementation for the LH_2, as opposed to a zero boiloff (ZBO) design. Cryocooler options include pulse tube, Stirling, and reverse turbo-Brayton cooler designs. The pulse tube and Stirling (regenerative) coolers require a gas circulator to supply the BAC network. The reverse turbo-Brayton (recuperative) cooler can utilize the inherent circulation loop; therefore, a separate circulator is not required. Vibration of any of these cryocooler systems is minimal. The cryocooler refrigerant, either helium or neon at 90 K, is circulated through an array of small-diameter cooling tubes embedded within the MLI on the storage tanks. These tubes, mounted to a thin aluminum foil shield, which encapsulates the total surface area of the tank (as mounted inside the MLI), comprise the BAC. The BAC is mounted within the MLI layers such that it effectively intercepts heat conducted through the MLI (at a position predicted to be approximately at the 90 K temperature of the refrigerant). Options are under consideration to additional heat interception by active cooling of the structural support struts and the fluid and instrumentation penetrations via straps to the BAC.

2.3.2.5.12 Payload Fluid System Liquid Hydrogen Transfer. To move LH_2 from the storage tank to the transfer tank, there are connecting lines, valves, and pumps to accomplish this transfer. Due to the need to use separate exit and entry connections to the tanks and minimize design complexity, unidirectional flow paths were used for the transfer subsystem. A bi-directional configuration would have been possible, but with a more complicated valve arrangement.

It can be seen in the LH_2 transfer subsystem schematic, shown in figure 25, that there are parallel flow paths per flow direction. These parallel paths are needed to demonstrate both pressurized and pump-fed transfer modes.

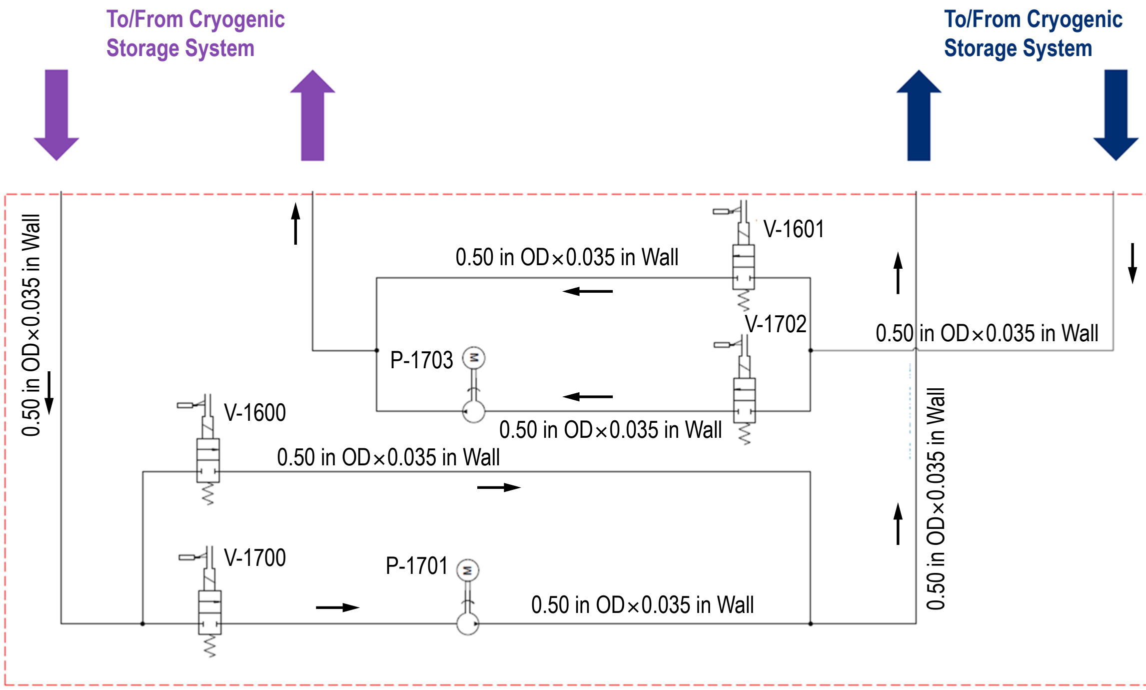


Figure 25. LH_2 transfer subsystem schematic.

2.3.2.5.13 Alternative Configurations. Early in the design study the vehicle contained an automated fluid coupling (AFC) that would have been used to demonstrate mating and demating of a coupling in space, then transfer propellants through the coupling with zero leakage. Incorporating this coupling in the current configuration would have resulted in a portion of the transfer system with bi-directional flow, since adding a second AFC may not have been desired for cost reasons. Adding a bi-directional flow portion would require many more two-way valves to route the fluid to the proper tank.

Therefore, an alternative configuration was created that minimized the total number of components to accomplish propellant transfer. The alternative transfer system, which can be seen in figure 26, has three-way valves to transfer fluid either by pressurized, pump-fed, or across AFC methods. The valve configuration allowed for bi-directional flow in almost the entire system, except across the transfer pump, which required unidirectional flow to operate.

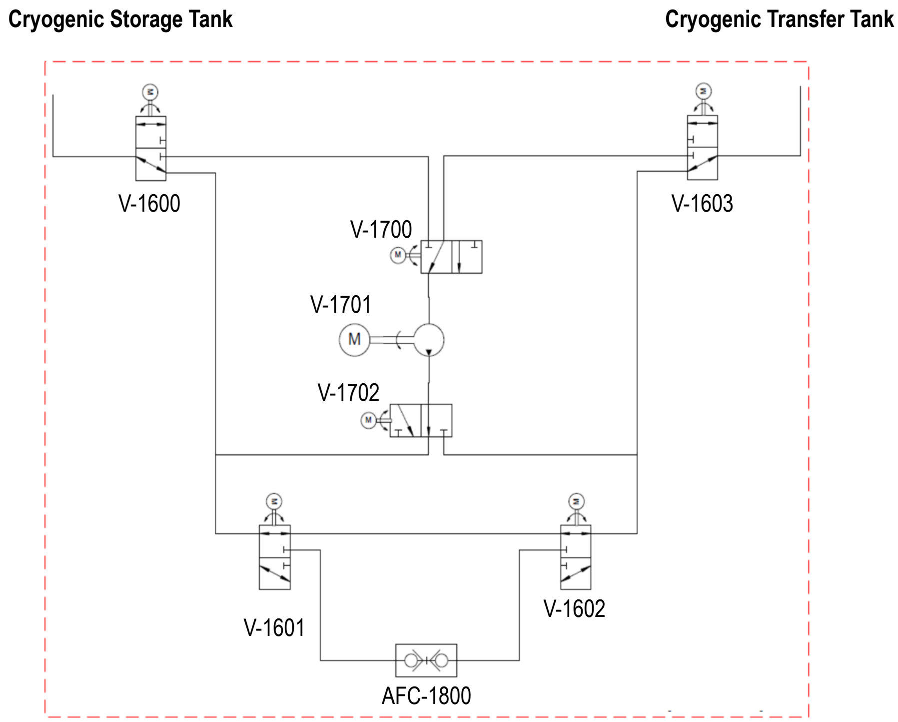


Figure 26. Alternative LH_2 transfer system schematic—cryogenic transfer subsystem (three-way valve configuration; simplified rep).

This system was not chosen for the baseline concept due to the strong desire to use common valves throughout the vehicle to minimize DDT&E costs. The use of this three-way valve would be a higher risk than a standard two-way valve due to the need to create a new component design for this program.

2.3.2.5.14 Payload Instrumentation. Instrumentation is robustly provided in the CPST payload to assure thorough measurement of thermal conditioning, transfer, and quantity gauging aspects of the flight experiment and employed technologies. Redundancy is provided for the critical sensors needed to prove technology performance advancement in order to validate predictive analytical models and validate mission objectives. Specifically, on both storage and transfer tanks, the following instrumentation is provided:

- 90 external temperature measurements.
- 16 internal temperature measurements, also used for liquid level sensing, silicon diode devices.
- Four TVS JT/mixing internal temperature measurements.
- 18 external and 6 internal cryocooler/BAC temperature measurements.
- Two tank pressure measurements.
- Settled and unsettled mass gauging (e.g., CryoTracker®, radio frequency mass gauging (RFMG), and possibly capacitance probe).
- Video on tank internal and outflow legs for propellant positioning and flow insight.

On the pressurization and transfer systems, the following instrumentation is provided:

- 42 external temperature measurements.
- Flow rate measurements on all propellant flow paths between tanks and through vent legs.
- Operation measurements on all valves and pumps.
- Up to 20 leak detection devices (electronic devices with capability for direct hydrogen presence detection).
- Redundant tri-axial accelerometer measurements for settling/unsettled acceleration rates.

The CPST payload design employs mass gauging techniques for settled and unsettled propellant orientations. Settled mass gauging includes a CryoTracker device using silicon diode devices to sense the liquid level within the tanks, with the option to include full tank range capacitance probe as backup. Unsettled mass gauging employs an RFMG, which detects tank frequency mode responses and matches characteristics to analytically predicted responses to determine propellant mass levels (under unsettled and settled acceleration states). For additional data for correlation, mass flow rates will be measured on all propellant transfer and vent legs, with positioning of flow rates for best results regarding thermal conditioning of propellant passing through these devices.

2.3.2.6 Payload Supporting Analyses. Several conceptual sizing analyses were performed for the CPST payload fluids system. These include pressure budget sizing to support pressure level definition and component sizing activities, and helium consumable mass sizing to support high-pressure helium bottle sizing and envelope definition. Additionally, a conceptual layout assessment for tank penetrations and mounted components is provided (in part, supporting the rationale for the CPST payload propellant storage tank diameter).

2.3.2.6.1 Fluid System Pressure Budget—Introduction. The CPST technology demonstration will transfer LH_2 from the storage tank to the transfer tank and back using pressure and pump-fed transfers. The pressure budget for each type of transfer is assumed to be the same flowing from or to the storage tank. In order to mix the propellant in the tank, there is also a recirculation loop on the LH_2 storage tank. A preliminary pressure budget was developed to understand design limitations required to support the transfer and mixing functions.

2.3.2.6.2 Propellant Transfer System. Two sets of transfer lines run from the storage tank to the transfer tank as shown in figure 27. This allows the propellant to be pulled from the LAD in one tank and distributed outside the LAD in the other tank. All of the transfer lines are sized at 0.5 in (1.27 cm) outer diameter (OD) with a wall thickness of 0.035 in (0.0889 cm). Each transfer line has a path where the transfer can utilize the pump or bypass the pump. The line length was conservatively estimated at 15 ft (4.572 m). In each line, there are three isolation valves. The first isolation valve is close coupled with the tank while the other two isolation valves allow flow through either the open line for pressure-fed transfer or the line with the pump for pump-fed transfer. For a given transfer, the propellant will only flow through two of the three isolation valves. The cryogenic isolation valves are assumed to have an equivalent sharp edge orifice diameter (ESEOD) of 0.278 in (0.706 cm). The ESEOD was provided by the MSFC components group, ER33, and is representative of a common off-the-shelf cryogenic valve. There is a filter at each tank outlet and

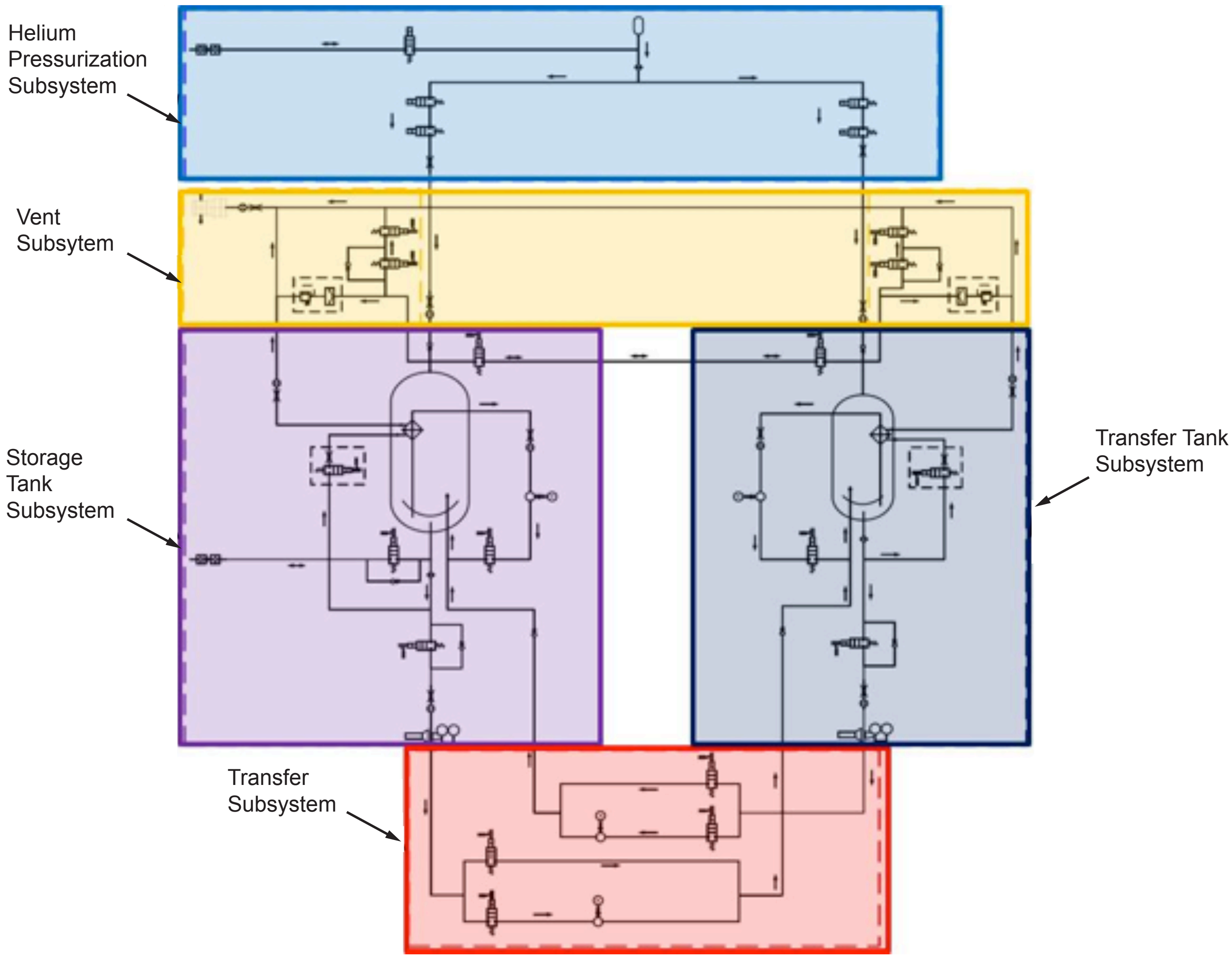


Figure 27. CPST fluid system schematic.

a check valve at each tank inlet. Losses for other components and lines were based on nominal resistance coefficients. These loss coefficients will need to be updated as better component and system information becomes available.

