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ARES I UPPER STAGE SUBSYSTEMS DESIGN AND DEVELOPMENT

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From 2005 through early 2011, NASA conducted concept definition, design, and development of the Ares I launch vehicle. The Ares I was conceived to serve as a crew launch vehicle for beyond-low-Earth-orbit human space exploration missions as part of the Constellation Program Architecture. The vehicle was configured with a single shuttle-derived solid rocket booster first stage and a new liquid oxygen/liquid hydrogen upper stage, propelled by a single, newly developed J–2X engine. The Orion Crew Exploration Vehicle was to be mated to the forward end of the Ares I upper stage through an interface with fairings and a payload adapter. The vehicle design passed a Preliminary Design Review in August 2008, and was nearing the Critical Design Review when efforts were concluded as a result of the Constellation Program's cancellation. At NASA Glenn Research Center, four subsystems were developed for the Ares I upper stage. These were thrust vector control (TVC) for the J–2X, electrical power system (EPS), purge and hazardous gas (P&HG), and development flight instrumentation (DFI). The teams working each of these subsystems achieved 80 percent or greater design completion and extensive development testing. These efforts were extremely successful representing state-of-the-art technology and hardware advances necessary to achieve Ares I reliability, safety, availability, and performance requirements. This paper documents the designs, development test activity, and results.

INTRODUCTION

Ares I was conceived during extensive trade studies conducted as part of the Exploration System Architecture Studies (ESAS) [1] conducted in 2005. Following these studies, the Constellation Program was redirected to the new architecture formed during the ESAS. Ares I had the function of Crew Launch Vehicle (CLV) in what was termed the 1.5 launch solution. It boosted the Crew Exploration Vehicle (CEV), later named Orion, to low Earth orbit (LEO). The Constellation architecture purpose was to initiate beyond-LEO human exploration by conducting lunar missions for extended periods, enabling exploration of the whole surface for sortie missions and up to 6-month-duration outpost missions. A companion heavy lift Cargo Launch Vehicle (CaLV) named the Ares V was planned with the Ares I to launch the balance of the mission package including an Earth departure stage and lunar lander to conduct the lunar missions. The architecture was sized and conceived to be extensible to future Mars missions, and the Ares I/CEV also had a requirement to provide crew transfer to the International Space Station if needed.

The Ares I comprised three major elements: a single solid rocket booster first stage derived from the space shuttle solid rocket boosters, a new liquid oxygen/liquid hydrogen (LOX/LH2) upper stage (US), and an US engine. The ESAS configuration consisting of a 4-segment solid booster and a Space Shuttle Main Engine-derived US engine was changed in December 2005 to a 5-segment solid booster and a J–2X US engine. US sizing was adjusted to account for the changes in propulsion.

The December 2005 configuration was the basis for the subsequent design and development efforts that concluded with Constellation Program cancellation in the summer of 2011. The US development strategy was to conduct definition and design within NASA, and select a production contractor around the time of the Preliminary Design Review (PDR). The production contractor would work with NASA to complete the final design and then be responsible for production. Boeing was selected as the production contractor.

The US comprised many subsystems, including a thrust vector control (TVC) system, an electrical power system (EPS), a purge and hazardous gas (P&HG) system, and development flight instrumentation (DFI). NASA Glenn Research Center was responsible for the definition and design of these subsystems, along with risk reduction hardware building and testing, to deliver the final designs in collaboration with Boeing. At the time of cancellation, each of these four subsystems achieved 80 percent or greater design completion, and each had undergone significant hardware building at varying levels of fidelity and extensive development testing. This paper describes the four subsystems and the design and development efforts associated with them.

THRUST VECTOR CONTROL SYSTEM

Objectives and Requirements

The TVC system is critical to launch vehicle subsystems as it provides the steering function for the vehicle in the pitch and yaw directions. The TVC system for the Ares I US had two main objectives. One was to hold the J–2X engine still and centered in the null position for about 120 sec during first stage ascent. The other objective was to gimbal the engine in response to commands from the guidance, navigation and control (GN&C) system while the J–2X engine was firing during the US portion of the flight for a duration of approximately 480 sec. The first objective prevents the J–2X engine and its nearly 8 ft long nozzle extension [2] from coming into contact with the Interstage (IS) during the separation of the first stage and US. Keeping the engine centered provided the maximum clearance between the engine and IS, which was only about 15 in. Any movement of the engine during the first stage ascent would have reduced that clearance for the separation and increased the likelihood of contact between the engine and IS that would potentially result in a mission failure. The second objective of active gimballing redirects the thrust of the engine, steering the vehicle to keep it on the proper attitude and trajectory so that the payload, the Orion crew capsule, could reach the proper orbit.

Some of the key driving requirements for TVC to meet these objectives included: holding the engine still within 1 degree of centered, ability to gimbal the engine 5 degrees in any direction and at a rate up to 5 degrees/sec, a variety of engine parameters dealing with its mass, thrust, thrust offset, moments of inertia, and vehicle requirements such as acceleration and attachment stiffnesses for the gimbal actuation hardware. All these requirements drove the design of the system so that it was sized properly to handle the vehicle and engine loads and to have acceptable response time.

System Description

During the early stages of the program in 2005 and 2006, the TVC team at NASA Glenn traded many architecture and hardware options that ultimately led to the system that was intended to fly. Initial trades looked at architecture options including hydraulic, electric, and pneumomechanical. The hydraulic architecture was selected because of the higher technology readiness level and lower cost and schedule risk. The second major trade was to evaluate the redundant power sources for the hydraulic pump. Options considered were a 270 VDC lithium-ion battery, a turbine pump assembly (TPA), and a power takeoff (PTO). For the 270 VDC battery, the primary consideration was lithium-ion, which was selected in the initial trade as the best battery chemistry for this application. The TPA used supercritical helium or hydrogen to spin a turbine at high speed, and through a gear box reduction, spins the shaft on an attached hydraulic pump. The PTO used a shaft coming off the oxidizer turbopump on the J–2X to run a hydraulic pump. This had been used on the Saturn V rocket's second and third stages, which were powered by the J–2 engine, the predecessor of the J–2X.

The TVC system architecture was single-fault-tolerant for all of the components except for the actuators. This was accomplished by having two power sources and two hydraulic systems that were cross-strapped at the actuators such that the failure of any component in either hydraulic system would allow both actuators to run off the working system. Therefore, each hydraulic power source had to be sized to handle the flow requirements for 2 actuators simultaneously. The architecture selected from the initial trade consisted of one hydraulic system being powered by a TPA and one by a 270-VDC-battery-driven pump. As requirements evolved, certain power sources became more or less favorable. For example, the PTO began to fall out of favor as the J–2X engine design development process evolved and their turbomachinery margins began to fall and the expected development cost to add the pump shaft increased. The 270 VDC battery was eventually dropped from the architecture because of corona mitigation issues, the ground and flight operations needed to support batteries, and the mass. The Ares I TVC architecture that evolved into the flight design, as a result of the second major trade to evaluate the power sources, was two identical hydraulic systems, each powered by a TPA.

Turbine Pump Assembly Power Sources

There were only a few vendors in the United States that manufactured turbine-powered hydraulic pumps, although in 2006, none had been produced for human-rated spacecraft. NASA Glenn released a request for proposal (RFP) to perform advanced development work to modify existing turbine pump technology for the Ares I US application. Hamilton Sundstrand was awarded the contract. Their TPA had been used successfully on the Delta IV booster stage TVC system.

One of the changes needed for Ares I was that the turbine housing thickness had to be increased to meet NASA requirements for human-rating such that a turbine failure would be completely contained within the housing. Another change was to replace the existing pump with a smaller version. For Ares I, the hydraulic system needed to operate at 3000 psi with a maximum flow requirement of less than 15 gpm.