2.3.2.6.3 Propellant Transfer Pressure Budget. The propellant will be transferred by two means—pressure fed or pump fed. For each type of transfer, the expected steady-state flow rate is about 78 lbm/hr (0.0098 kg/s). This flow rate equates to a velocity of ≈5 ft/s through the transfer line. During all transfers, the propellant is assumed to be saturated at 20 psia (137.9 kPa) which corresponds to a temperature of –421 °F (21.5 K).

The pressure-fed transfer pressure budget is shown in table 21. The propellant tank is assumed to be pressurized to 25 psia (172.4 kPa) with helium pressurant. The total pressure drop through the transfer line and transfer line components is ≈0.5 psid. The minimum propellant tank maximum design pressure (MDP) is 40 psia (275.8 kPa) for pressurized transfer or 15 psi above the propellant tank ullage pressure.

Table 21. Pressure-fed system pressure budget.

Pressurized Transfer	Hydrogen
Mass flow rate	77.9 lbm/hr
Velocity	4.9 ft/s
Estimated transfer line length	15 ft
Storage tank ullage pressure	25 psia
Line ΔP (tank outlet/inlet/bends/ fittings/check valve)	–0.2 psid
Valve ΔP (two valves)	–0.3 psid
Transfer tank inlet line pressure	24.5 psia
Propellant tank MEOP/MDP	40 psia

The pump-fed transfer pressure budget is shown in table 22. The propellant tank is assumed to be at 20 psia (137.9 kPa) with saturated propellants in the tank. The total pressure loss through the transfer line and components is estimated to be 0.5 psid. The pressure increase required by the pump was calculated as the pressure loss in the system at the expected flow rate with a 30% margin for growth and flexibility. The minimum propellant tank MDP for pump-fed transfer is 35 psia (241.3 kPa).

Table 22. Pump-fed system pressure budget.

Pump Transfer	Hydrogen
Mass flow rate	77.9 lbm/hr
Velocity	4.9 ft/s
Estimated transfer line length	15 ft
Storage tank ullage pressure	25.0 psia
Line ΔP (tank outlet/inlet/bends/ fittings/check valve)	–0.2 psid
Valve ΔP (two valves)	–0.3 psid
Pump ΔP (P2-P1)	
Transfer tank inlet line pressure	20.2 psia
Propellant tank MEOP/MDP	35 psia

2.3.2.6.4 Propellant Recirculation System. As shown in figure 27, the propellant tank recirculation system pulls propellant out of the tank LAD with the recirculation pump and sprays back into the tank to mix the propellant and ullage. The recirculation line is 0.5 in (1.27 cm) OD with a wall thickness of 0.035 in (0.0889 cm). The line length was conservatively estimated to be 15 ft (4.572 m). There is a single isolation valve with an estimated ESEOD of 0.278 in (0.706 cm) (representative of a common off-the-shelf cryogenic valve). Other component loss coefficients were estimated based on nominal resistance coefficients. These coefficients will need to be updated as component and system information becomes available.

2.3.2.6.5 Propellant Recirculation Pressure Budget. The tank recirculation loop will be driven with a pump. The expected steady-state flow rate is about 244 lbm/hr (0.0307 kg/s). During the recirculation, it is assumed that the propellant will be saturated at 20 psia (137.9 kPa), which corresponds to a temperature of –421 °F (21.5 K).

The TVS pressure budget is shown in table 23. The total pressure through the transfer line and components is estimated to be 2.9 psid. For the pump, the change in total pressure was calculated as the pressure loss in the system at the expected flow rate with a 30% margin for growth and flexibility.

Table 23. TVS system pressure budget.

TVS Pump Mixing	Hydrogen
Mass flow rate	243.8 lbm/hr
Velocity	15.5 ft/s
Estimated transfer line length	15 ft
Storage tank ullage pressure	20 psia
Line ΔP (tank outlet/inlet/bends/ fittings)	–1.4 psid
Valve ΔP (one valves)	–1.5 psid
Pump ΔP (P2-P1)	3.8 psid
TVS mixing flow line outlet pressure	20.9 psia

The system pressure budget needs to be reevaluated as the system design matures. The final system MDP was set to 50 psia (344.7 kPa) to allow for system changes.

2.3.2.6.6 Helium Tank Sizing Analysis—Introduction. Helium will be used as the pressurant for the LH_2 pressure-fed transfer demonstrations on the CPST technology demonstration. The quantity of helium required depends on the ullage volume of the propellant tanks during the transfer, the required pressure for the transfer, and the heat transfer between the helium pressurant and the fluid in the propellant tank.

2.3.2.6.7 Propellant Transfer. There are two transfer series, each consisting of settled pressure-fed transfer, settled pump-fed transfer, unsettled pressure-fed transfer, and unsettled pump-fed transfer. For each transfer, the propellant is sent to the transfer tank and then returned to the storage tank. The maximum ullage pressure driving the transfer is assumed to be 30 psia. For pressure-fed transfers, the storage tank will be pressurized and maintained until the propellant transfer is complete, filling the transfer tank ≈90%. The storage tank will then be vented overboard to a nominal pressure. The transfer tank will then be pressurized and maintained until the propellant is transferred back to the storage tank. The transfer tank will then be vented overboard. At no time during this process will any of the pressurant be recovered. No pressurant will be used during the pump-fed transfers.

Figure 28 shows the propellant inventory during the mission timeline based on thermal analysis. Figure 29 shows the corresponding storage tank ullage. This ullage provides the storage tank volume that must be pressurized during transfers to the transfer tank. The total transfer tank volume is required to be pressurized to return propellant to the storage tank.

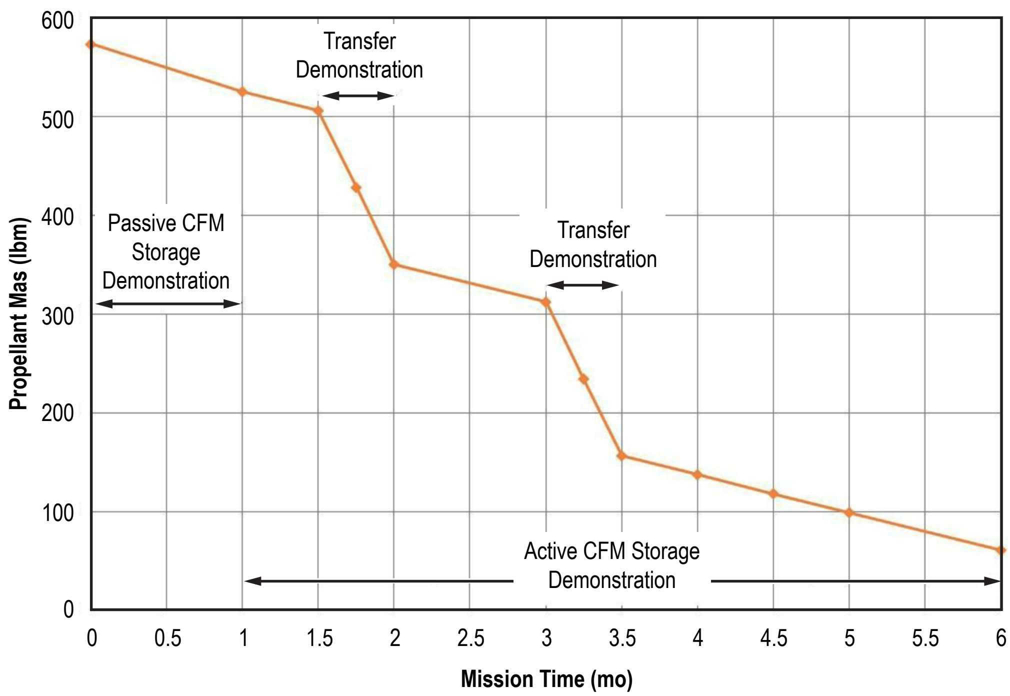


Figure 28. Assumed propellant inventory over the mission timeline.

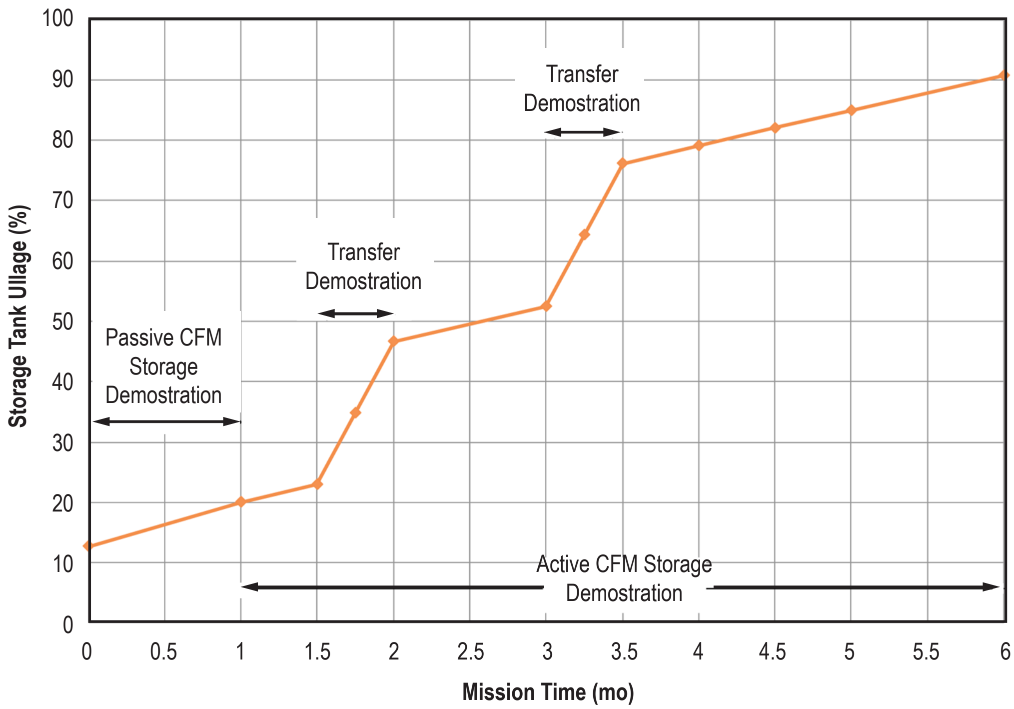


Figure 29. Assumed storage tank ullage percentage over the mission timeline.

During pressurization, the helium is assumed to cool to the liquid temperature. For this analysis, the LH_2 is assumed to be saturated at 20 psia. The cooling of the pressurant is a conservative assumption for determining helium requirements, although it may not be the case for larger ullages. Even though the helium is depositing energy into the propellant tank, the distribution of this energy is unknown and therefore cannot be assumed to assist in tank pressurization. The calculated pressurant required to compete the two transfer series with the assumptions made is ≈53.5 lbm of useable pressurant.

Helium pressurant bottles—The pressurant bottles are composite overwrap pressurant vessels (COPVs) with a maximum expected operating pressure (MEOP) of 4,500 psia. The tanks are assumed to be loaded at 4,050 psia (90% of MEOP) to allow for the relief valve variability during loading. The nominal ground-loading temperature is 70 °F.

In flight, the pressurant tanks will be held at a storage temperature of –20 °F. This temperature was selected due to the soft goods heritage temperature capability in the bang-bang pressurization valves. Heaters on the COPV will be used to keep the pressurant at the storage temperature. This temperature corresponds to an initial pressure of 3,370 psia in the COPV. During pressurant usage, the helium temperature can fall below the storage temperature (–20 °F) but the assumption is that there is adequate time to thermally recover to the storage temperature prior to the next transfer.

The system model is a simplistic, conservative spreadsheet that assumes the fluids are at bulk conditions. The storage tank ullage is calculated from the tank volume and the propellant inventory for each transfer. The change in ullage volume is then calculated based on the mass required for chill-in of the feed system, transfer tank, and the quantity of transferred propellant. This provides the ullage volume that must be pressurized. For the transfer back to the storage tank, it is conservatively assumed that the entire volume of the transfer tank was required to be pressurized. Knowing the pressure, temperature, and volume required to be pressurized, the helium mass required for pressurization was calculated. This mass is then removed from the helium COPV. A higher fidelity model is required to adequately predict propellant tank ullage conditions, which may allow a reduction of the required helium.

Figure 30 shows the helium COPV tank pressure over the mission timeline. No helium leakage was assumed. The minimum allowable pressure in the COPV is assumed to be 250 psia. A total volume required to provide the necessary useable helium is 24 ft^3 with a total helium mass loaded at 60.5 lbm. This volume equates to a 3.6-ft-diameter sphere.

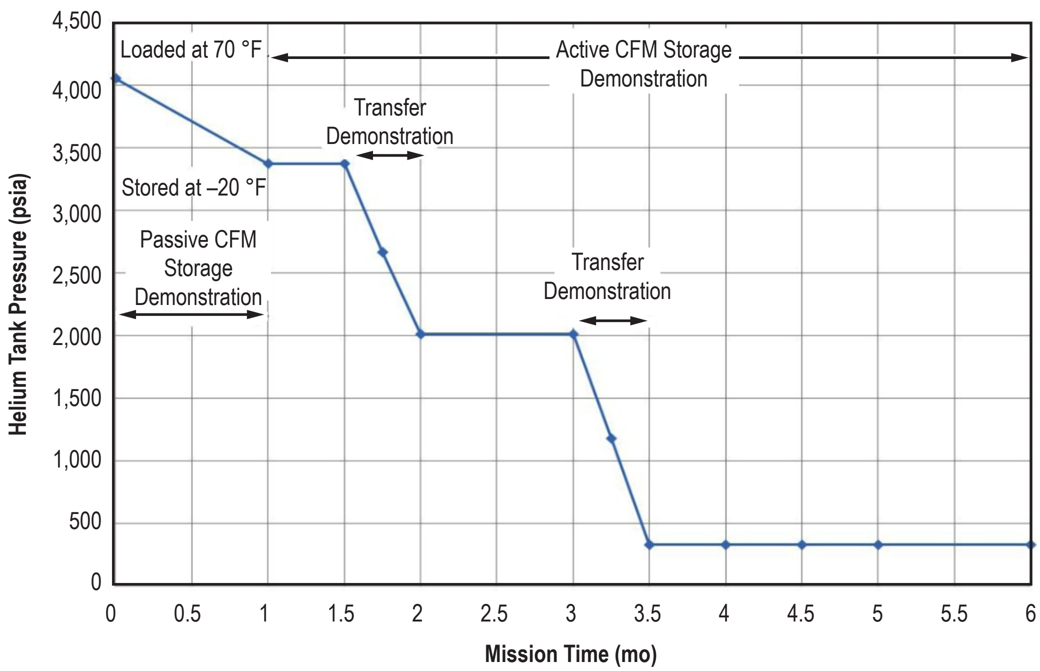


Figure 30. Helium pressure over the mission timeline.

Two tank configurations were proposed based on vendor studies; one has six COPVs while the other has eight. The information for the two configurations is in table 24. Both configurations have the same total volume—the difference is in the packaging and weight.

Table 24. Configuration options.

Number of Tanks	6	8
Total length to end to end (external)	36.04 in 0.915 in	28.15 in 0.715 in
Outer diameter (external)	17.7 in 0.450 m	17.7 in 0.450 m
Total internal volume (per tank)	6,912 in^3 0.113 m^3	5,184 in^3 0.085 m^3
Mass (per tank)	46 lbm 20.9 kg	35 lbm 15.9 kg

2.3.2.6.8 Payload Storage Tank Component Layout Assessment. An assessment was performed to evaluate packaging of planned components on the end domes of and internal to the propellant storage tank in order to validate the reasonableness of the CPST payload storage tank diameter. This conceptual packaging assessment was made under the constraint of minimal maturity associated with these components and also the assumption of providing access to these components for repair or replacement without impacting the assembled payload system. To achieve this latter assumption, the conceptual configuration installed the various hardware items on cover plates for the top and bottom domes, which then are installed on these domes. Additionally, some of the identified hardware items are assumed to be removable without the removal of the integrated cover plate assemblies. Also, pending future stress analysis maturation, the guideline was assumed to minimize the discontinuity angle between the cover and the tank dome, ideally to an angle no more than 10 deg.

To support these installation operations, there is a 2.5-in, 360-deg clearance around all ports/hardware installation covers for structural integrity, manufacturability, and operability. Other significant assumptions include:

- A 1-in area around the manhole cover for a flat sealing surface.
- A 4-in camera viewing port.
- A 2-in camera light source port.
- The diffuser is preinstalled on the cover.
- Temperature/wet-dry liquid sensor racks are installed through the 2-in port on the cover.
- The TVS heat exchanger is preinstalled on the cover.
- The spray bar and axial jet mixing devices are installed through the 2-in port.
- The LAD has a 6-in sump/basket and 2-in gallery channels.
- The mixing and transfer pumps are mounted on the outside of the tank cover.
- The RFMG requires at least a 2.75-in port so a 3-in port is provided.

The capacitance probe is not included in this layout assessment. Also, it is assumed that the BAC tubing manifold does not interface with the covers.

The conceptual cover to dome interface, as a function of cover size and using the CPST payload storage tank diameter of 60 in (5 ft) and elliptical end dome geometry, is shown in figure 31.

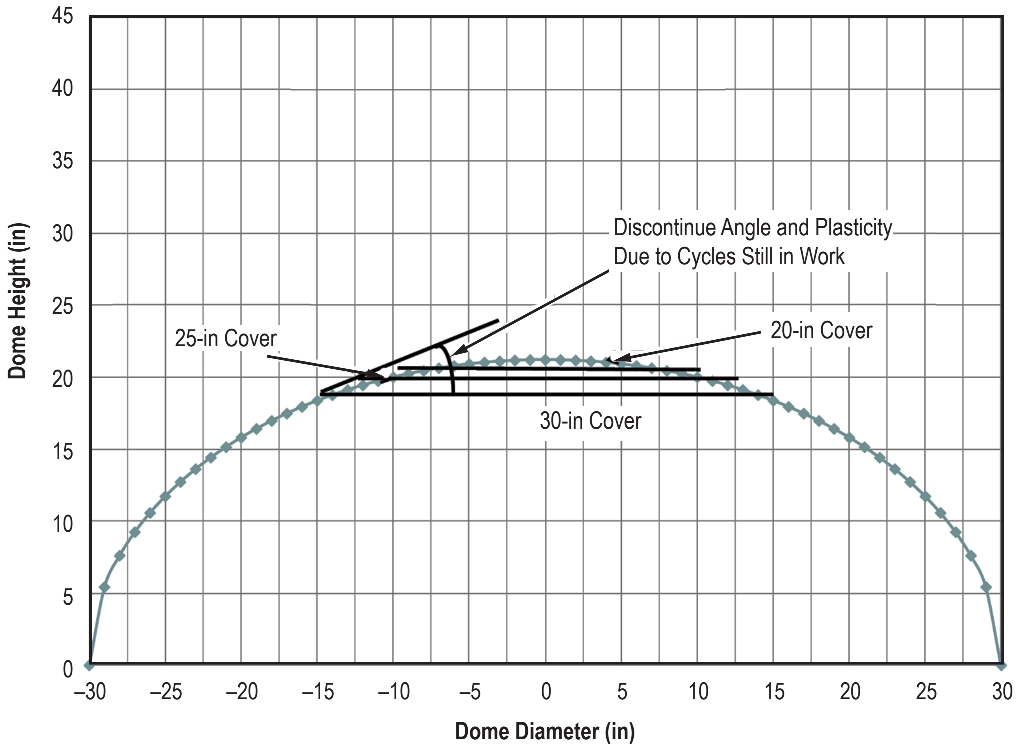


Figure 31. CPST payload 5-ft storage tank end dome—cover geometry.

The list of conceptual components to mount to the end domes of the storage tank is provided in table 25. Conceptual layouts of these components are shown for the top and bottom dome cover plates in figures 32 and 33.

Table 25. CPST payload storage tank end dome and internal components.

No.	Component	Top Dome	Bottom Dome
1	LAD		×
2	Mixing pump	×	
3	Transfer pump		×
4	Diffuser	×	
5	Temp rack and wet/dry rack	×	
6	Camera view port	×	
7	Camera light port		×
8	Spraybar and axial jet		×
9	Heat exchanger		×
10	RFMG	×	

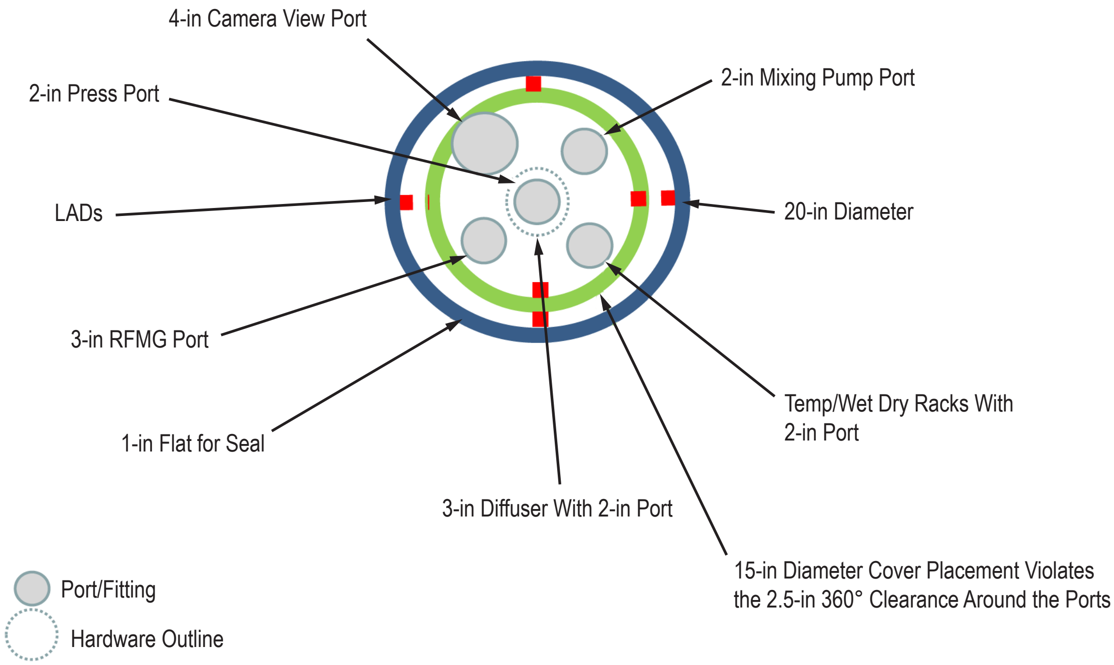


Figure 32. CPST payload tank top dome cover conceptual layout.

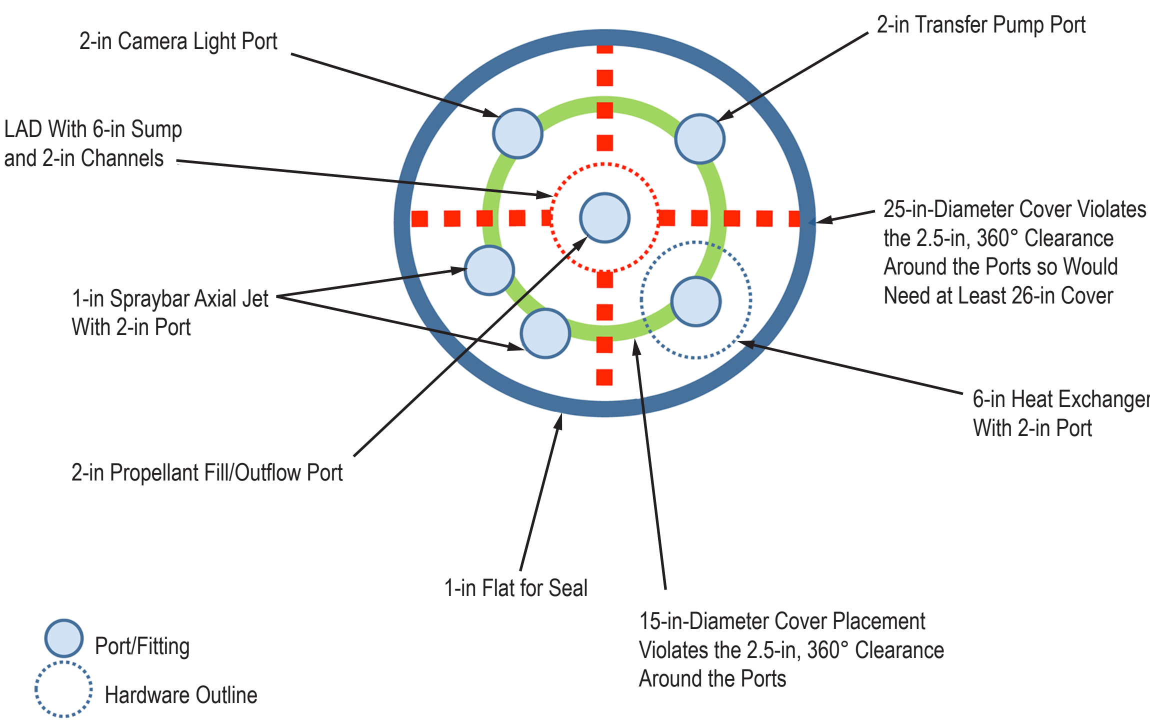


Figure 33. CPST payload storage tank bottom dome cover layout.

Based on the updated guidelines and assumptions, component list, and if the component layout is the deciding factor, the storage tank would need to have a diameter of at least 60 in. This allows for the current component list to be accommodated in a 25-in cover. The discontinuity angle and plasticity requires further evaluation for structural integrity; it is noted that the 25-in cover and 60-in-diameter tank does not meet the 10-deg discontinuity goal.

2.3.2.6.9 Payload Fluid System Forward Work. As part of future maturation of the CPST payload fluid system, several key areas of design effort have been identified. These include, but are not limited to, propellant transfer tank components and interfaces, and resultant packaging and size; conceptual design maturation for the cryocooler circuit (design, component selection, and assembly), and for the BAC design (line and manifold configurations, mounting, and assembly, active cooling of structural, fluids and instrumentation penetrations, and leakage assessment and potential refrigerant resupply approaches); trades regarding transfer operations (flow rate and capability for variation, planned sequences, chill-in approaches, and venting operations).

2.3.3 Structural Analysis

2.3.3.1 Ground Rules and Assumptions. The standard ground rules and assumptions for NASA in-space vehicles were used per NASA-STD-5001a.[3] The prototype verification approach was assumed. The metallic factors of safety for yield and ultimate were 1 and 1.4, respectively. The composite ultimate design factors for acreage and discontinuities were 1.4 and 2, respectively. The proof pressure factor of 1.5 and burst pressure factor of 2 were used for all pressurized vessels. A local panel-buckling knockdown of 0.65 was used on all panels. The minimum allowable frequency was assumed to be 25 Hz, which is standard for payloads on the range of launch vehicles under consideration during the study.

2.3.3.2 Analysis Methodology. The standard structural analysis procedure from the Advanced Concepts Office at MSFC for prephase A studies was employed. This process begins by obtaining the CAD configuration model. This model includes all the primary structures, all pressure vessels, and any large supporting structures. The CAD model is imported into FEMAP, a Windows-based pre- and post-processing program designed for constructing finite element models (FEMs). Once the FEM is constructed, the NX Nastran structural solver is employed to determine the stresses and deflections in all structural members. These results are then imported into HyperSizer, a structural optimization program, to determine the lightest structure possible, given the configuration. A subprogram within HyperSizer, HyperFEA®, updates the structural properties based on the optimization, and then reruns Nastran to obtain new loads. This process is repeated within HyperFEA until the loads and stresses converge, leading to a properly optimized structural mass.

Any required spacecraft adapter is included in the FEM to properly constrain the model. However, the adapter is purposely oversized, driving more loads into the CPST structure. This helps lead to a conservative mass result.

For the current study, the structures were sized for strength, stiffness, and local buckling. A global buckling analysis was not performed due to time constraints. However, for payloads of the size of CPST, global buckling is typically not a controlling load case.

2.3.3.3 Structural Description. Since the primary focus of CPST is based on a cryogenic test fluid, it was determined that the storage and transfer tanks would be nonstructural pressure vessels. These tanks would be structurally supported via composite struts only, thus limiting the heat transferred into the tanks through conduction. This led to an outer panel construction with appropriately placed ring frames from which the tanks could be mounted. All structures, other than the composite struts, were assumed to be aluminum for cost purposes. Figure 34 shows the CPST payload configuration.

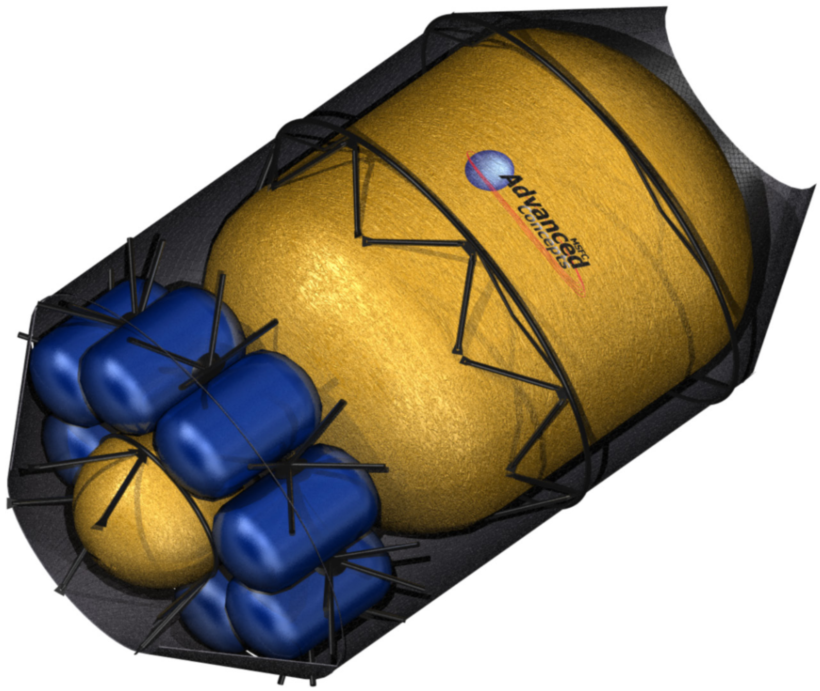


Figure 34. CPST payload configuration.