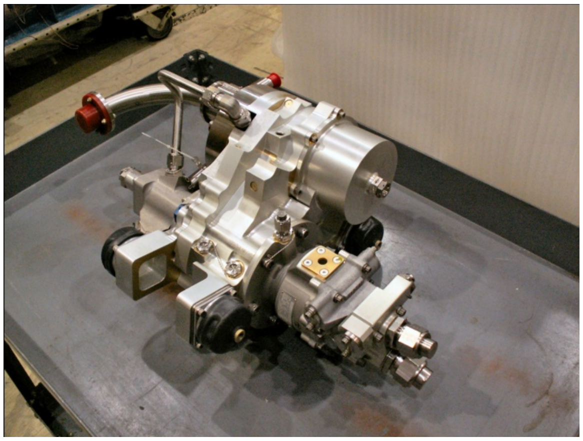

Fig. 1: Engineering Model Turbine Pump Assembly (TPA).

As a result, the pump was changed to a smaller version available from the same manufacturer which was capable of providing 3000 psi and 25 gpm. The most significant change was related to performance life. The Ares I mission was longer than Delta IV and, for human-rated vehicles, the qualification requirement was to demonstrate 4 times required life, and for Ares I US that equated to 7500 sec. Hamilton Sundstrand recommended adding a lubrication pump inside the gearbox section that would pump oil to the turbine shaft bearings, which were considered a life-limiting component. Figure 1 shows one of the engineering model TPAs built under the advanced development contract.

Actuators and Controllers

Another component of the TVC system was the actuators, which represent the load on the system. This was where the work was done to move the J–2X engine. One end of the actuator was mounted to the engine and the other end to the thrust cone of the US. The hydraulic actuators were basically a rod-and-piston device that can extend or retract as controlled by an internal servo valve. The servo valve reacts to commands from the GN&C flight computer. There were two actuators, 90 degrees apart, that together moved the engine 5 degrees in any direction as required. In addition to performing the gimballing, the actuators were also the means by which TVC held the engine still during first stage ascent. Internal to the actuator was a hydraulic lock valve that prevents any fluid flow to either side of the piston, thereby preventing any motion of the actuator. Also, a selector valve on the actuator can switch from the primary hydraulic system to the secondary hydraulic system in the event the pressure drops too low in the primary system.

Hydraulic actuators had been used successfully for decades on both human space flight vehicles, like the space shuttle and its solid rocket boosters, and unmanned rockets. A new hydraulic actuator for human space flight had not been designed since the 1970s, and there were none with an active production line. As a result, NASA Glenn put out another RFP for an advanced development contract to design a new human-rated actuator and controller sized for Ares I US and to establish the production line for flight. Moog was awarded the contract. The most significant upgrade implemented for the Ares I actuators compared to those from the shuttle era was to have electronic position feedback control compared to mechanical feedback. This was accomplished with linear variable displacement transducers. This feedback was sent to the actuator controllers to compare the commanded position to the actual position so corrections could be made if necessary.

One design change that occurred just prior to PDR was to increase the actuator's stall force. It was determined by Pratt & Whitney Rocketdyne (the manufacturer of the J–2X engine) that the side load generated during ground test startup and shutdown would be greater than any load during flight so the actuator piston area had to be increased, and other mechanical components inside the actuator had to be sized to handle this increased load.

In addition to the actuators, controllers had to be designed and built. On the vehicle, the GN&C flight computer would communicate directly with the controllers telling them where the engine needed to be pointed. The controllers would convert that command into electrical signals that the servo valve used to control the hydraulic fluid flow direction and speed internal to the actuators, which resulted in the actuator rod extending or retracting. To provide redundancy, the TVC system had 3 actuator controllers; each communicated with one of the 3 flight computers on the vehicle. Also each controller sent commands to both actuators. The TVC system would still be fully operational with the loss of a single flight computer or a single actuator controller.

In the early TVC system architectures, there was a box to collect all of the data from the entire TVC system and forward it to the flight computer to assess the TVC system health and status. Moog proposed to include that function into the actuator controllers because they already had an interface to the flight computer. This was accomplished with the use of field programmable gate arrays (FPGAs) that also handled the actuator closed-loop control and data bus communications. This consolidation reduced the TVC system overall mass and reduced the number of components that would have to be built and qualified. As a result, the controllers were named data and control units (DCUs). Figure 2 shows an engineering model actuator and DCU built under the advanced development contract.

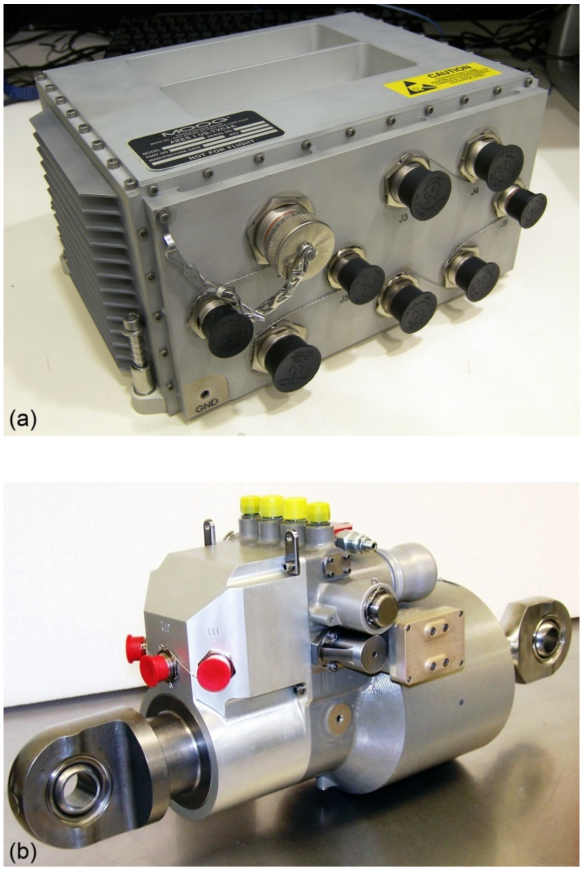

Fig. 2: An engineering model
data and control unit (a) and an actuator (b).

Hydraulic System and Other Components

Between each hydraulic power source (TPA) and each hydraulic load (actuator) there was a hydraulic system that delivered the hydraulic power from the source to the load. The hydraulic systems were more than just tubing to transfer fluid. The systems had several requirements to increase overall system stability and to maintain the proper thermal environment. The hydraulic system was required to contain enough hydraulic fluid to act as a heat sink for the heat generated by the TVC system during flight operation. The hydraulic system had to prevent pressure spikes in the system during operation. While sitting on the launch pad, the hydraulic system had to provide a means to warm the hydraulic fluid because the TVC hardware was mounted to the thrust cone, which was the underside of the cryogenic oxygen tank. Lastly, the hydraulic system had to keep the hydraulic fluid clean and prevent debris from getting into the actuators and causing damage.

Each hydraulic system contained a reservoir that could hold about 3.5 gallons that was filled to nearly 80 percent capacity. The reservoir was a cylinder with hydraulic fluid on one side and a gas-charged bellows on the other. The reservoir not only provided fluid storage but also allowed for thermal expansion of the fluid in the system and provided a heat sink for the heat generated by the TPA and the actuator in the system. During flight there was no active cooling so the reservoir had to contain enough fluid so that the overall temperature rise in flight would not exceed the hydraulic fluid maximum operating temperature of 275 °F. The reservoir also provided fluid pressure to the inlet of the TPA pump that was required for startup.

Each hydraulic system also contained an accumulator that had similar construction to the reservoir but with a much smaller volume. The accumulator's function was to dampen any pressure spikes or oscillations in the system. Each system also had a circulation pump. The circulation pump operated off ground electrical power and was only operational on the launch pad. Once the US LOX tank was filled at the pad, TVC needed a way to keep the hydraulic fluid warm. If the fluid got too viscous, the response of the actuators would be too sluggish. This pump would only operate at 1 to 2 gpm and would cycle on and off to maintain the fluid within the required temperature range for launch.