The structural analysis was performed in parallel with the launch vehicle selection. A list of candidate launch vehicles was available from which the worst-case constraints were selected. The configuration was designed to fit within the dynamic envelope of the Minotaur IV or Taurus XL shroud, leading to an OD of roughly 2 m. One-foot spacing between the outer shell and tank wall was assumed to allow room for tank insulation, test instrumentation, wiring, etc. These constraints were used to calculate the required tank dimensions.

While a small shroud was used to size the dimensions of the CPST concept, the worst-case launch loads from the launch vehicles under consideration were determined to be from the Falcon 9. The maximum axial loads with a simultaneous lateral component were applied to the FEM. Due to the symmetric nature of the configuration, the lateral loads were only applied at orientations of 0, 45, and 90 deg. This led to three load cases for each of the load conditions shown in table 26.

Table 26. Falcon 9 load conditions.

Load Condition	Axial g's	Lateral g's
Maximum axial (compression)	6.5	0.5
Maximum axial (tension)	–2	0.5
Maximum lateral (compression)	3.5	2
Maximum lateral (tension)	–1.5	2

For both the storage and transfer tanks, the test instrumentation was also being determined in parallel with the structural analysis. The uncertainty in mounting the test instrumentation led to a decision to force one-quarter of both tanks to have a 0.300-in wall thickness. This wall thickness was believed to be adequate for any mounting needs of the instrumentation. Applying the structures ground rules and assumptions and methodology described in this section with the worst-case loads led to the prephase A FEM shown in figure 35.

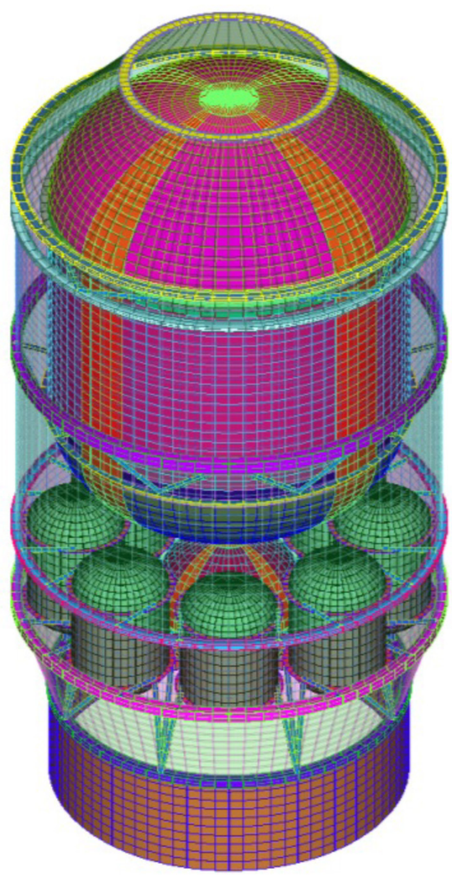


Figure 35. CPST finite element model.

2.3.3.4 Analysis Results. The optimized results for the primary structures and tank attachment struts were obtained through an advanced prephase, a structural analysis. Secondary structures are estimated to have a mass that is 20% of the primary structure mass based on previous studies and as-built masses for other space structures. The same rational is used to estimate the mass of the joints and fittings for the struts at 50% of the mass of the struts themselves. These estimates are added to the master equipment list (MEL). The storage and transfer tank are sized using the FEM, but they are book-kept under a different portion of the MEL (not within the structures section of the MEL). They are included in table 27 for completeness. Due to the preliminary nature of the study, and the lack of maturity of full-scale flight requirements, a 30% contingency has been added to the basic mass to calculate the predicted mass of the structural components of CPST.

Table 27. Structural analysis mass results.

CPST MEL		Qty.	Unit Mass (kg)	Basic Mass (kg)	Contin-gency (%)	Contin-gency (kg)	Predicted Mass (kg)
Pathfinder							
1.2.3	Storage tank	1	108.78	108.78	30	32.63	141.41
1.2.7	Transfer tank	1	14.30	14.30	30	4.29	18.59
5.0 Structures			150.53	150.53	30	45.16	195.69
5.1	Primary structure	1	112.47	112.47	30	33.74	146.21
5.2	Secondary structure	1	22.49	22.49	30	6.75	29.24
5.3	Tank attachments	1	10.38	10.38	30	3.11	13.49
5.4	Joints and fittings	1	5.19	5.19	30	1.56	6.75

2.3.4 Thermal Analysis

Thermal control is a major issue in the design of CPST. The primary concern is to minimize the heat input into the cryogenic system. This section provides details of the thermal design and analysis performed for CPST including orbital attitude, propellant storage tank heat loads (tank penetrations, support structure, and insulation system), and required heat rejection capacity and propellant loss rate.

2.3.4.1 Top-Level Ground Rules and Assumptions. Table 28 contains the ground rules and assumptions relating to cryogenic thermal management of the LH_2 experiment fluid and experiment thermal control. A key design decision to maintain the spacecraft attitude with the bus towards the Sun was made on the basis of thermal impact on the cryogenic experiment. CFM is critical to the successful completion of the experiment timeline and relies on a number of important ground rules and assumptions, including ground replenishment and gaseous nitrogen (GN_2) purge for the LH_2 tank until $\approx T=0$, a nonstructural, composite strut supported tank design, and active/passive thermal control features on the supply tank.

Table 28. Ground rules and assumptions.

3.0	**CFM GR&A**	
CFM GR&A	Ground replenishment of cryogens	Ground replenishment until ~ *T*–0
	Vehicle structural/thermal design	Similar to CPS nonstructural tank concept
	Cryogenic fluid management design	Similar to COLDSAT plus active system
	Target storage conditions	Normal boiling point
	CFM performance demos	Active and passive CFM
	Transfer modes for transfer demos	Pressure-fed and pump-fed
	Thrust accelerations for transfer demos	Zero thrust, 0.00003 g axial, 0.001 g axial
	Other demos	Mass gauging, liquid acquisition
	Ground purge	Dry GN_2 purge provided for LH_2 and LO_2 tank MLI util ~T–0
	Tank venting during ascent	Active venting to relieve pressure
	Post-ascent thermal transient	Passive CFM initiated ASAP to assess thermal transient
	Margin on all heat loads	50%
7.0	**Spacecraft Thermal Control**	
Thermal GR&A	Thermal control system	Thermal control of the spacecraft shall utilize standard, flight-proven materials such as MLI, selected surface finishes, foils and tapes, conduction (coupled/decoupled mounting details), optical solar reflectors, resistance heaters, thermostats, and controllers to maintain acceptable spacecraft subsystem component temperatures with adequate margins during all mission phases
	Thermal isolation	Spacecraft bus and CFM experiment subsystems electronics shall be isolated to the maximum extent possible from the cryogenic (experiment) portions of the spacecraft
	Heat rejection	Deployed radiators, optical properties α=0.1, ε=0.85, areal density = 8.5 kg/m^2
	Attitude control	Controlled so that the aft end of the spacecraft is always facing the Sun in order to minimize heating of the cryogenic systems by direct solar flux

2.3.4.2 Analysis Methodology. A geometric representation of the CPST experiment system including vehicle and propellant tank dimensions from the structures and configuration subsystem experts was developed using Thermal Desktop®. Environmental radiative effects were analyzed using RadCAD, which is integrated into Thermal Desktop. The vehicle orbit is circular about the Earth at an altitude of 500 km and a beta angle of 63 deg; this is assumed to be a worst-case hot environment. This orbit is shown pictorially in figure 36. The vehicle orientation is solar inertial such that the aft end of the spacecraft (bus end) is always facing the Sun in order to minimize heating of the cryogenic systems by direct solar flux.

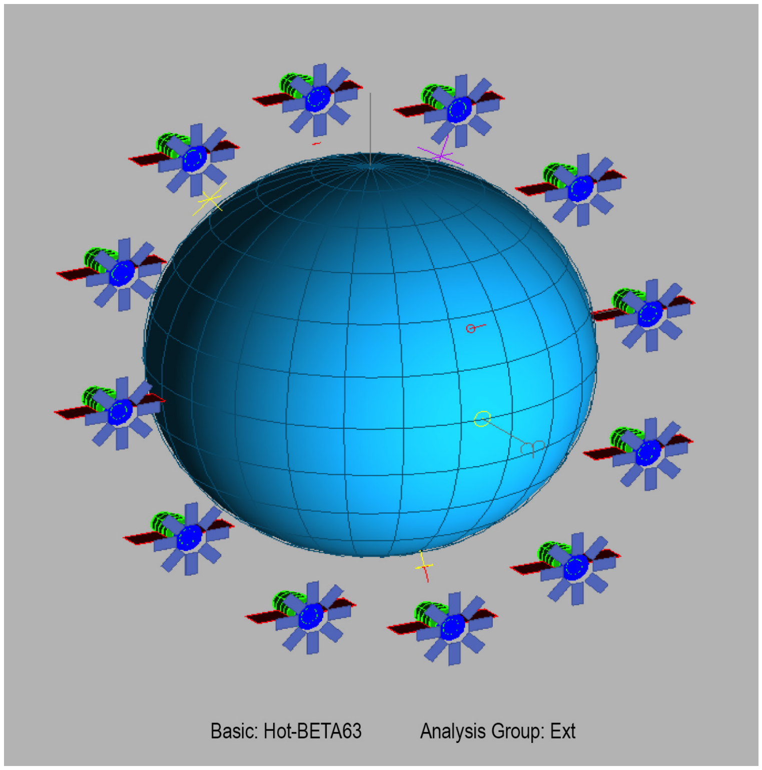


Figure 36. CPST orbital plot.

The thermal analyzer utilized for the analysis is SINDA/FLUINT, also integrated into Thermal Desktop. Environmental boundary conditions, radiator temperatures, and heat leak into the storage and transfer tanks due to structural attachments were determined from the vehicle thermal model.

The active cryogenic storage analyzer tool was used to estimate mass and power for the fluid storage tank insulation and other hardware used to maintain in-space cryogenic propellants using active cooling. An RBO system was designed for the CFM active system on the supply tank, which uses a 90 K cryocooler of 20 W cooling capacities and a BAC network to intercept heat at the actively cooled radiation shield imbedded within the hydrogen tank insulation. Helium gas is circulated through tubing that is bonded to the low mass foil shield.

The passive cryogenic storage analyzer (PCSA) tool was used to optimize and estimate thermal insulation required for the LH_2 transfer tank, which will utilize only passive thermal control features.

Thermal control of flight avionics is the responsibility of the spacecraft bus. Experiment avionics are located between the cryogenic experiment system and the spacecraft bus and rely on active and passive thermal control techniques. Heaters are used in the electronics bay to assure that minimum temperature requirements are met when the equipment is powered down. The electronics bay is isolated to the maximum extent possible from the cryogenic experiment.

2.3.4.3 System Description. The CPST CFM/thermal system is shown in figure 37. Tailoring component layouts, surface finish selection, tank supporting details, and heater sizing to maintain experiment and avionics temperatures developed the thermal control system. The thermal model incorporated white paint on the exterior of the spacecraft. Supply and transfer tank struts were sized for aluminum, and composite properties were used in the model. Pressure tank temperatures are maintained using MLI and heaters.

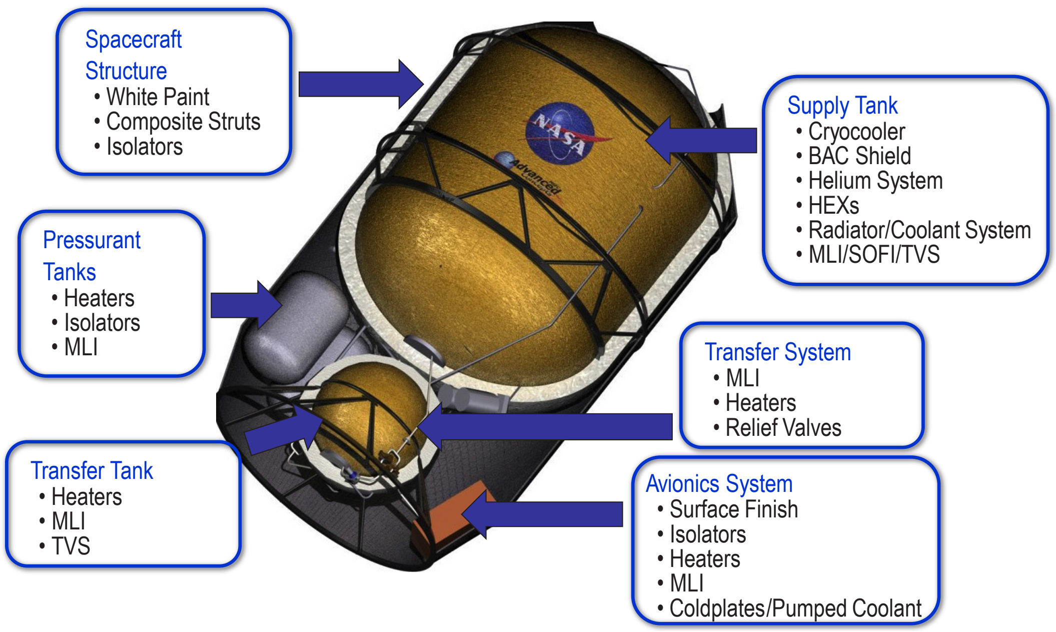


Figure 37. CFM/thermal system description.

The flight avionics are located on the spacecraft bus and are outside the scope of this section. CPST avionics consists of components required to support the payload experiments/demonstrations only and includes an experiment computer and data unit, a video system, and a thermal control system. This equipment is located between the experiment and the spacecraft bus and utilizes active cooling (coldplates and pumped fluid system) and passive thermal control (MLI, heaters, select surface finishes, and isolators) to maintain temperatures within an acceptable range. On-orbit power dissipation levels for the experiment avionics is shown in table 29.

Table 29. CPST avionics power dissipation.

Subsystem/Component	Quantity	Power (W)	Total Power (W)
Experiment flight instrumentation and electronics			
Experiment computer and data unit	1	120	120
Video System			
Cameras (four total, only two active at a time)	2	2.4	4.8
Video power and control	1	36	36
Video data storage	1	65	65
Thermal control system			
Pump control unit	1	56	56
Heater controllers	2	7	14
Totals			295.8

The supply tank is outfitted with 90 layers of MLI, SOFI, a BAC shield, helium system, heat exchangers, TVSs, and a radiator system for heat rejection. The transfer system and transfer tank relies on MLI, heaters, and relief valves for thermal control. Heaters are used to raise the temperature of the transfer system between experiment transfers in order to simulate initial conditions in a timely manner.

2.3.4.4 Analysis Results. This section provides analytical results of the CFM system thermal design and analysis.

Table 30 summarizes the mission details that were incorporated into the analysis. Propellant loss was calculated for active and passive storage, and each transfer demonstration based on tank and transfer system heating rates, as specified in table 30.

Table 30. CFM/thermal summary.