Another component of the hydraulic system was the filter manifold. The primary filter was 5 micron, and there was a 15-micron filter on the case drain. In addition to filtering the fluid in the system, the filter manifold contained check valves to prevent backflow of the hydraulic pumps. It also contained relief valves that would relieve the high-pressure side of the system to the low-pressure side in the event of a pump compensator failure. The last component of the hydraulic system was the hydraulic service panel. This panel was the interface between the ground support equipment (GSE) carts and the hydraulic systems. The GSE carts were used to fill and drain the hydraulic systems as well as ensure the cleanliness of the fluid they supplied. Figure 3 shows the entire TVC system hardware mounted on the US thrust cone. Both hydraulic systems are visible. For clarity, the other vehicle system hardware on the thrust cone is not shown.

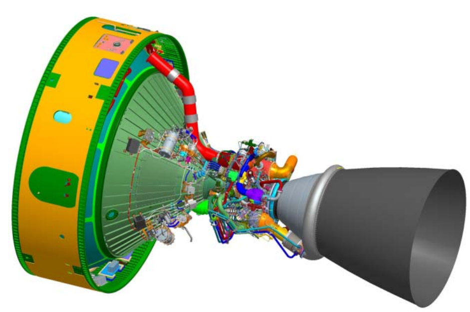

Fig. 3: The J–2X engine and the Ares I US thrust cone
showing the TVC hardware layout.

Unlike the TPA and actuators, advanced development was not needed for the development of the hydraulic components. For these items, RFPs were developed and suppliers selected to develop the design and produce engineering model units. NASA Glenn developed the specifications and oversaw the design development at the suppliers. Lastly, some of the TVC system component work stayed in-house. NASA Glenn designed and fabricated engineering model cable harnesses that were internal to the TVC system and all of the secondary structures that were used to attach the TVC hardware to the Ares I vehicle.

System Integration and Interfaces

One design change that occurred after the PDR was the result of a new requirement from GN&C to gimbal the J–2X engine before it reached full thrust. This was challenging because the hydrogen that was used to power the TVC TPAs came from the hydrogen pressurization line in the main propulsion system (MPS) and that was not available at the proper temperature and pressure until approximately 3 sec after the engine was at full thrust. Supercritical hydrogen was used as coolant for the J–2X engine, and this autogenous pressurization line was routed back to the hydrogen tank to provide the proper pressure in the tank. A portion of this warmed, but still supercritical, hydrogen was bled off and used as the TPA propellant. Spinning up the TPAs before this hydrogen was available required a different approach. The solution was to start up the TPAs using helium that was already available in an existing tank on the thrust cone and then transition to hydrogen when it was at the proper conditions. A similar helium-to-hydrogen startup transition occurred with the Delta IV TPAs. Although this could have been implemented just seconds before engine start, to make the system more robust it was implemented several seconds earlier, prior to the first stage separation. This allowed the TVC to re-null the engine and then actively hold it still during the separation, thereby increasing the chances for a clean separation. Then the actuator hydraulic locks became the backup, adding redundancy to that phase of the flight.

NASA Glenn was responsible for the overall system design including interface definition, thermal analysis, and performance modeling as well as integrating this work with the vehicle-level analyses and models. The TVC system had a significant number of interfaces. Structural and thermal interfaces included the mechanical interfaces to the thrust cone for mounting all the TVC hardware. Also, because the thrust cone was very cold due to sharing an interface with the LOX tank, TVC had a significant thermal interface to manage. The interface with the J–2X engine consisted of lugs and clevis pins for the 2 actuators. TVC had several interfaces with avionics including power for the DCUs and the propellant supply valves that controlled propellant flow to the TPAs, and command and data using MIL-STD-1553B protocol between the DCUs and the flight computers. As mentioned above, the interface with the MPS consisted of both helium and hydrogen supplies for the powering of the TPAs. There were also a variety of ground interfaces ranging from power for the circulation pumps, helium for ground testing and checkout of TPAs, and the hydraulic cart GSE used to fill and drain the hydraulic systems.

Breadboard Testing

Prior to the PDR, some TVC breadboard-level testing was performed. The breadboard comprised a TPA, a hydraulic system, an industrial actuator, and a commercial hydraulic pump that was the primary power source because the amount of testing would exceed the expected life of the TPA. The primary goal of the breadboard testing was to evaluate and tune various hydraulic system parameters. To do that, a 1-axis test rig was designed and built at NASA Glenn. This 1-axis rig housed an actuator that was the test article and 2 other actuators that provided resistive forces for the test article to oppose. These 3 actuators in the 1-axis rig are shown in Figure 4.

A breadboard hydraulic system was assembled from spare components from other programs and commercial parts that were readily available. This system was assembled in a cart, along with the pumps, and situated next to the 1-axis rig as shown in Figure 5.

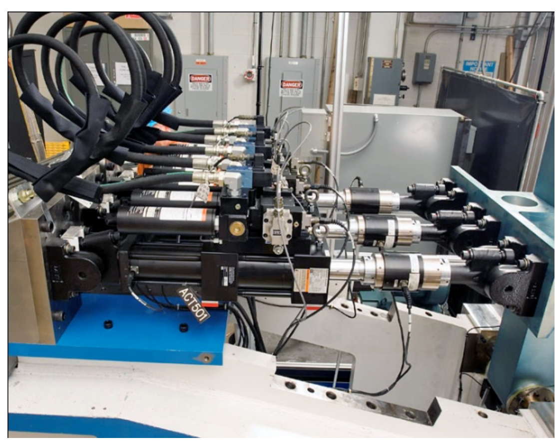

Fig. 4: Actuators in 1-axis test rig at NASA Glenn.

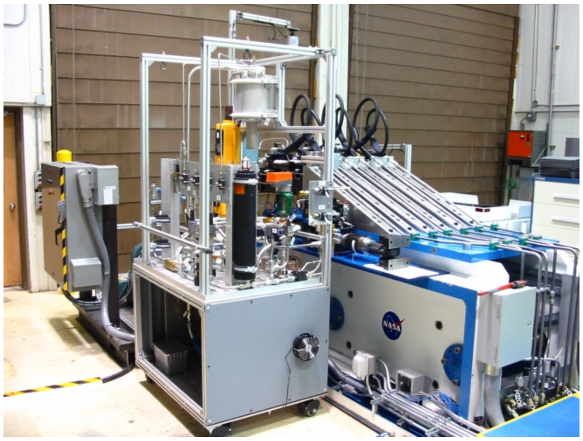

Fig. 5: Breadboard hydraulic system
and 1-axis test rig at NASA Glenn.

Some of the key findings from the testing included that the system performance was not significantly affected by: the location of the accumulator relative to the actuators or the hydraulic pump, accumulator size, fluid operating temperature, filter size, pump discharge check valve cracking pressure, and pump speed when tested over the ranges expected for the flight design. Testing also provided valuable insight as to performance variation as the amount of entrained air in the hydraulic fluid increased which allowed flight limits to be established. Lastly, testing provided characteristics of transient pump heat rejection under flight simulation, allowing rough approximation of system thermal behavior during flight.

The breadboard TPA that was used was the Delta IV version with the lubrication pump added and the hydraulic pump compensator tuned to run at 3000 psi. It did not have the other modifications that would come out of the Advanced Development program. This TPA ran very well and lasted over 20,000 sec, which provided some confidence that it could meet the 7500-sec requirement.

Engineering Model Testing

After the PDR, the TVC team focused on procuring and assembling an engineering model TVC system. This system consisted of a full flight-like system including hardware from the advanced development contracts, hardware procured from hydraulic component vendors using flight-like specifications, and cable harnesses and secondary structure hardware designed and built at NASA Glenn. The goal was to test this system in a flight-like simulation and use that data to prove the design met the system requirements to support the Critical Design Review (CDR).

To best simulate the flight conditions, NASA Glenn designed a 2-axis test rig. Typical 2-axis test rigs only simulate the inertial mass of the engine. The NASA Glenn 2-axis rig was an 80,000-lb rig that included the base, a full size simulation of the Ares I US thrust cone where the TVC hardware was mounted, and an inertial mass simulator for the J–2X engine. In addition to the engine mass, the 2-axis rig also simulated both viscous and Coulombic friction in the gimbal joint, J–2X/thrust cone flexure, actuator attachment stiffness, and J–2X fuel duct spring forces, center of gravity, and thrust offset. The components of the 2-axis rig were manufactured and delivered to NASA Glenn where it was assembled and checked out over the course of several months. Then the entire TVC system engineering model hardware was assembled on the thrust cone in the flight layout as seen in Figure 6. The test rig included a helium conditioning rack that delivered helium to the TPAs at temperatures and pressures simulating both the energy of the helium startup conditions and hydrogen operating conditions. The entire 2-axis rig with some of the support hardware is shown in Figure 7.