	CPST Govt POD LH_2
Mission timeline	
Propellant load	260 kg
Transfer tank vol. (% of storage tank)	5%
Transfer duration (each)	15 min
Active storage duration	5 mo
Passive storage duration	1 mo
Number of transfer demo sequences	2 (3 desired)
Propellant loss	
Propellant loss (active storage)	8.2%/mo
Propellant loss (passive storage)	9.3%/mo
Propellant loss (transfer demo, each)	20.7%
Propellant loss (active storage)	21.4 kg
Propellant loss (passive storage)	24.2 kg
Propellant loss (transfer demo, each)	53.9 kg
Tank heating rates	
Spacecraft internal ambient temperature	186 K
Storage tank heat load (passive storage)	4.1 W
Storage tank heat load (active storage)	3.7 W
BAC heat load (active storage, W)	2.1 W
Transfer tank heat load (during transfer)	2 W
Transfer system heat load (during transfer)	10 W
Tank insulation	
Storage tank No. MLI layers	90
Transfer tank No. MLI layers	90
Storage tank MLI thickness	0.0896 m
Transfer tank MLI thickness	0.0877 m
Storage tank SOFI thickness	0.0127 m
Transfer tank SOFI thickness	0.0000 m
CFM subsystem mass	
Storage tank active CFM subsystem mass	27.9 kg
Storage tank passive CFM subsystem mass	72.1 kg
Transfer tank passive CFM subsystem mass	30.7 kg
Vehicle thermal control	
Radiator effective temperature	244 K
Total heat rejection	931 W
Total radiating area	9.3 m^2
Radiator mass	79.1 kg
Insulation and thermal control materials	7.5 kg
Axial thrust for transfers and mass gauging:	
Total duration at 0.001 g	72 min
Total duration at 0.00003 g	472 min
Fluid system parameters:	
Transfer fluid velocity	1.48 m/s
Transfer mass flow rate	0.0098 kg/s
Transfer system length	3.9 m
Transfer system diameter	0.0109 m
Storage tank mixing rate	7 gpm

The thermal analysis temperature results are shown in figure 38 and indicate a spacecraft internal ambient temperature of 186 K. The radiator effective temperature is 244 K and required radiating area is 9.3 m^2. The radiator is sized to accommodate experiment cryocooler and avionics heat rejection in a LEO environment. A 50% margin on power/heat dissipation has been included to account for growth in the preliminary estimates and a 50% increase in heat rejection area has been considered to accommodate reduced radiator capability due to micrometeoroid and orbital debris damage.

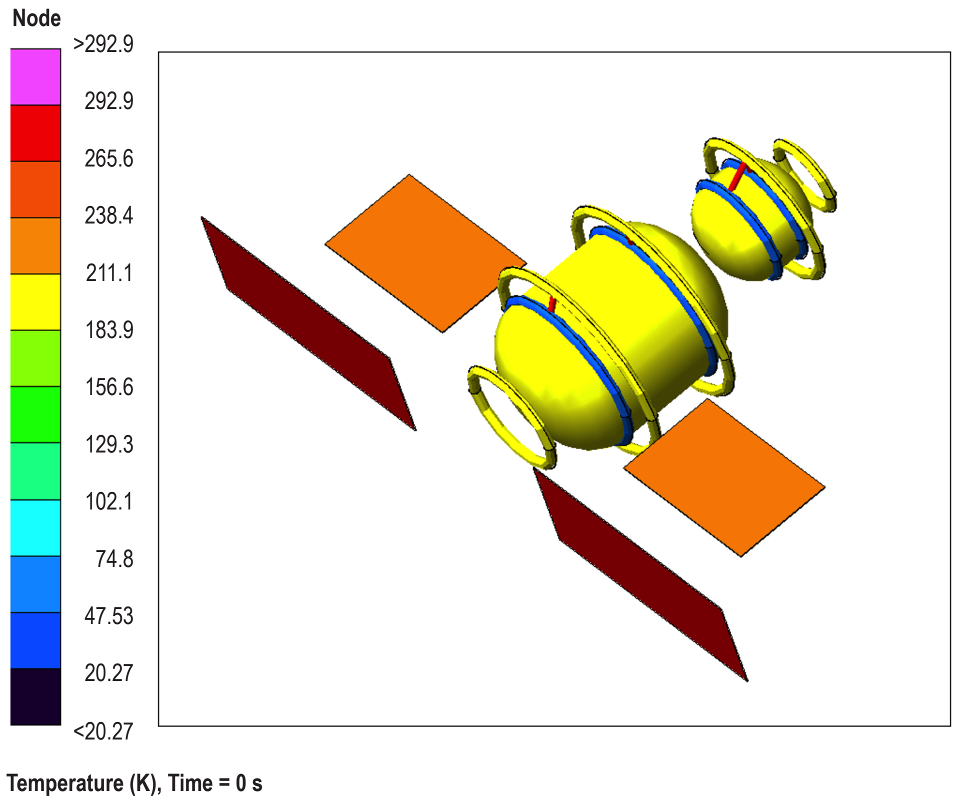


Figure 38. CPST temperature results.

The current solar array configuration is different from that represented in the thermal model, but impact to preliminary results should be minimal. Power requirements for the active and passive CFM, transfer tank, and transfer system warm-up, transfer pumps, and avionics heaters are shown in table 31. Thermal input power is shown in table 32.

Table 31. Propellant loss and heating rates.

CPST Government POD Propellant Losses	Propellant Loss (kg)	Steady-State Heating Rates					
		Tot Q Tank (W)	Str Q Tank (W)	Pen Q Tank (W)	Rad Q tank (W)	Tot Q BAC (W)	Rad Q BAC (W)
Passive storage	24.2	4.13	2.50	0.61	1.02	–	–
Active storage (reduced boiloff)	107.2	3.66	2.50	0.61	0.55	2.10	2.10
Transfer tank (includes chilldown and residuals)	50.4	2.02	1.31	0.53	0.18	–	–
Transfer system (includes chilldown and residuals)	57.5	10.01	–	–	10.01	–	–
Storage tank residual	7.8	–	–	–	–	–	–
Total propellant loss	247	–	–	–	–	–	–

Table 32. CFM/thermal input power.

LH_2 Operation		Power (W)	Duration	Comments
Active CFM		51	5 mo	Constant
Passive CFM	Average	1.8	1 mo	
	Peak	111	5% duty	TVS 5% duty
Transfer tank/transfer system warm-up		22.6	48 hrs	After each two-way transfer (8 total)
Transfer	Pumps	6.7	0.5 hr	Each two-way transfer (8 total)
Thermal control	Avionics heaters	25	6 mo	Entire CFS avionics subsystem, 50% duty

Additional work is required to incorporate spacecraft bus, experiment avionics, and pressurant tank details into the vehicle thermal model. Evaluation of conductive isolation and heat rejection options for avionics, pressurant tanks, and cryocoolers is needed. Incorporation of active heat interception at the tank support structure is also included in future work.

2.3.5 Avionics, Communication, and Data Handling

2.3.5.1 Avionics Approach. This TM focuses on the results of the Pathfinder POD design, which has only an LH_2 system; no LO_2 system is included. The avionics system for the CPST demonstrator consists of all necessary command, data handling, and instrumentation required to control and monitor the various experiments of the mission. Components of the system include an experiment computer/data unit (ECDU), a pump control unit, two heater controllers, a video system, and cabling. No redundancy or fault tolerance is required for this mission, and all systems are single string. The plan for this POD study was to develop the payload independently of the spacecraft bus and integrate the two systems later in the development cycle. The two systems will be connected and communicate through a serial data bus, where an RS422 is suggested. Given this approach, the only avionics sized are those required to support the payload experiments. No spacecraft bus hardware is included. However, the experiment requirements were used to derive spacecraft bus requirements, which are also included in this TM. The spacecraft requirements were used to bound potential commerical off-the-shelf bus options. It is realized that a fully integrated payload with the spacecraft could be more efficient in mass, but saving would be in the order of tens of kilograms and would not impact the mission in that way. From a program administration perspective, independent experiment development makes more sense.

The spacecraft bus is to perform all stationkeeping operations, maneuvers, and ground communications including uplinks and downlinks. The payload ECDU computer will contain the experiment software and sequencing control, and order propulsive settling maneuvers to the flight computer as required for the various experiments. The payload avionics will store and forward experiment data, including video, to the spacecraft flight computer for downlink. The flight computer is to call experiment data from the ECDU based on ground link availability and memory capacity. The payload ECDU will contain sufficient memory capacity to store up to 24 hr of peak experiment data. It is believed that the data transmission rates of this mission are now well within standard capacities of both spacecraft and ground network systems, and that no memory overload should occur.

In order to bound the data rate problem early in the study, a minimum and maximum video data rate was determined based on reference missions and video system demands. Assuming two out of four cameras (for the Pathfinder POD) would be running, a lower bound of 6 Mbps per camera for black and white and 15 Mbps per camera for color was assumed would be running. With these lower and upper bounds, it was determined by analysis that no pointing high-gain antenna would be necessary if transmission power were highly amplified (from 10 to 20 W) using X-band. Using S-band frequencies would require excessive amplification, and was thought impracticable at these data rates. After the video requirements were better defined, it was determined that even the lower data bound estimate was too high. A total data rate after compression is only 4.2 Mbps for Pathfinder. Further link budget analysis showed that by using multiple ground stations, S-band could be used. One ground station would probably suffice using X-band. Without knowing the cost impact/comparison of the two options, both were carried forward in the documentation as ground communication options for the spacecraft bus supplier to decide.

An experiment instrumentation list was compiled by the CFS definition group and was used to establish data rates required for sizing onboard data handling systems and downlink communication capabilities. Instrumentation includes resistance temperature sensors, silicon diode temperature sensors, pressure sensors, voltage and current sensors, accelerometers, wet/dry mass gauging, and RFMG sensors. Instrumentation also includes video recordings of fluid transfers from the propellant storage tank to the transfer tank and back again, along with monitoring the propellant state within the tanks.

Sampling of temperature sensors was assumed to be at 1 Hz, while most other assumed sampling of sensors was at 10 Hz. Adding up the number of sensors at the appropriate sampling rates and multiplying by assumed bit counts required per sample gives the total data rate per sensor type. Bit counts were assumed to be either 8 or 12 per sample to include required resolution and formatting overhead. With the one fluid system of Pathfinder, there were over 300 temperature sensors alone. Although this housekeeping instrumentation data rate is <100 Kbps total, it can become significant over long periods of accumulation.

2.3.5.2 General Ground Rules and Assumptions. Because of the CPST mission class, payload avionics has no fault tolerance requirement. However, given the criticality of this technology development for in-space propulsion systems and for the future of space exploration in general, it would make sense to incorporate some fault tolerance if it has little impact on mission cost. Since the ECDU mass estimate is small, only 8 kg, some built-in redundancy can be accomplished without affecting mission performance. A backup processor board could increase mission assurance significantly while adding only a couple of kilograms to the ECDU mass.

Earlier in the study it was thought desirable to have a real-time uplink capability through the TDRSS communication system in order to be able to observe data and video, to alter a test in process, or simply shut a test down if there was a problem. After considering both the operational cost and spacecraft systems needed to communicate with TDRSS, including a gimbaled high-gain antenna, it was decided this real-time feature was not worth the trouble. It was subsequently ground ruled that no real-time data link was required and that data would be stored and forwarded during ground station passes. From this point, it became a matter of how much time was available to downlink the data and video. Uplink commands would be required only when necessary to modify operation or troubleshoot.

To keep operational costs down, an effort to utilize only one ground station was made. Early study analysis, with twice the data rate required, showed an X-band link would be required using an omni-directional antenna. However, with an extension of $\approx$7 days between test sets given and a reduction in data rates due to program descope, further ground link analysis showed that multiple S-band ground stations could accomplish the task. Using at least two S-band ground stations, with a minimum of six links per day, all data could be downlink conservatively within 4 days, leaving 3 days for test data analysis. This plan would give mission controllers time to troubleshoot, derive contingency plans, and uplink test alterations before the next test sequence started.

It should be noted that no STK ground track analysis had been done until later in the study, by the GSFC team, and all link time assumptions were based on reference missions. Ground track analyses are sensitive to altitude and inclination, and link budgets are sensitive to required data rates, all of which were fluctuating throughout the study. To do a ground track analysis prematurely would not have been beneficial. The GSFC ground track/link analysis, with a 40-deg inclination, shows that 10 links per day are possible using Wallops and Santiago ground stations. With this many links, it should be possible to downlink all test data in 1 day using S-band. It was assumed that the cost of using multiple ground stations is acceptable.

The video requirements of the demonstration testing were driven mainly by the need to see if bubbles were being generated within the system, either by the LADS or by the transfer process itself (pumps, valve turbulence, etc.). The CFM investigators judged that a 0.20-mm, bubble size detection capability would be suitable. The 0.20 mm is thought to be detectable using common video techniques and equipment. Smaller sizes would require exceptional resolution and drive up the data rates, which ultimately drives the spacecraft communication requirements. Without knowing which spacecraft bus was going to be used, which ground stations or how many, data backlog at the spacecraft level was a serious concern.

The video system for Pathfinder POD consists of four black-and-white cameras—one attached to each transfer line, one line to the transfer tank, and another line back to the storage tank. There are four cameras in the Pathfinder design, with two of them being active at a time. It is thought the cameras will view the fluid within the lines through a window viewing port that does not protrude into the line itself and would not cause any turbulence. Also, lighting ports will be required to illuminate the bubbles. A cold mirror approach with narrow-band, light-emitting diodes or multispectrum white lighting was suggested. The transfer lines were kept at ½-in diameter. Larger diameters were considered (1 to 2 in) to reduce flow rates and frame rates, and to represent more realistic line sizes of an actual CPS. Due to program budget constraints, it was decided to keep with the ½-in line since the ½-in valves were already qualified. It was thought to be cost prohibited to develop and qualify larger valves.

Another camera is mounted to each of the storage and transfer tanks for viewing the propellant state within the tanks. Viewing and lighting ports will also need to be incorporated on the tanks. These cameras can operate at a much lower frame rate than the transfer line cameras since there are no high-speed fluid movements within the tanks. However, given the much bigger viewing angle requirement, higher resolution cameras will probably be required. Since fluid transfers occur in only one direction at a time, only two cameras at a time are to be actively recording data. The active transfer line camera should be running continuously during fluid transfers. The tank cameras can be selected to monitor the acquisition tank or the receiving tank, or alternate between the two tanks at any desired duty cycle. With slow activity in the tanks, both cameras do not need to run simultaneously. This approach keeps the video data rate burden to a minimum.

Actual cameras have not been selected at this point, but a miniature imaging camera from Comtech AeroAstro was suggested. Modifications would have to be made to achieve the 120 fps required for the transfer line cameras.

It was planned to keep the video system a stand-alone system that performs data compression and provides its own data storage buffer. The system will accept commands for configuration and operation from the ECDU or flight computer directly. Analysis by the CFM investigators showed that with a frame rate of 120 fps and a data rate of 15.7 Mbps (with a flow rate of 1.48 m/s) for the transfer line video, and 1 fps at 8.4 Mbps for the tank video assumed sufficient, a total camera data rate will be 24.1 Mbps for two active cameras during transfers. Using a 10:1 compression ratio judged by the MSFC video group to be easily achievable with black-and-white video, the after-compression total video data rate is ≈2.4 Mbps. This video rate is based on the 0.20-mm bubble detection size and the ½-in transfer lines.