Fig. 6: TVC engineering model hardware
mounted on 2-axis rig.

Fig. 7: The entire 2-axis rig with support hardware.

Testing began in July 2010 and concluded in June 2011. The purpose of the testing was to run the TVC system in a variety of nominal and off-nominal conditions and assess its performance compared to the system requirements. Lastly, this data would be used to validate the TVC system model.

The results of the testing demonstrated that the NASA Glenn-designed TVC system for Ares I US performed very well in both nominal and off-nominal conditions. The data also showed that the TVC system model predicted the observed behavior quite well. One anomaly in the testing was the performance of the TPAs. These TPAs had all the modifications from the advanced development contract. The TPAs that were tested did not consistently meet the 7500-sec life requirement. At the time of the writing of this paper an investigation was underway to determine what may be causing the variation in the TPA performance life.

ELECTRICAL POWER SYSTEM

The purpose of the Ares I US electrical power system (EPS) was to provide the power, distribution, control, and monitoring functions for the Ares I US. The EPS architecture experienced several design cycles to optimize and assure the power needs of all the US electric loads assigned to the EPS were met. This system was designed to provide safeguarded, reliable, redundant electrical power to the US avionics subsystems, the US engine control unit (ECU), the US reaction control system (ReCS), the first stage roll control system (RoCS), the TVC system, MPS including the LOX and LH2 recirculation pumps, and the range safety system C-band transponder (CBT). To accomplish this, the EPS would provide an onboard energy source to power the loads from the time ground power was removed until the end of mission. The EPS was designed to protect both the system and loads from electrical hazards, and provide means for controlling and monitoring system performance. The US EPS was capable of receiving power from ground systems through an umbilical interface to support ground operations and system checkouts.

The EPS was designed to provide power compatible with the requirements of the Constellation Power Quality Specification. The EPS architecture was distributed within the Ares I US and located in the instrument unit (IU), aft skirt (AS), and interstage (IS) portions of the vehicle as seen in Figure 12. Simplified block diagrams of the system can be seen in Figures 8 to 10 with detailed descriptions and acronym definitions provided in the following System Description sections.

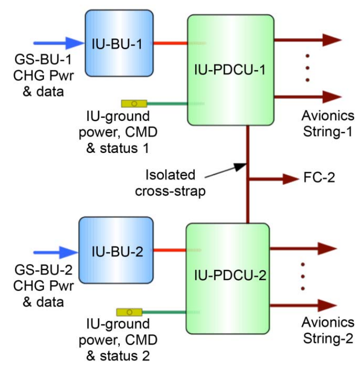

Fig. 8: Upper stage instrument unit (IU) EPS architecture.

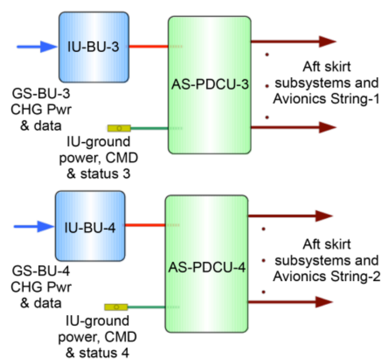

Fig. 9: Upper stage aft skirt (AS) EPS architecture.

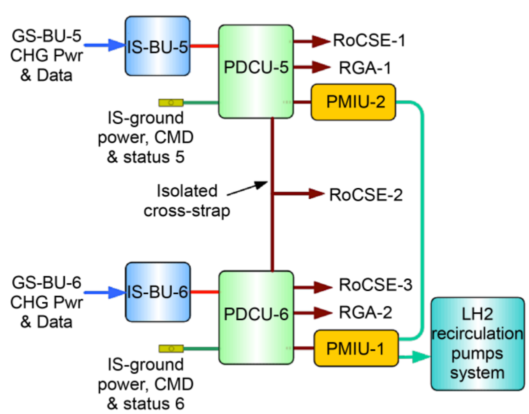

Fig. 10: Upper stage interstage (IS) EPS architecture.

Extensive trade studies and laboratory experiments were performed at NASA Glenn of several different size, voltage, and configurations of batteries as seen in Figure 11. The EPS team determined that lithium-ion batteries would provide a better solution, lower weight, re-chargeability, and reduced operations costs over traditional silver-zinc batteries. A hazard analysis and human safety report supported the choice of lithium-ion battery use, which were also to be used on the Orion crew capsule. The EPS team also developed, built, and tested a breadboard pump motor inverter unit (PMIU) to help guide the development of the US PMIU.

The Ares I US EPS team primarily consisted of NASA Glenn engineers but also included staff from NASA Kennedy Space Center (KSC), NASA Marshall Space Flight Center (MSFC), and various prime and subcontractors. The EPS team was responsible for system requirements definition, analysis, and trade studies leading to an overall architecture as well as procurement approach and selection of vendors. Much of what this team accomplished is usable and being considered for deployment on NASA's Space Launch System (SLS) vehicle.

<u>System Description</u>

The four major components of the EPS were the battery unit (BU) line replaceable unit (LRU), the power distribution and control unit (PDCU) LRU, PMIU LRU, and the interconnecting cable harnesses. The EPS was a distributed architecture with LRUs located in three different locations on the US vehicle: the IU (Figure 8), the AS (Figure 9), and the IS (Figure 10). Each EPS was electrically isolated from the others; this means that there were no power cables going across the IU, AS, and IS. The distributed architecture significantly reduced the mass of the power cables to the loads and reduced the system tunnel's size requirements, while providing a flexible architecture that met the vehicle's redundancy requirements.

<u>Battery Unit</u>

The BU was the prime source of power for the abovementioned Ares I US subsystems, from the time ground service (GS) power was removed until the end of the mission. The BU consisted of a battery module, a battery module charging interface with a deadface relay, a fuse, a local data acquisition and control (LDAC) card, and its sensors. The battery module contained a specific number of cells, electrically configured in series and/or in parallel, designed to meet the required DC voltage range and energy demand profile of the mission with sufficient energy margin.

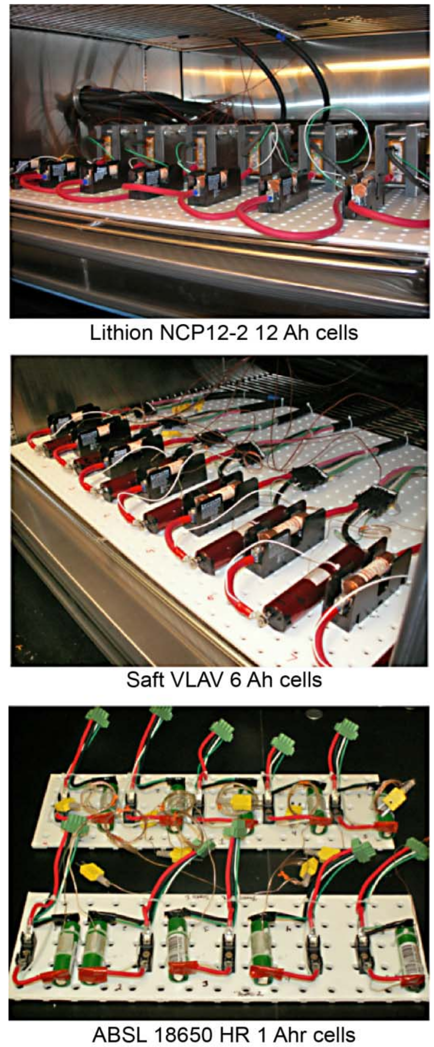

Fig. 11: Examples of EPS batteries tested.