2.3.5.3 Guidance, Navigation, and Control Assumptions. Ground rules and assumptions for guidance, navigation, and control (GN&C) were developed to facilitate spacecraft bus selection to meet the needs of the experiment/demonstrations. Sun inertial pointing (aft end) was assumed the attitude to be maintained as much as possible for thermal control reasons (i.e., maximize the mission lifetime by minimizing boiloff). However, it is conceivable that the investigators might want to thermally load the demonstrator to challenge the cryogenic coolers, and they might want a different orientation for short periods of time. At one time early in the study, propulsion burns and even docking maneuvers were considered. Either way, the spacecraft bus should maintain some amount of maneuvering capability.

If reaction wheels (RWs) are used for the attitude control system (ACS) by the spacecraft bus, a Sun-pointing attitude should not be a problem in LEO since the momentum accumulation is cyclic per orbit, requiring little RCS desaturation of RWs. The microgravity levels of LEO will not be an issue for mass gauging experiments. An approximate value of 0.21 μg for orbital acceleration is well under the 30 μg settling level, as is the slew rate of 0.1 deg/s generating ≈0.51 μg. However, if an RCS is used for ACS, acceleration levels near 1.7 mg may result unless very low-level thrusters are used. Even so, the fidelity of low acceleration—achievable using RWs—will be difficult to achieve using only an RCS.

With RWs used for attitude control, it is assumed all major maneuvers will be performed by RCS, including propellant settling, tip-off correction, and deorbit operations. Reaction wheels can be used for a slow backup slewing capability in a contingency situation. The RCS will be used to desaturate RWs as necessary. If a commercial bus does not use RWs for attitude control, the RCS propellant mass required may be significant.

The pointing accuracy for the mission was not seen to be critical in any way. Therefore, 1 deg/axis was chosen to keep the attitude control burden low. Even with RWs, tighter pointing accuracy would require more RCS propellant for desaturation of the wheels. The RW attitude rate control was chosen to be 0.01 deg/s. Low RW rates will minimize the RW mass required. The pointing knowledge of 0.005 deg (18 arcs) was chosen to accommodate the RW rate control, and should be easily achievable with a standard low-resolution star tracker. The slew rate of 0.1 deg/s minimum (90 deg in 15 min), was judged sufficient since no high slew rate operations were required after the elimination of the propulsion and docking requirements.

2.3.5.4 Avionics Data Summary. An uplink of 1 Mbyte per week was assumed sufficient for troubleshooting commands, script modifications, and parameter changes. Estimated data storage rates for the Pathfinder POD payload are defined in table 33 for a 7-day period:

Table 33. Estimated data storage rates over a 7-day experiment.

Mission Activity	Data Storage Rate	Period
Fluid transfer	2.4 Mbps	Total active cameras during two 15-min transfers
Settled mass gauging	800 kbps	Each tank for 60 min, with a 50% duty cycle
Quiescent mass gauging	1.6 Mbps	5-s measuring time, four times per hour, continuous
Wet/dry settling gauging	160 Kbps	Both tanks for 60 min
CFM housekeeping (P, T, status)	35.6 Kbps	Continuous

The data timeline in figure 39 shows the POD data acquisition required for a 'settled' demonstration test sequence lasting 306 min. It consists of two fluid transfers—one from the supply tank to the transfer tank and another transfer back to the supply tank. There are several operations inbetween transfers, including chilldowns and mass gauging of fluids within the tanks at different settling thrust levels. The 'unsettled' transfer demonstration test sequence is very similar, except without settling thrust being done. These demonstration sequences are repeated about every 7.5 days, with two settled and two unsettled test sequences being done for a total campaign duration of 30 days. Continuous housekeeping monitoring and quiescent mass gauging is done before, after, and for about 14 days between the campaigns. Data rates shown in this graph are before overhead and margin are added, which are variables that will probably change throughout the program. However, the general profile of the curve should remain the same, showing the relative proportions of data required for each operation. Notice that the video required during the fluid transfers is the dominant data rate although the transfers only last for 15 min each.

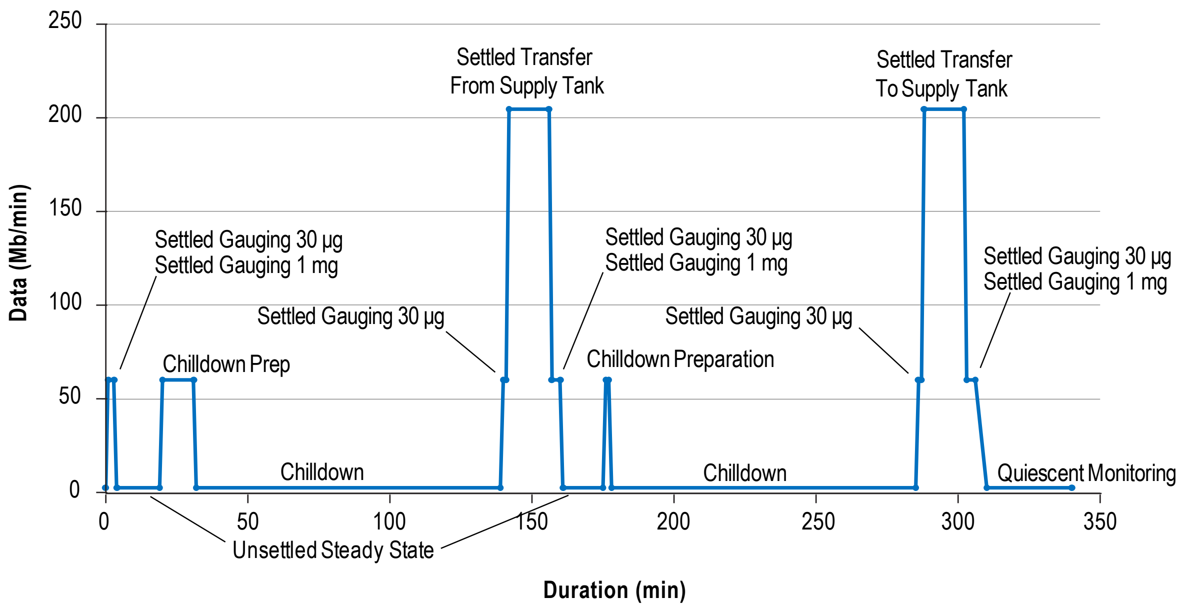


Figure 39. A typical experiment data acquisition settled demo timeline (306 min).

Table 34 shows the data volume acquired over time given the various duty durations. Because the last three line items of the housekeeping date are acquired continuously, it becomes significant over a 24-hr period, about 30% of the total data volume. The video data rates shown are after compression. RFMG is continuous during transfer operations, but reduced with a 50% duty cycle (i.e., two units at 800 kbps, 50% duty cycle). Wet/dry mass gauging is done at 1 Hz with 5 kbps per sensor. During nontesting time, quiescent RFMG is done at 15-min intervals throughout the mission (for 5 s, four times per hour). Only 8 min of these data accumulate in 24 hr. Accumulation of housekeeping instrumentation data will occur while downlinking the transfer test data, but any system selected should be able to catch up in under the 4-day maximum downlink limit.

The variables in table 34 are the downlink rates, links per day, and link durations. The daily data remainder will equal the sum of the day's volume, plus the previous day's remainder, minus the downlink volume for the day. Several graphs were generated using this matrix with different variables. It can be seen that the resulting slope of the stored date line after downlink is highly sensitive to the links per day variable.

Table 34. Data volumes acquired with duty durations: CCSDS overhead, 15% and prephase A data margin, 30%.

Measurement No.		Data Type	Notes	Collection rate (Mbps)	Duration (hr)	Volume (Gbits/hr)	Volume w/CCSDS (Gbits/hr)	Volume w/Margin (Gbits/hr)
1	Periodic events	Video-line	1	1.570	0.50	2.83	3.25	4.22
2		Video-tank	1	0.840	0.50	1.51	1.74	2.26
3		RFMG-setting	2	0.800	1	2.88	3.31	4.31
4		Wet/dry gauging	3	0.160	1	0.58	0.66	0.86
						7.79	8.96	11.65
1	Continuous data collection	RFMG-quiscent	4	1.600	0.0056	0.032	0.037	0.048
2		Pressure/temps/etc. S/C eng. data	5	0.036	1	0.130	0.149	0.194
3			5	0.005	1	0.018	0.021	0.027
				5.01		0.18	0.21	0.27
				Peak data rate	Totals	7.97	9.17	11.92

Downlink Rate (Mbps)	10	10	10	10	10	10	10
Links per day	2	2	2	2	2	2	2
Link duration (min) each	8	8	8	8	8	8	8
Total downlink time per day (mm)	16	16	16	16	16	16	16
Downlink volume (Gbits)	9.6	9.6	9.6.	9.6	9.6	9.6	9.6

Measurement No.	Daily Multiplier (hr)	Volume (Gbits/day)	Day 1	2	3	4	5	6	7
1	1	4.22	4.22						
2	1	2.26	2.26						
3	1	4.31	4.31						
4	1	0.86	0.86						
		11.65	11.65	–	–	–	–	–	–
1	24	1.16	1.16	1.16	1.16	1.16	1.16	1.16	1.16
2	24	4.65	4.65	4.65	4.65	4.65	4.65	4.65	4.65
3	24	0.65	0.65	0.65	0.65	0.65	0.65	0.65	0.65
		6.45	6.45	6.45	6.45	6.45	6.45	6.45	6.45
		18.11	18.11	6.45	6.45	6.45	6.45	6.45	6.45
		Remainder	8.51	5.36	2.21	–	–	–	–

For example, in figure 40, increasing the number of links to four per day reduces the link time to 2 days even with a lower bit rate of 7 Mbps, versus the two links per day and 10 Mbps in figure 41.

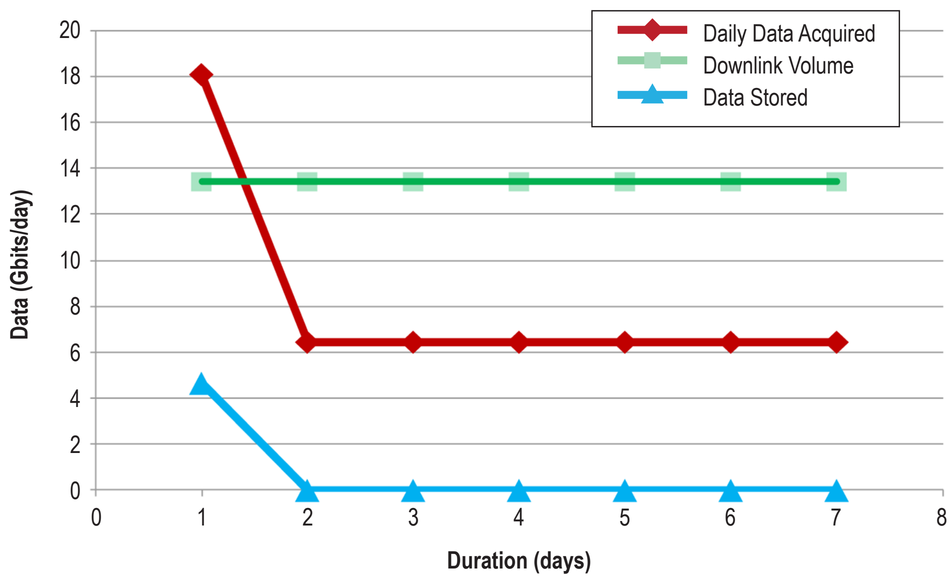


Figure 40. Data acquired and stored after downlink with X-band at 7 Mbps, four 8-min links per day.

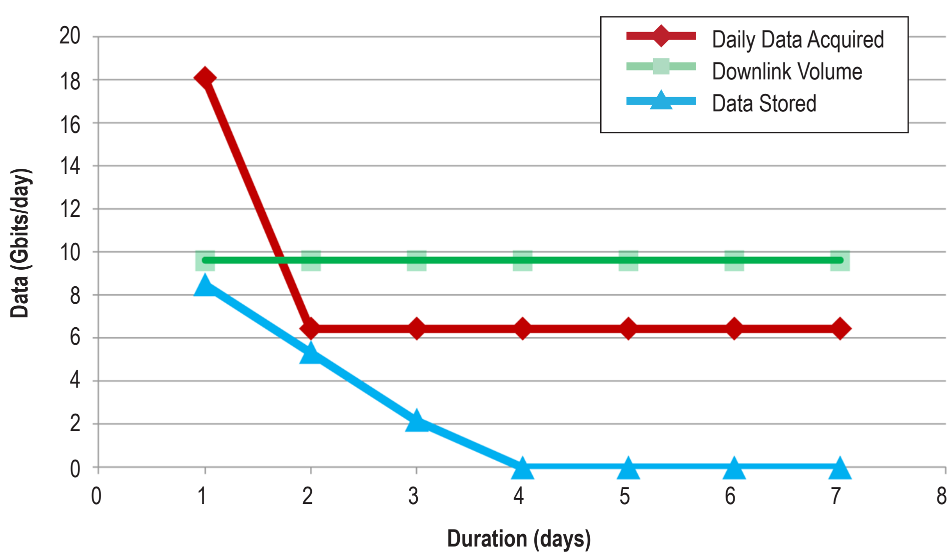


Figure 41. Data acquired and stored after downlink with X-band at 10 Mbps, four 8-min links per day.

Increasing the links per day to 10 would reduce the link time required to only 1 day even with S-band, as shown in figure 42.

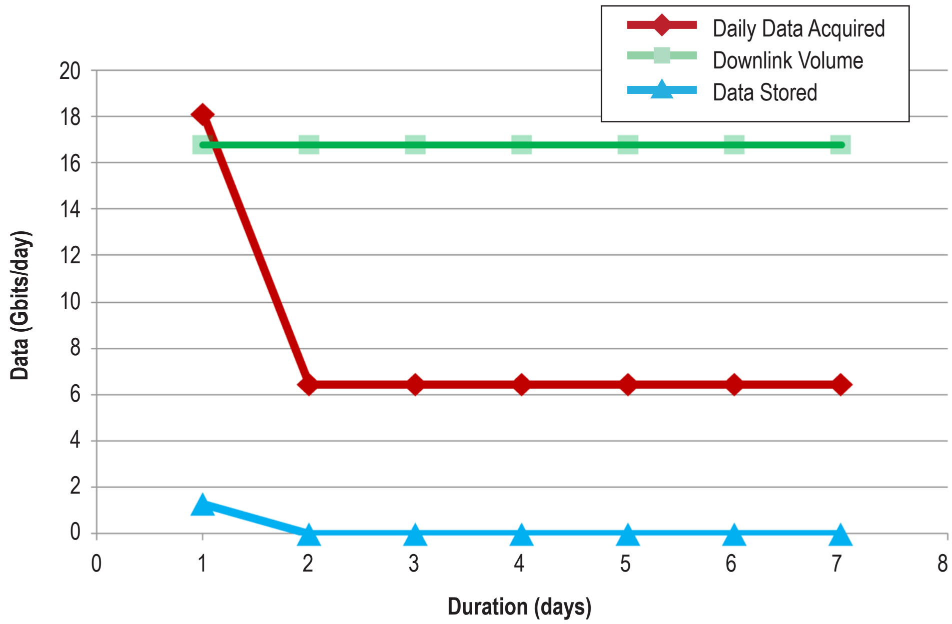


Figure 42. Data acquired and stored after downlink with S-band at 3.5 Mbps, ten 8-min links per day.