Lithium-ion cells were selected as the baseline for the final battery module design for the EPS BUs. The battery module charging interface and the BU monitoring data provided by the BU-LDAC was used to recharge the BU battery modules in case of battery self-discharge, prelaunch activities, or a launch scrub. The fuse was used to protect the battery module in case of an over-current. The deadfaced relay was to prevent voltages from appearing at the umbilical plate during flight.

The LDAC was responsible for collecting and processing sensor data to monitor the health and state of charge of the battery module. The data was to consist of cell voltages and internal temperatures, and the data was to be sent to the GS to monitor battery module performance and cell state of charge thru the battery charge discharge equipment (BCDE). The LDAC was powered during prelaunch by GS power (from the BCDE), and not powered in flight.

Because commonality, within engineering reason, was encouraged in Ares I, the EPS employed the same battery design across the IU, AS, and IS EPS power strings. This was due to the similar amp-hour (A-H) consumption between the IU and the AS EPS power. Although the A-H consumption numbers in the IS differed from the IU and AS, it was not significant enough to design a different BU. Also a common battery in the IS did not impose a significant mass penalty on the US because it detached from the US prior to J–2X engine start. The BU was specifically designed to provide required power quality voltage range to the loads with sufficient capacity to reliably supply the mission load profile. To achieve the best possible battery module design, the optimal cell size and voltage was chosen.

Each of the three US volumes used a pair of busses to satisfy the EPS single-fault-tolerant (1FT) requirement. In the IU, the BU was sized to supply the energy required for one of the two busses, assuming the second bus had failed, and also the power required to power the third power string for flight computer (FC)-2. The AS loads were the largest in the vehicle and actually sized the batteries. Again, each battery was sized to provide the worst-case power that would be required from the bus, assuming failure of the other bus. Like the IU, the IS batteries were designed to supply the loads for a complete string of loads, plus provide a redundant power source for the Rate Gyro Assembly (RGA)-2 and Roll Control System Electronics (RoCSE)-2.

<u>Power Distribution and Control Unit (PDCU)</u>

The PDCU distributed nominal power from the two core sources, the BU and the GS DC power supply, to all subsystem electric loads. In addition, the PDCU provided status data, event data, and command communication with the assigned flight computers as well as turned power on or off to any of the subsystems loads. It provided proper power sequencing and test and verification operations, protected against any voltage drops specified by the Power Quality Specification and any load over-current or fault conditions. It protected subsystem load cables against overheating or fire and provided single point ground (SPG) connection for the entire EPS. The PDCU-LRU consisted of a DC power transfer switch, remote power controllers (RPCs), housekeeping power supplies (HK-PSs), MIL-STD-1553B communication bus cards, and LDACs. The PDCU DC power transfer switch was a make-before-break-type switch that selected between the two sources of power as commanded by the GS. These sources of power were the GS DC power supply (external to the US), and the onboard battery (BU). The selection between power sources was only to be performed during prelaunch for checkup and test operations. Once the mission had started, the DC power transfer switch was to be set only to the BU power source.

The RPC functioned to turn power on and off to any of the subsystems electrical loads when commanded by the FC or portable non-flight computer via MIL-STD-1553B and LDAC. Also, the RPCs were responsible for protecting against downstream electrical load over-current or fault conditions, thus inherently protecting the EPS from any voltage dropouts and maintaining power quality. The RPCs were solid state current limiting power controllers (SSPCs). The ratings of the SSPCs were determined once the load requirements and EPS specifications were finalized. The RPC used to feed power to a FC could only be turned off through a discrete signal issued from the other FCs through majority voting.

The PDCU provided event and status data to respective FCs via MIL-STD-1553B and LDAC electronic cards. It also received any commands originated by the FC or portable non-flight computer to power any of the RPCs on or off. The LDAC was responsible for gathering and filtering all PDCU measurement data and distributing FC commands to the corresponding PDCU elements (e.g., RPCs) and an on/off command to the PMIU.

The PDCU MIL-STD-1553B and LDAC was also the data communication and command hub for the PMIU, collecting, filtering, and transmitting the inverter data to the FC. The PDCUs in the IU and the IS also contained DC-DC converters. These converters were used to isolate the power so that they could be paralleled with a DC-DC converter in the second PDCU used in each volume to create a third independent redundant power channel to power loads such as the third FC in the IU or the third RGA in the IS. HK-PS, powered directly off the PDCU bus was to convert the nominal voltage to regulated voltages required by the MIL-STD-1553B card, the LDAC card and the RPC bias voltages.

As required by the Constellation Program Electromagnetic Environmental Effects (E3) Requirements Document and the Power Quality Specification, all Constellation electrical power systems needed to be designed to incorporate a distributed SPG. Therefore, the PDCU was to provide an SPG connection. This was an external and removable connection to facilitate test and verification prior to launch. There were as many SPG connections as power busses. Three different PDCU types were implemented across the US. The type I PDCU was used in the IU, the Type II in the AS, and Type III in the IS. The IU PDCU (Type I) managed a single power string, sourced from either the GS or the BU, and distributed power to a set of IU avionics loads. The Type I PDCU internally also powered an isolated DC to DC converter to produce a third power string that energized FC-2. This was done in tandem with both IU-PDCUs to provide redundant and isolated power to FC-2.

The IU PDCU also contained a discrete interface for PDCU-1. The discrete interface was used to shut off power to an FC by turning off the RPC when commanded by the other two FCs. A similar circuit was in PDCU-2 to turn off power to FC-3 when commanded by FC-1 and FC-2. This FC shutdown function was to be executed any time during the mission in the event of a FC malfunction.

The AS PDCU (Type II) managed a single power string, sourced from either the GS or the BU, and distributed power to all the AS electrical loads. Unlike the IU PDCU, the AS PDCU had no additional design features such as the DC-DC converters or the discrete interface.

The IS PDCU (Type III) was similar to the IU PDCU type in that it provided redundant power, but to three loads whereas the IU only had one redundant power source. However, it did not have the discrete shutdowns used in the IU PDCU. Another distinction in the IS EPS architecture was that GS power that source the PDCU was routed via the first stage element T-0 connector.

Pump Motor Inverter Unit (PMIU)

The PMIU was required to provide phase power to the induction motor used for the MPS LH2 recirculation pump. The upper stage PMIU was fed input power by the PDCU.

Each PMIU LRU was sized and configured to power one pump motor. In the event of a power system or PMIU failure, a redundant PMIU powered by another power bus was available to take over and continuously power the LH2 pump motor between liftoff and US J–2X main engine start. The PMIUs performed output power current limiting to protect against motor starting transients. The switching between the redundant PMIUs was provided within the PMIUs, controlled by the FCs through the PDCUs.

The PMIU consisted of an input filter and a DC-DC converter to boost the power to a nominal voltage required. The DC input filter maintained power quality by filtering the frequency and its harmonics components produced by the inverter switching scheme and the switching noise from the DC-DC converter. The output filter attenuated the inverter high frequency switching noise produced by the high dv/dt of the inverter switching scheme.

The HK-PS card, powered by the nominal voltage value bus, converted the nominal voltage to the regulated voltage required by the LDAC card and the inverter control and drivers' bias voltages.

The PMIU LDAC electronic cards provided constant event and status data to their respective PDCUs at a specific data rate. It also received any power on or off commands originated by the FC or portable non-flight computer. Two PMIUs were employed to provide 1FT AC power. The PMIUs were located in the IS so that their mass was not carried to orbit because the PMIUs were only required before launch and during the first stage burn. Also, because the LH2 recirculation pump motor was located at the US LH2 tanks, the AC power from the PMIU was required to cross the IS and AS boundary. This was the only instance in which any kind of high power from the EPS crossed volumes during the Ares I mission.

Fault Tolerance (FT) for Catastrophic Hazards

The US Element Requirements Document required the EPS to be no less than 1FT for hazards that led to a catastrophic event. This EPS architecture design satisfied that requirement via the design approaches used in each of the US locations.

The IU, AS, and IS used two power strings to provide 1FT capability. Note that the IU and the IS derived third power strings, using DC-DC converters to power FC-2, RoCSE-2 and RGA-2. Two PMIUs in the IS were implemented to provide 1FT AC power to the LH2 recirculation pump motor.

PURGE AND HAZARDOUS GAS DETECTION

Objectives and Requirements

NASA Glenn led the design and development of the Ares I US Purge and Hazardous Gas (P&HG) Detection subsystem and was thus responsible for implementing the design, development, test, and engineering for the subsystem.

The role of the purge system was to provide an inert, dry, clean, thermally conditioned gas flow to three Ares I US compartments, the IU, AS, and the IS (Figure 12).

The role of the three independent Hazardous Gas Detection Systems (HGDS) was to provide the capability to sample, detect, and measure concentration of hazardous gases in the Ares I US compartments prior to launch.

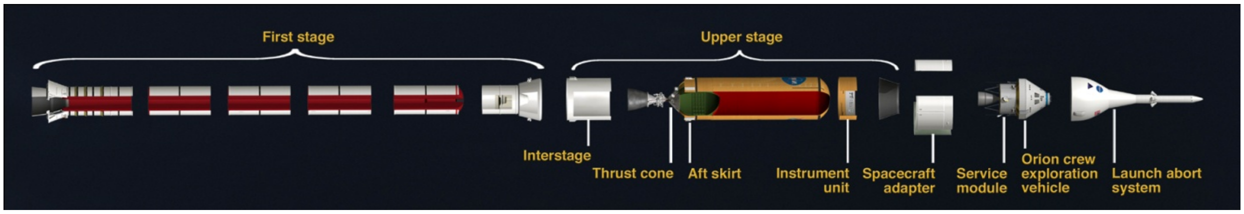

Fig. 12: Overall vehicle configuration (expanded view).

The P&HG project team interfaced with the NASA MSFC Program Office, which was responsible for the overall development and integration of the Ares I US and the Ares I US Engine. The P&HG team also interfaced with NASA KSC, which was responsible for the Constellation Launch Control Systems including vehicle umbilical interfaces and ground support equipment.

Purging is a design solution to actively provide an environmentally controlled atmosphere within a launch vehicle or spacecraft payload prior to liftoff.

Purge operations for Ares I would start upon shipment of the vehicle from the Michoud Assembly Facility and continue almost without interruption until launch at T-0. Purges provide ventilation and maintain internal vehicle temperatures by supplying heated or cooled purge gases depending on weather conditions and solar insolation. By maintaining the vehicle at a positive internal pressure relative to the external atmosphere, purges prevent contaminants and atmospheric gasses from entering through leaks in the structure and vent doors.

The P&HG subsystem performed a safety-critical function in the rocket upper stage prior to launch. Launch vehicles that utilize LH2 and LO2 cryogens as propellants purge nitrogen into compartment dry volumes when liquid fuels are present. Dry nitrogen displaces atmospheric gases, hazardous gases, and water vapor in the compartment to prevent liquefaction, eliminate explosion hazards, and keep water vapor from condensing into liquid water leading to the buildup of ice. In cases where LH2 or its vapor from boil-off may cause liquefaction, helium purges are used instead of nitrogen. Propellants, such as hydrazine and nitrogen tetroxide, need to be kept from freezing on cold days while the thruster systems that use these fuels must be maintained within temperature limits.

The P&HG subsystem was designed to meet two key driving requirements for the vehicle. First, prevent an accumulation of hazardous gases. This is a safety-related requirement for preventing hazardous conditions, such as fires or explosive environments, within the US dry bay volumes. Second, provide an interface to the ground support, for the detection of hazardous gases, prior to launch. This is also a safety-related requirement for detecting system leakage and verifying no accumulation of hazardous gases that could lead to fire or explosion on the launch pad.

P&HG System Description

P&HG systems are heritage systems taking their architectures from the Space Shuttle and Saturn V vehicles. All P&HG systems are operational and considered to be safety critical before launch until T-0, but not during launch or flight.

The P&HG system operates in the three major US compartments, the IU, AS, and IS. The purge system promotes mixing of gaseous components in the US compartments to prevent potentially harmful gases from accumulating in hazardous concentrations. The HGDS detects the accumulation of hydrogen and oxygen gases.

At T-0 and during flight, the P&HG system has no safety-critical function. Instead, the system must be capable of withstanding the random loads of the launch and flight vibroacoustic environment without breaking apart and endangering other vehicle components or systems.

Purge System

The role of the purge system was to provide an inert, dry, clean, thermally conditioned gas flow to three Ares I US compartments: the IU, the AS, and the IS. The purge flow also provided thermal cooling for the avionics boxes in the US compartments. Each compartment had its own dedicated purge supply through the T-0 umbilical interface panel, quick disconnects. The purge flow had sufficient velocity to mix with any hazardous gases, such as hydrogen; this mixing reduced the concentration of hazardous gas to safe levels, and pushed the mixture out of the compartment through the vent doors. In addition, the purge flow passively cooled these compartments and any avionics boxes. (Fig. 13, IU purge duct; Fig. 14, AS purge duct; Fig. 15, IS purge duct).

A conditioned air purge flowed from the KSC Environmental Control System (ECS), through umbilical panels in each compartment, the purge ducting, and out through orifice holes into the IU, AS, and IS compartments. The purge occurred during qualification testing, Ares I checkout and testing at the Vehicle Assembly Building, rollout/rollback, and at the launch pad. Purge flow was driven by a pressure of

0.5 to 0.7 psid above ambient pressure. A gaseous nitrogen purge flowed to each Ares I compartment, starting 2 to 6 hours prior to cryogenic tanking and this nitrogen purge flow was maintained up to liftoff at T-0. Although the purge system did not operate during the flight phase, it was sized to precondition the vehicle compartments to maintain a safe environment through the ascent phase of the launch.

Within the IU, the purge system routes purge gas to the IU volume and the forward end of the systems tunnel (ST). Within the US, the purge system routes purge gas to the AS volume, the thrust cone (TC) interior volume, the aft end of the systems tunnel, the helium pressurization line fairing, ReCS modules, the LH2 feed line fairing, and the LH2 fill and drain line fairing. Within the IS, the purge system routes purge gas to the IS volume and the RoCS modules.

The purge system was a low-pressure ducting system that carried thermally conditioned air to US compartments and their avionics prior to US fueling. Nominally, the nitrogen purge flow rate was sized for one compartment volume exchanged every 4 min, with a maximum back pressure at the inlet of 0.5 psig and temperature between 57 to 87 °F. The purge system operated at less than 0.5 psi above ambient.

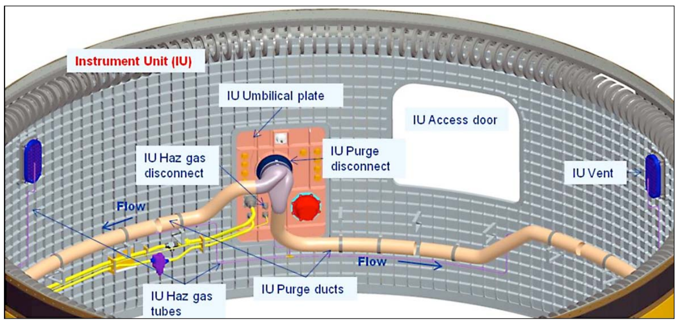

Fig. 13: Instrument unit with purge duct.

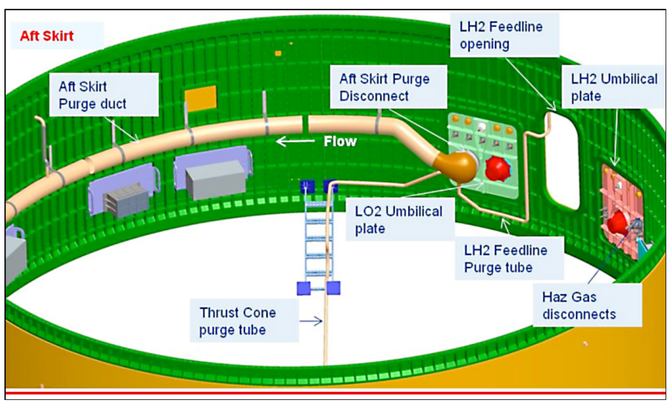

Fig. 14: Aft skirt with purge duct.