The downlink data rates for X-band and S-band were based on international bandwidth limits for the near-Earth network system of 10 MHz for X-band and 5 MHz for S-band. Link budget analysis showed a reasonable link rate for X-band would then be 7 Mbps and 3.5 Mbps. Higher or lower rates would depend on factors like modulation techniques and coding strategy. The link budget analysis giving these results used quadrature phase shift keying (QPSK) modulation and RS/Viterbi coding. In figure 40, an idealized X-band rate of 10 Mbps was used for comparison sake. The data rates in these graphs include 15% overhead and 30% margin, as indicated in table 34.

2.3.5.5 Avionics Payload System Description. Figure 43 shows a schematic of the avionics system. Note that only the payload experiment avionics system is shown. No spacecraft systems such as GN&C, communications and data handling (C&DH), or communications are included. A serial data bus will connect the ECDU to the spacecraft flight computer for command and control of the experiments, and for uploading experiment software changes. It is expected that the ECDU will command the flight computer to perform settling maneuvers as it processes the experiments through the experiment timeline and receives go status from experiment sensors. An additional dedicated bus will be used to transfer experiment and video data to the flight computer for downlinking to ground. A video control unit is shown that will accumulate and store all video data from the four cameras, perform compression, and possibly transfer video to the flight computer directly via a third data bus, bypassing the ECDU.

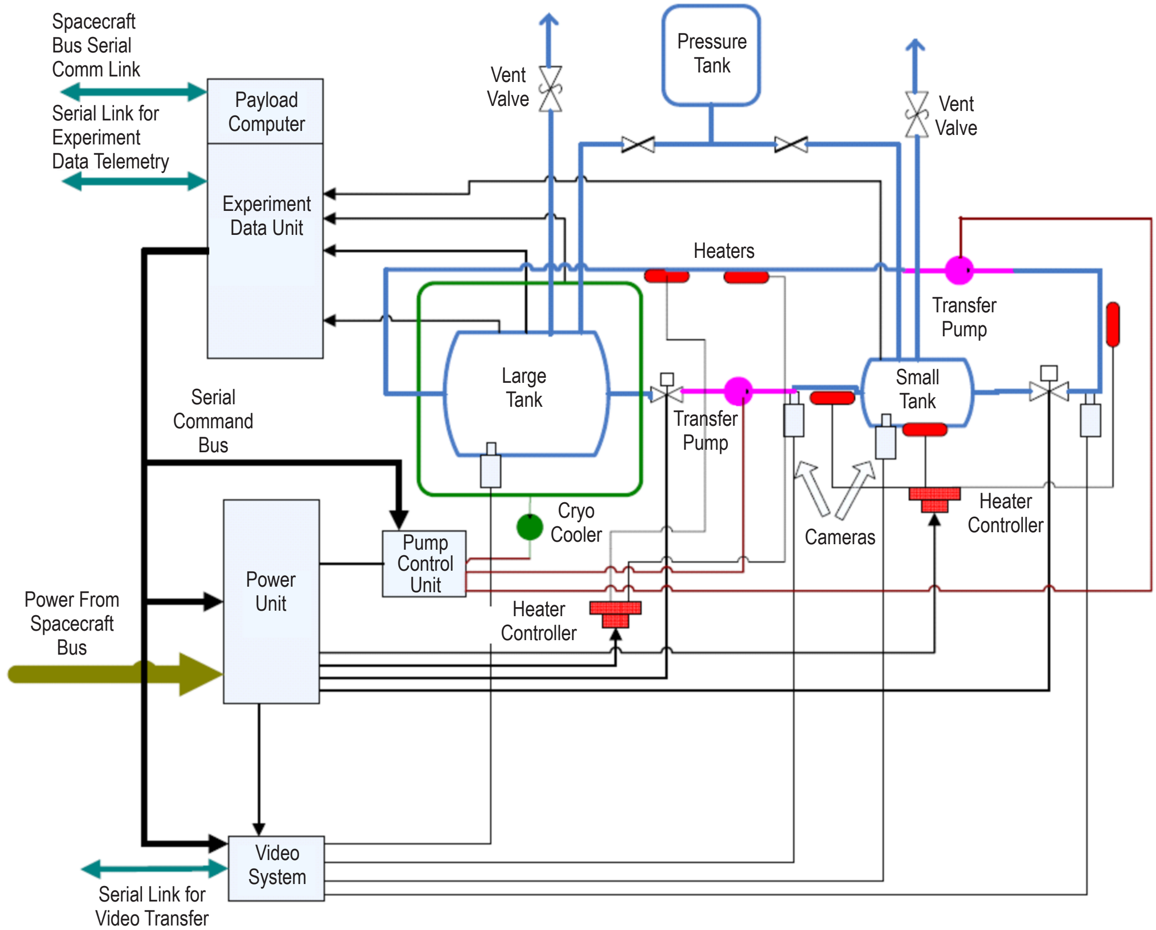


Figure 43. Pathfinder POD payload avionics and power system.

A payload power unit distributes power to the various valves, pumps, and controllers. This unit is described in more detail in section 2.3.6. Valve commands will be sent to the power unit by the ECDU. A dedicated pump controller is used to control the cryogenic coolers and transfer pumps. The ECDU will send operational commands to the pump control unit as needed to run the transfer test and maintain cryogenic cooling. With a large number of heaters and temperature sensors required, two heater controllers seemed justified for the system. It is thought that smart/active thermal control of heaters is preferred for this demonstration testing, giving flexibility to alter temperature parameters during the mission. In an actual CPS system, many of the heater settings can be fixed points.

2.3.5.6 Avionics Mass Summary. Table 35 shows the results of the avionics system mass. Despite the enormous amount of effort put into determining data rates and ground link strategies, the total avionics mass is a small portion of the overall vehicle mass. Since the communication system is to be part of the spacecraft bus, it is not included in this avionics mass rollup. For the most part, the masses shown were based on representative components and estimates for such a system. As the program develops, particularly CFM testing procedures and hardware, including video, the avionics system will also mature.

Table 35. Avionics mass summary.

Avionics	Quantity	Unit Mass (kg)	Basic Mass (kg)
ECDU	1	8	8
Pump control unit	1	5	5
Heater controllers	2	5	10
Video system	1	15.19	15.2
Harnessing	1	9	9
Total	–	42.19	47.2

2.3.5.7 Future Work. Future work for avionics systems includes the need to perform transfer video experiments to verify capture of bubbles, adequate frame rates, resolution, etc. A detailed design of a digital video system needs to be done, including memory storage, compression software development, and a spacecraft bus interface. Detailed design of the ECDU also needs to be done, including processor selection, data storage requirements and type, data buses, instrumentation, and control input/output. Software functional requirements to be developed include defining experiment timeline and control routines, defining contingency and fail-safe modes, and defining spacecraft interface protocol.

2.3.6 Power System

The spacecraft bus provides a complete power generation and management system to provide all of the power for both itself and the demonstration CFS payload. The power system described here accepts an umbilical cable carrying the payload power from the spacecraft bus and switches individual payload circuits on/off under the control of the payload computer system. The power from the spacecraft bus does not need to be conditioned but should maintain a potential of 22 to 34 V under specified loads.

2.3.6.1 Power Systems Ground Rules and Assumptions. Ground rules and assumptions for the power systems are as follows:

- Off-the-shelf spacecraft bus provides all required power conversion and energy storage.
- The power required for the payload was analytically determined in this study and used to configure the spacecraft bus.
- This power is provided via one or two umbilical input circuits to the secondary power-switching unit. The switching is controlled by the payload computer system interfaced to the switching unit using a MIL-STD-1553B interface.[4]
- Cabling between the secondary power switching unit and the payload elements was sized to 1% power loss using physics-based methods.

2.3.6.2 Power System Analysis Methodology. Because the only requirement of the power system is to provide secondary switching and distribution, the analysis consists of sizing an appropriate secondary power distribution unit to switch each of the payload demonstration circuits. Based on the power levels required and the number of circuits, a representative off-the-shelf power distribution unit for which data were available and configured for this application was selected. It must be emphasized that the particular unit is selected for sizing purposes only; it is not recommended by this study as being superior to other off-the-shelf units, but is representative of the sort of off-the-shelf units available from a variety of makers.

The cabling connecting the secondary power distribution unit to each of the payload circuits is sized to incur a 1% power loss. The cables are sized using physics-based methods.

2.3.6.3 Power Requirements. Table 36 details the maximum power requirement for each demonstration operation. Note that a 30% design margin has been added to accommodate power requirement growth for each operation.

Table 36. Power requirements.

	Avionics Power	Valve Power	Heater Power	Pump Power	Cryo Power	Total Power	W/30 (%)	Comment
Checkout	360	0	12.5	0	5.35	377.85	491	
Passive storage only	360	0	12.5	0	5.35	377.85	491	No cryocoolers running
0-g mass gauge	395	1	12.5	0	42	450.5	586	
Chilldown	395	3	12.5	0	42	452.5	588	No data transmit
Low-g mass gauge	395	1	12.5	0	42	450.5	586	
Pump transfer	395	1	12.5	7	42	457.5	595	
Pressure transfer	395	106	12.5	0	42	555.5	722	
Warmup	395	1	12.5	0	39	447.5	582	
Active idle	360	1	12.5	0	42	415.5	540	Cryo and heaters on, no data transmit

The peak power (exclusive of very short-duration transients) is 722 W. The average power over the entire mission is 539 W. Figure 44 shows the power level with respect to time over the 6-mo mission. The power level for each 30-day demo period is shown in figure 45.

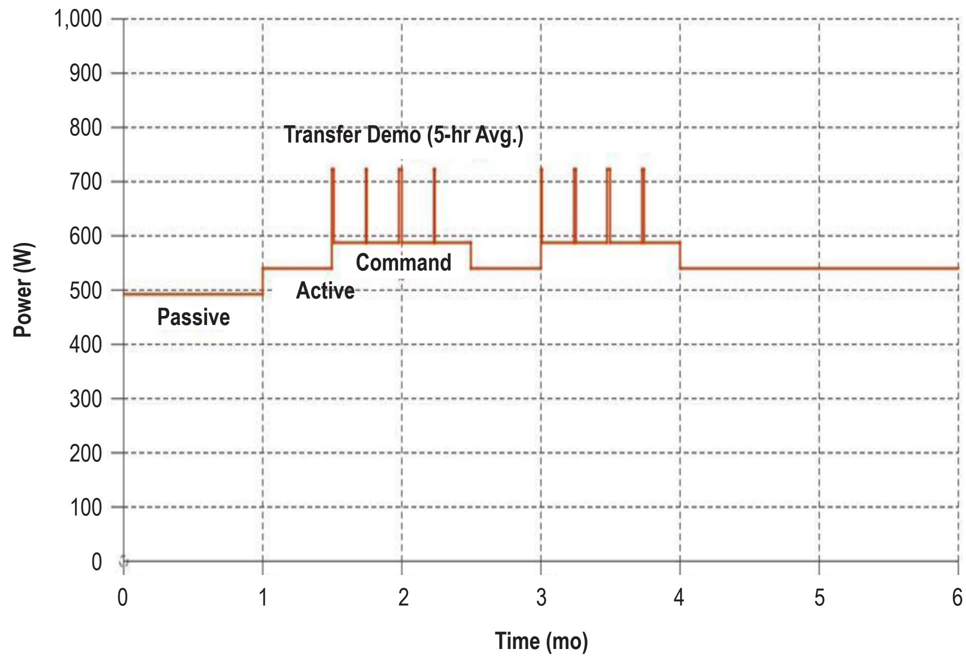


Figure 44. Six-month power profile.

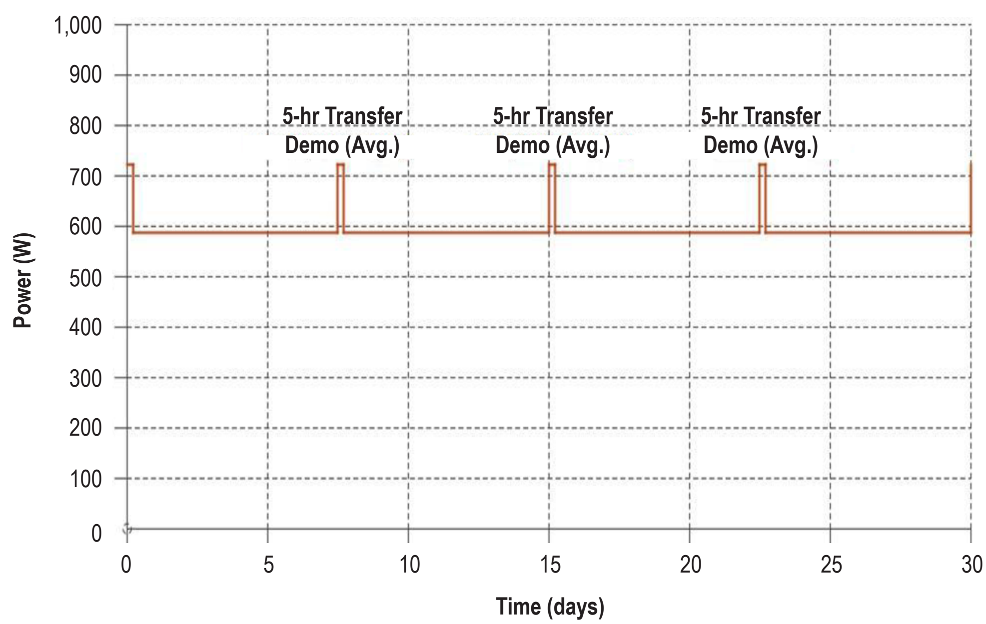


Figure 45. Propellant transfer power profile.

2.3.6.4 Power System Description. The secondary power distribution system consists of a minimal VME (Versa Module Eurocard) cage together with a set of VME power boards. Each board in the set carries its own external connectors and outer structure so that the card cage itself may be kept simple and light. The advantage of this approach is that the designer may configure a power unit containing those features (and hence boards) that are needed for a given application without carrying extra mass for connectors and structure that enable the features that the application does not need. The disadvantage is that the chassis does not provide anything (power supply, interfacing) other than a bus bar and an enclosure. The power supply, interfacing, and other required functions must be implemented on some of the VME boards themselves. The boards used for this application include the following types:

- Power distribution module (PDM)—each board switches power from its input up to 20, 4-A circuits.

- Low voltage power supply—provides 3.3, 5, and ±12 V power to the other boards.

- Advanced programmable interrupt controller (APIC)—provides interface (including MIL-STD-1553) to payload computer.

Figure 46 illustrates the configuration of the power system.