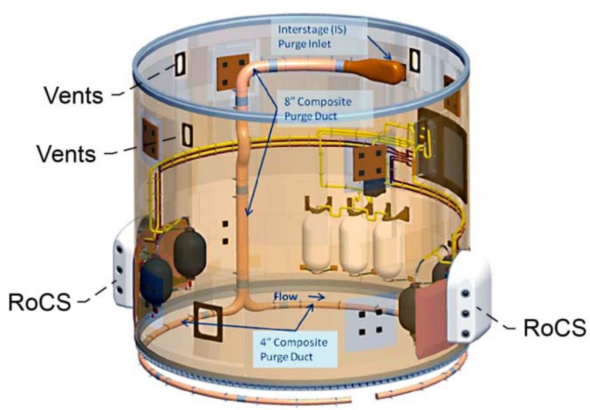

Fig. 15: Interstage purge duct.

Purge Duct Materials Selection

Initially, the purge ducts for aerodynamic testing were aluminum. Trade studies indicated composite ducts would have a 50% mass savings and 60% cost savings (with volume production) over aluminum. Also, composites ducts in complex shapes are more easily manufactured. Although cloth ducts had a 60% mass and 80% cost savings over aluminum, the purge ducts were required to thermally condition avionics boxes by directed jets. Cloth ducts were too flexible to maintain a consistently repeatable jet and cooling pattern.

The Shuttle purge ducts were fabricated from Aramid/epoxy with an electrostatic discharge (ESD) coating painted on the exterior surface of the ducts to remove deposited static charge. Records for Shuttle purge ducts indicated cracking damage as well as separation of the ESD coating. Development proceeded with Aramid/epoxy as the baseline material with a focus on increased material toughness and better ESD coating adherence. The Shuttle Aramid/epoxy was not readily available, and contemporary products were sought. ESD requirements were achieved by using a conductive film as the innermost layer of the composite layup, co-curing the layers together. Testing verified that the co-curred materials met NASA's flammability and toxicity requirements.

Composite Materials and Purge Duct Construction

The purge ducting was designed to have three plies of Aramid/Epoxy, (0.030 in. wall thickness), Kevlar®49, weave style 285.

The orientation of these three plies was not easily controlled given the nontrivial, tubular shape of the purge ducts. Consequently the research, analysis, and data available for idealized orientations did not perfectly match the purge duct layup. During layup, plies would be required to overlap except for an exclusion zone, 3 in. long, at the ends of each duct. Disallowing overlap in this exclusion zone would help to ensure the duct would fit inside the connector hose. Orifice holes were to be drilled in the composite purge ducts to allow the purge gas to flow into the compartment at high speed.

The design included an additional ply of electrically conductive surfacing film added on the inner surface of the ducts. This layer gives the duct sufficient conductivity—a requirement for nonmetallic tubing (NASA-STD-4003)—to remove any electrostatic charge. By adding this ply during layup and co-curing with the Aramid/Epoxy, fabrication is simplified and superior bonding of the conductive coating is achieved when compared with legacy operations of the Shuttle.

Hazardous Gas Detection System

The three independent HGDSs provided the capability to sample, detect, and measure concentration of hazardous gases in the Ares I US compartments prior to launch. The hazardous gas sampling locations included the IU vent locations, the AS thermal blanket opening, and the IS vent locations. This system did not operate during the Ares I flight phases.

The hazardous gas designs were based on the existing Space Shuttle system with the exception that the mass spectrometers would be located on the gantry launch tower structure near the umbilical interface quick disconnect panels to minimize the diagnostic time lag associated with ground-based mass spectrometers (due to the lengthy tubing arrangement).

The HGDS was an open-ended tube system. The design used pneumatic tubing and a slight vacuum (~1.5 psi below ambient) to transport gases from each US compartment to ground-based mass spectrometers located on the gantry launch tower at each compartment level for gas analyses. The compartment gases were sampled continuously until liftoff at T-0. All components associated with the HGDS were designed to promote flow.

Gas samples would be taken near the compartment vent locations and transported through a single 0.25-in.-O.D. stainless steel sampling tube (0.028 in. wall thickness), exiting the launch vehicle through the ¼-in. hazardous gas quick disconnect interface. The mass spectrometers and vacuum system were the responsibility of the KSC ground systems element. Compartment gases were to be sampled continuously, but analysis was to be performed on a rotating basis. At launch, the HGDS was disconnected from the gas analyzers.

Detection of hazardous gases was required in critical compartment volumes where hazardous gases could accumulate during ground operations. Hazardous conditions include flammable or explosive concentrations of hydrogen and oxygen.

The HGDS would have taken gas samples within the IU at the IU vents, within the AS at the thermal blanket opening near the J–2X engine gimbal and within the IS at the IS vents. (Fig. 16, IU Unit HazGas schematic; Fig. 17, AS HazGas schematic; Fig. 18, IS HazGas schematic).

The HGDS was a tube system that carried gas samples from each compartment to mass spectrometers located on the gantry launch tower at each compartment level.

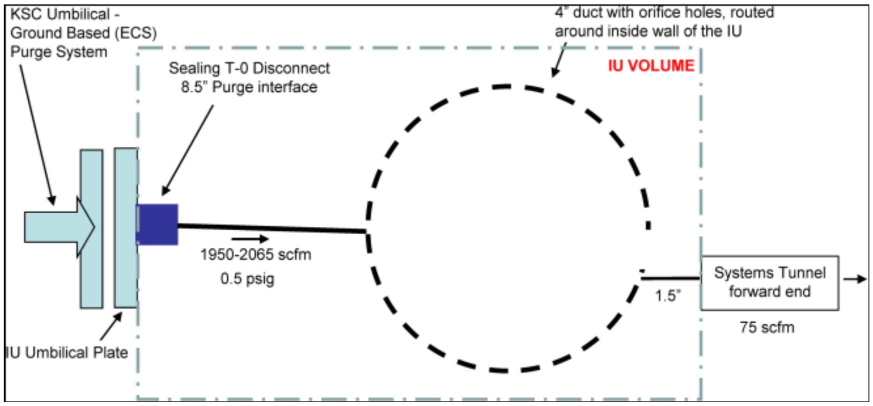

Fig. 16: Instrument Unit HazGas schematic.

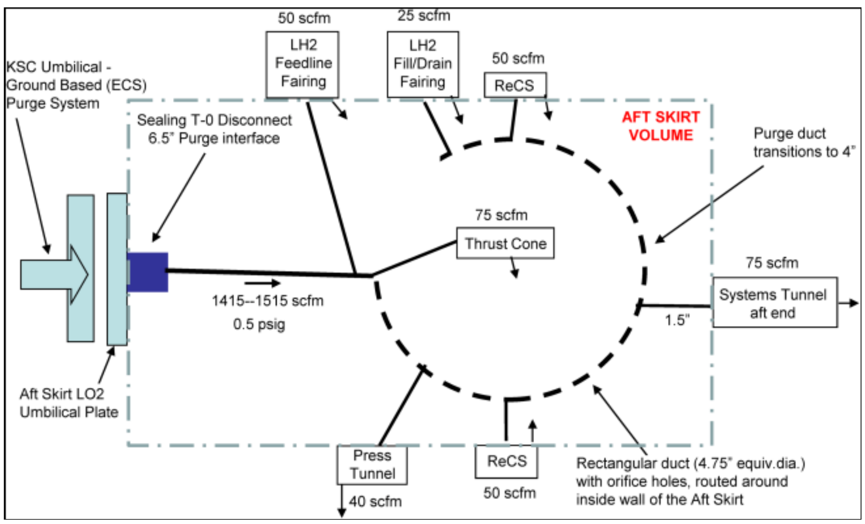

Fig. 17: Aft Skirt HazGas schematic.

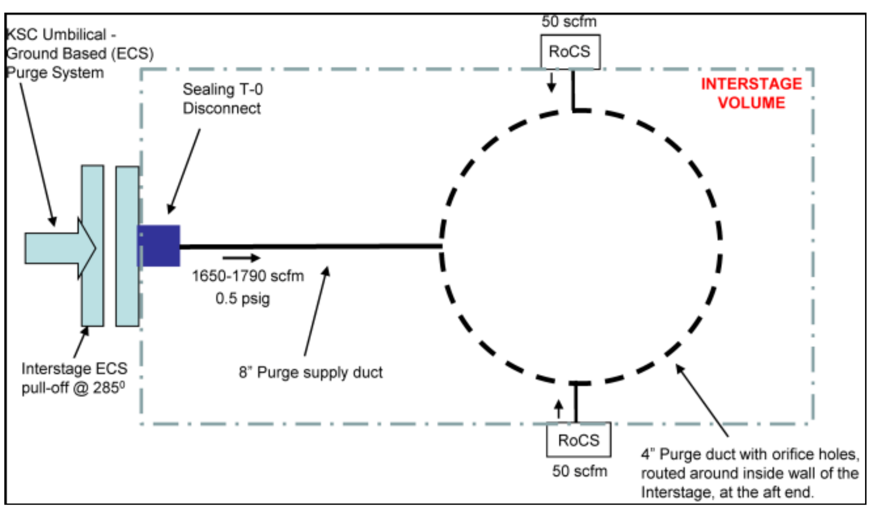

Fig. 18: Interstage HazGas schematic.

DEVELOPMENT FLIGHT INSTRUMENTATION

The “unknown unknowns” are the surprises even the best engineers and spacecraft designers cannot plan for. Thus, new spacecraft, especially those intended to fly humans, include a Developmental Flight Instrumentation (DFI) system which helps to identify the unexpected as it occurs.

On the Ares I, a DFI system was to be installed on the first five flights to monitor vehicle behavior during the initial flights for the purpose of attaining data on unexpected events as well as providing correlation of test flight data to earlier modeled data. As its purpose was early detection leading to elimination of issues, the US DFI was designed to be an independent package that would be installed as a standalone system and later removed without impact to permanent design systems. DFI was self-powered with an independent telemetry system, harnessing, and sensor excitation source with designed-in ease of removal after DFI missions were completed.

The Ares I US DFI system was conceived from the beginning to use commercial off-the-shelf (COTS) hardware to meet affordability and ease of change goals and was distributed within the vehicle in the IU and AS. A simplified block diagram of this system is seen in Figure 19. In addition to overall approach and system design, the Ares I US DFI team wrote the component end item (CEI) specifications and was granted approval for use in the development and eventual procurements of the system components. Verification and validation requirements as well as Systems Engineering and Management Plans were also created by this team. Contracts are in place for the Data Acquisition Units, Battery, and Power Distribution Units with some hardware delivered and tested to date. An engineering development unit (EDU) of the system (Figures 20 and 21) was built and tested at NASA Glenn and is still being studied in anticipation of its potential use on NASA’s currently planned SLS vehicle. The EDU included a Master Data Acquisition Unit (MDAU) and workstation that has been tested as well as forty sensors and 33 NASA GRC fabricated cable/ harnesses used in the sensor simulations. A Remote Data Acquisition Unit (RDAU) is currently being tested as of this writing with a second RDAU ordered for further testing. A BU has been procured and planned for testing later this year.

Fig. 20: Ares I US DFI MDAU, workstation, and sensors.

Fig. 21: Ares I US DFI sensor simulator.

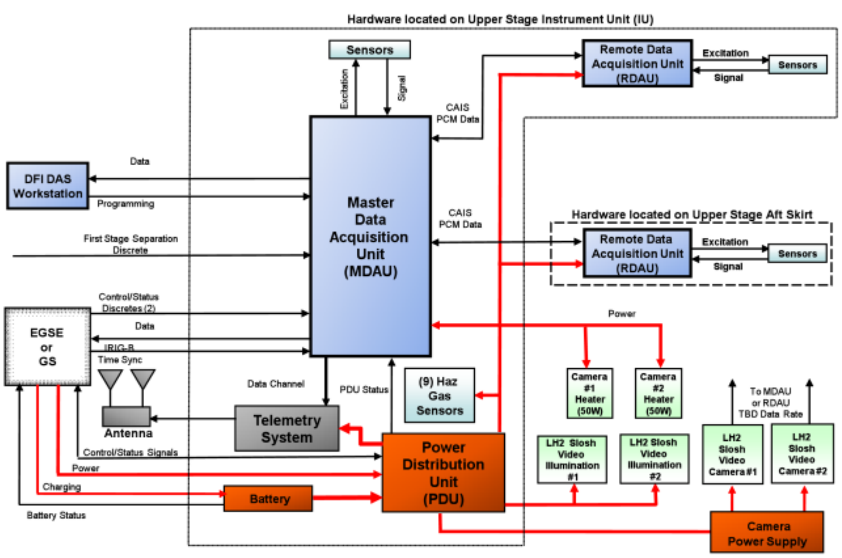

Fig. 19: Ares I US DFI System Block Diagram.

The Ares I US DFI is considered at the readiness level for a CDR for Ares I and is currently being studied for deployment on the SLS because of its robustness, versatility, affordability, and ability to use existing hardware and procurement mechanisms. The DFI team primarily consisted of engineers from NASA Glenn but also included contributions from NASA KSC and NASA MSFC personnel. In total, the team included over 20 individuals working part to full time as well as personnel from four outside contractors.

System Description

The Ares I US DFI system was a distributed architecture that consisted of an MDAU and two RDAUs located at various locations on the US structure to minimize long cable runs, a power distribution system, a battery, and a telemetry system. The MDAU and two RDAUs accommodated data from a variety of sensors. This system collects, manages and processes various types of sensor and instrumentation. Included are analog, digital, discrete, and other additional data transferred from sensors and instrumentation located across the US to monitor the subsystems and video data of LH2 slosh activity. This information was then telemetered to the Ground System Network. The DFI system provided its own power and downlink data to the ground via its own telemetry system and received ground power prior to launch.

DFI System Components

The Ares I US DFI MDAUs collected data which included sensors and instrumentation connected directly to it as well as data acquired by each of the two RDAUs. The MDAU was an LRU. Each LRU was replaceable at any processing facility for the Ares I vehicle or at the launch pad.

The Ares I US DFI RDAUs collected, managed, processed, and acquired various types of sensor, instrumentation, and video data, including analog, digital, and other additional data transferred from sensors and instrumentation located across the US to monitor the subsystems. Each RDAU transferred its acquired data to the MDAU for telemetry to ground. Each RDAU was a LRU.

The Ares I US DFI Power Distribution Unit (PDU) distributed and switched unregulated battery power from either the onboard DFI battery or GSE power supply. The PDU was one individual unit and a LRU.

The Ares I US DFI lithium-ion battery provided power to each DFI data acquisition unit, telemetry system, LH2 tank heater, and illumination lamp for the LH2 tank video cameras. The battery was one individual unit and a LRU.

The Ares I US Telemetry System consisted of three major components: a transmitter, two S-band antennae, and an S-band coaxial cable. Each of the components was a LRU.

The Ares I US DFI system supplied all sensors that were necessary to characterize the Ares I US not included in the operational flight instrumentation. The DFI MDAU or RDAU supplied sensor excitation and received data from each sensor. Sensor types included K-type thermocouples, pressure gauges, strain gauges, heat flux transducers, calorimeters, accelerometers, microphones, position sensors, and hazardous gas sensors.

The DFI system supplied all power, data and sensor cabling necessary for the DFI system integration into the Ares I vehicle.

SUMMARY AND CONCLUSIONS

NASA is in the process of transitioning the efforts of the Constellation Program as well as the Shuttle Program and its assets to the SLS. The Ares I upper stage subsystems developed at NASA Glenn have significant utilization potential on SLS. The current SLS approach is to initially "fly out" existing NASA assets. Shuttle TVC systems will be used but will need to be replaced with new designs after several flights. The EPS and DFI systems developed are flexible and can be reconfigured to any launch vehicle architecture. Knowledge gained in cooling avionics systems with purge air and how to handle common shared volumes will also be transferred to SLS. Other significant efforts from Ares I may also potentially transfer to the SLS program.

ACKNOWLEDGMENTS

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