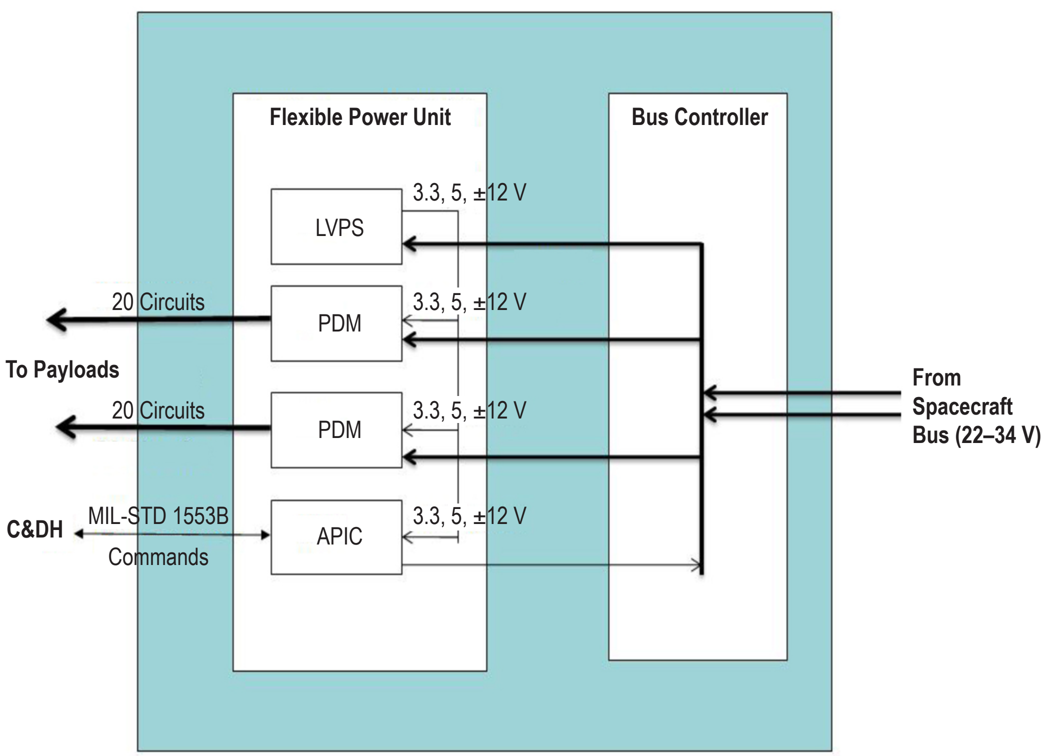


Figure 46. Power system schematic.

2.3.6.5 Power System Results. Table 37 summarizes the mass breakdown for the power system.

Table 37. Power system mass summary.

Item	Quantity	Unit Mass (kg)	Total Mass (kg)
SC-11 VME chassis	1	8.4	8.4
Power distribution module	2	1.6	3.2
Low-voltage power supply	1	0.6	0.6
APIC housekeeping card	1	0.96	0.96
Power bus control assembly	1	4	4
Cable/harness	1	1.1	1.1
Total mass			18.3

2.3.7 Mass Summary

2.3.7.1 Point-of-Departure Concept Mass Summary. This section provides an indepth mass overview for the CPST technology demonstration CPS-Pathfinder POD concept study. The masses in this section include the total wet mass/launch mass for the CPST CPS-Pathfinder POD vehicle, the CFM system/vehicle payload, and the CFM subsystems, as well as the spacecraft bus and margin for each CPS-Pathfinder element.

For clarity, a brief description of mass terminology used in this section is included here. Basic mass is defined as mass with no margin, contingency, or mass growth allowance (MGA) added to it. Margin, contingency, and MGA are basically interchangeable terminologies used in the aerospace industry. Predicted mass is the sum of basic mass added to contingency/MGA. Dry mass is defined as mass that does not include propellant or gasses/fluids loaded. Wet mass is the spacecraft's mass before liftoff and is the same as launch mass. Wet mass, also known as launch mass, includes dry mass plus the vehicle propellant and any fluids.

The CPST technology demonstration CPS-Pathfinder POD concept wet mass/launch mass is 2,298 kg. This mass includes the CFM system/payload and the spacecraft bus, as well as a 30% contingency/MGA applied to the basic dry mass of the CFM payload and the spacecraft bus. An image of the CPST CPS-Pathfinder concept is shown in figure 47 with the CFM radiators and spacecraft bus solar arrays deployed.

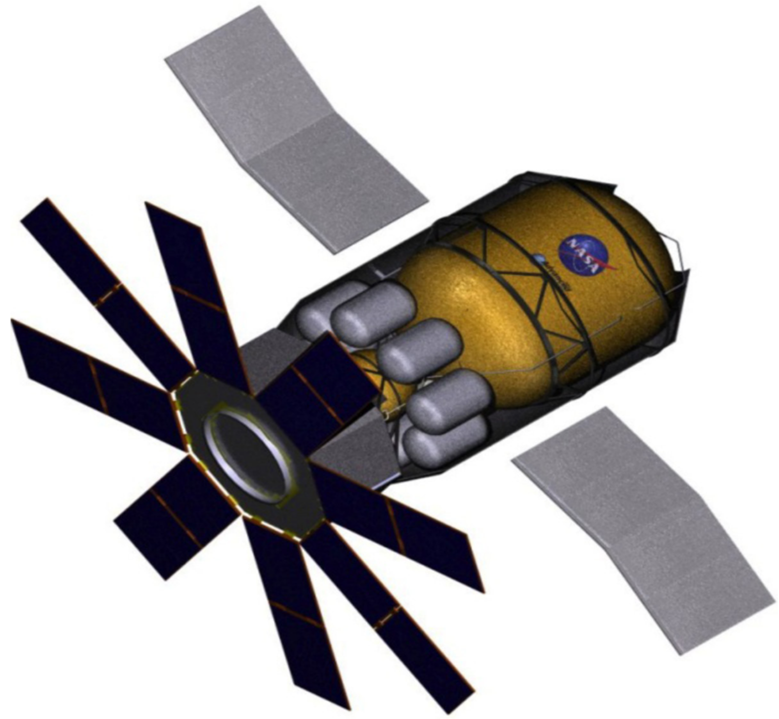


Figure 47. CPST POD configuration.

Table 38 provides summary level mass details for each CFM subsystem, as well as spacecraft bus mass data for the overall spacecraft bus dry mass and propellant. Table 38 also totals the CPST mass for the entire vehicle in the lower right under the ‘Predicted Mass (kg)’ column.

Table 38. CPST POD mass summary—CPS Pathfinder.

		Basic Mass (kg)	Contingency (%)	Contingency (kg)	Predicted Mass (kg)
CFM Experiment/Payload					
1.0	CFM demo—hydrogen system	605.10	30	181.53	786.63
2.0	Avionics	47.19	30	14.16	61.35
3.0	Power	18.16	30	5.45	23.61
4.0	Thermal	86.60	30	25.98	112.58
5.0	Structures	150.53	30	45.16	195.69
CFM experiment/payload dry mass		907.58	30	272.28	1,179.86
6.0	Nonpropellant fluids	27.50			27.50
7.0	CFM demo fluid (LH_2)	260			260
Total CFM experiment/payload system mass		1,195.08			1,467.36
Spacecraft Bus					
8.0	Mechanical	173	30	51.90	224.90
9.0	Attitude control	34	30	10.20	44.20
10.0	Thermal	22	30	6.60	28.60
11.0	Propulsion	65	30	19.50	84.50
12.0	Power	170	30	51	221
13.0	Avionics	10	30	3	13
14.0	Communication	12	30	3.60	15.60
Spacecraft bus dry mass		486	30	145.80	631.80
15.0	Bus propellant (hydrazine)	199			199
Total spacecraft bus wet mass		685			830.80
Total CPST spacecraft launch mass (CFM experiment and bus)		1,880.08			2,298.16

2.3.7.2 CFM System Mass Summary. The total predicted mass of the CFM system is 1,467 kg. This mass includes the CFM system basic mass of 1,195 kg and the 30% contingency/MGA applied to the dry mass, which is 272 kg. The CFM system contains five dry mass subsystems: CFM demonstration, avionics, power, thermal, and structures.

The CFM demonstration subsystem basic mass is 605. When combined with a 30% contingency/MGA of 182 kg, this yields 787 kg predicted mass for the CFM demonstration subsystem:

- The predicted mass for the avionics subsystem is 61 kg. This number includes a 30% contingency/MGA of 14 kg added to the basic mass of 47 kg.

- The power subsystem basic mass is 18 kg; the power contingency/MGA of 30% is slightly less than 5.5 kg. The total predicted mass of the power subsystem rounds up to 24 kg.

- The predicted mass for the thermal subsystem is 87 kg. This number includes a 30% contingency/MGA of 14 kg added to the basic mass of 47 kg.

- The structure subsystem basic mass is 151 kg. When combined with a 30% contingency/MGA of 45 kg, this yields 196 kg predicted mass for the structures subsystem.

- Nondry CFM mass includes nonpropellant fluids and CFM demonstration fluid. The non-propellant fluid is a cryogenic-fluid transfer pressurant, which is helium. The mass of the helium is 27.5 kg. The CFM demonstration fluid mass is 260 kg; the demonstration fluid is LH_2. No contingency/MGA was added to the cryogenic fluid transfer pressurant or to the CFM demonstration fluid.

Table 39 provides detailed information for each CFM payload subsystem.

Table 39. CFM subsystem mass.

CFM Demo-Hydrogen System			Basic Mass (kg)	Contingency (%))	Contingency (kg)	Predicted Mass (kg)
1.0	CFM demo-hydrogen system		605.10	30	181.53	786.63
	1.1	Storage tank active CFM subsystem	27.90	30	8.37	36.27
	1.2	Storage tank passive CFM subsystem	72.10	30	21.63	93.73
	1.3	Storage tank	108.78	30	32.63	141.41
	1.4	Storage tank subsystem	32.18	30	9.65	41.83
	1.5	Helium storage subsystem	142.16	30	42.65	184.81
	1.6	Transfer tank passive CFM subsystem	30.70	30	9.21	39.91
	1.7	Transfer tank	14.30	30	4.29	18.59
	1.8	Transfer tank subsystem	21.16	30	6.35	27.51
	1.9	Transfer system	24.86	30	7.46	32.32
	1.10	Vent subsystem	22.86	30	6.86	29.72
	1.11	Instrumentation	108.10	30	32.43	140.53
2.0	Avionics		47.19	30	14.16	61.35
	2.1	Experiment computer/data unit	8.00	30	2.40	10.40
	2.2	Pump control unit	5.00	30	1.50	6.50
	2.3	Heater controllers	10.00	30	3.00	13.00
	2.4	Video system	15.19	30	4.56	19.75
	2.5	Harnessing	9.00	30	2.70	11.70
3.0	Power		18.16	30	5.45	23.61
	3.1	Cable harness	1.00	30	0.30	1.30
	3.2	Secondary power distribution	17.16	30	5.15	22.31
4.0	Thermal		86.60	30	25.98	112.58
	4.1	Insulation and thermal control material	7.50	30	2.25	9.75
	4.2	Radiators	79.10	30	23.73	102.83
5.0	Structures		150.53	30	45.16	195.69
	5.1	Primary structure	112.47	30	33.74	146.21
	5.2	Secondary structure	22.49	30	6.75	29.24
	5.3	Tank attachments	10.38	30	3.11	13.49
	5.4	Joints and fittings	5.19	30	1.56	6.75
CFM experiment/payload dry mass			907.58	30	272.28	1,179.86
6.0	Nonpropellant fluids		27.50			27.50
	6.1	Cryo-fluid transfer pressurant (helium)	27.50			27.50
7.0	CFM demo fluid (LH_2)		260			260
Total CFM experiment/payload system mass			1,195.08			1,467.36

2.3.7.3 Spacecraft Bus Mass Summary. The total spacecraft bus mass for the CPST CPS-Pathfinder concept is 831 kg. The spacecraft bus basic mass for the dry mass elements comes to 486 kg. A 30% contingency/MGA of 146 kg is added to arrive at a predicted mass of 632 kg. No contingency/MGA is added to the 199 kg of spacecraft bus propellant hydrazine. The propellant mass total of 199 kg is added to the spacecraft dry mass (predicted mass) of 632 to arrive at the spacecraft bus mass total of 831 kg. Table 40 provides summary level mass detail for each spacecraft bus subsystem, as well as spacecraft bus dry mass and propellant (hydrazine) data and the total spacecraft bus wet/launch mass in the lower right corner under the 'Predicted Mass (kg)' column.

Table 40. Spacecraft bus mass summary.

		Basic Mass (kg)	Contingency (%)	Contingency (kg)	Predicted Mass (kg)
8.0	Mechanical	173	30	51.90	224.9
9.0	Attitude control	34	30	10.20	44.2
10.0	Thermal	22	30	6.60	28.6
11.0	Propulsion	65	30	19.50	84.5
12.0	Power	170	30	51	221
13.0	Avionics	10	30	3	13
14.0	Communications	12	30	3.60	15.6
Spacecraft bus dry mass		486	30	145.80	631.8
15.0	Bus propellant (hydrazine)	199			199
Total spacecraft bus wet mass		685			830.8

2.4 Study Accomplishments and Forward Work

The CPST Government POD prephase A study successfully accomplished all of the study objectives. The study accomplishments are summarized as follows:

- The Government POD concept provides a viable mission concept that provides a balanced approach to meeting mission objectives within project constraints.

- The Government POD concept provides a framework for evaluating alternative CPST mission concepts.

- The Government POD concept identifies a mission requirement to provide a framework for technology development and ground test planning.

- The Government POD concept provides a foundation defining a synthesized reference concept.

The results of the Government POD study provide a basis for preparing for the Mission Concept Review (MCR) project key decision point as well as moving forward with technology development activities and phase A design studies. To conclude this study, a number of areas for forward work were identified and are listed below:

- Support phase A level flight system design analysis trade studies.

- Refine planning for integrated technology and flight system component development.

- Refine test and integration plan.

- Refine mission costs estimate:
  – Launch vehicle (resolve differences between National Launch Services and WFF costs estimates.
  – Cryogenic fluid system (incorporate streamlines development approaches).
  – Spacecraft bus (investigate bus procurement options).
  – Mission operations.

- Refine project schedule inputs.:
  – Launch vehicle, cryogenic fluid system, spacecraft bus.
  – Long lead items (define critical path procurements).

- Support continued definition of system concept of operations.

- Build on Government POD study results to develop MCR (key decision point) products as required.

- Continue definition of partnership options based on project acquisition strategy.

- Review industry BAA concepts and begin mission concept synthesis.

This study culminates over 19 mo of prephase A study activities to define a mission concept for the CPST technology demonstration mission. Approximately 20 different mission concepts have been defined and evaluated. Over 90 study participants from five NASA Centers have supported the prephase A studies in which the study participants have demonstrated an exemplary level of enthusiasm, dedication, and collaboration in spite of ever-changing project constraints and mission requirements. The efforts of the study participants are greatly appreciated and provide a very solid foundation for a successful flight demonstration project.

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14. ABSTRACT
The primary purpose of this study was to define a point-of-departure prephase A mission concept for the cryogenic propellant storage and transfer technology demonstration mission to be conducted by the NASA Office of the Chief Technologist (OCT). The mission concept includes identification of the cryogenic propellant management technologies to be demonstrated, definition of a representative mission timeline, and definition of a viable flight system design concept. The resulting mission concept will serve as a point of departure for evaluating alternative mission concepts and synthesizing the results of industry-defined mission concepts developed under the OCT contracted studies.

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cryogenic fluid management technologies, cryogenic propellant storage and transfer, technology flight demonstration

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