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Lessons learned on design

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1. Introduction

Future spacecraft designers and managers need to be aware of problems, corrective actions, and the resulting lessons learned to avoid experiencing the same problems in new programs. Fewer and fewer people with firsthand experience of the design, test, and operations of past programs, such as Apollo, are available today to pass on their experience.

I worked on all the major human spaceflight programs beginning with Apollo. I started my career in 1964 at NASA's Manned Spacecraft Center (MSC), which is now Johnson Space Center (JSC), in the Engineering and Development Directorate, transferring to the Mission Operations Directorate, and later in the Safety, Reliability, and Quality Assurance (SR&QA) Directorate, later called the SMA Directorate. I retired from NASA in 2006, but returned as a consultant working for J&P Technologies, supporting the NASA JSC Safety & Mission Assurance Flight Safety office (FSO).

I hope this paper will communicate valuable experience to younger engineers, so they can continue to build on the lessons of the past to create even better human spaceflight programs. These selected articles are from JS-2018-009 JSC Flight Safety Office Gary Johnson: Lessons Learned from 50 + Years in Human Spaceflight and safety.

2. Redundancy can help or hinder

At the beginning of any spacecraft program the reliability goals and the redundancy philosophy for safety should be established. On the Apollo Program, the reliability goals were identified for each function to be performed. For example, the reliability apportionment established for the Service Module (SM) post-separation from the Command Module (CM) was 0.999985 (or 15 failures/10-to-the-6 missions, the reliability terminology used in 1965). If the function was safety-critical, the design should have no single-point failures and should be fail-safe. The amount of subsystem redundancy was determined by the criticality, flight experience, and maturity of technology. As a result the redundancy varied. For example, fuel cells were a new technology for aerospace applications, while the electrical power buses and power contactors for switching were proven technology. Therefore, the redundancy level was three fuel cells and two main buses. For the alternating current (AC) power system the solid state inverters were a new technology for aerospace, and three inverters with two AC buses was the redundancy level. One inverter was always offline as a spare.

2.1. Apollo 10 (May 1969): fuel cell failure

After docking with the Command and Service Module (CSM) and jettisoning the LM while the CSM was in lunar orbit, a caution and warning alarm sounded, and the Fuel Cell 1 AC circuit breaker tripped, due to a short in the hydrogen pump, causing the loss of Fuel Cell 1. The CM pilot told the commander he thought another one would go out as soon as they got to the back side of the moon. Halfway through the next night side pass, an alarm occurred on Fuel Cell 2 due to a fluctuation on the condenser exit temperature. With a minor electrical load reduction, Fuel Cell 2 continued to provide power, reference 1.

Lesson: Critical systems should be two-fault tolerant. Consumables, like electrical power, which are required for crew safety should have an additional level of redundancy for missions beyond Earth orbit.

Redundancy saved the crew in this case, since without redundancy in the fuel cells, the crew would not have had enough power to return to Earth. The CSM could safely return with one fuel cell down.

2.2. Apollo 12 (November 1969): lightning strike

During the Apollo 12 launch on November 14, 1969 lightning struck the spacecraft. At 11:22 am, T + 36 s, the crew saw a bright light. At T + 36.5 s many errors occurred: Fuel Cells 1, 2, and 3 disconnected; Main Buses A and B were under-voltage; AC buses 1 and 2 overloaded. The warning lights and alarm came on in the cabin, indicating the failure of the Inertial Stabilization System. At T + 52 s (13,000 feet) lightning struck the vehicle and the Inertial Measurement Unit platform tumbled, see Attachment for location on the chart.

The potential effect on the vehicle was induction into wiring, depending on the location and rate of change of potential and direct current (DC) flow in grounding. The high negative voltage spike (delta voltage/delta time) caused the Silicon Controlled Rectifiers to trip on the fuel cell and AC inverter overload sensors. Failures occurred in four SM Reaction Control System (RCS) helium tank quantity measurements, five thermocouples, and four pressure/temperature transducers.

Using power from the battery relay bus, the crew reconnected the fuel cells to Main Bus A and B, and reconnected the inverters to AC Bus

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1 and 2. The mission continued.

Good structural electrical bonding among the Launch Escape System, CM, SM, Spacecraft LM Adapter, and Saturn V Inertial Unit prevented major damage to the systems and vehicle. The Launch Vehicle Instrument Unit computer provided ascent guidance and control since the CM computer, which was the backup for ascent, stopped working at the lightning strike. Ascent abort sensor processing (Emergency Detection System) was performed by the Instrument Unit avionics. Having the redundant computer for ascent guidance and control located in the launch vehicle instead of the CM saved this mission. CM battery power prevented the loss of all electrical power when fuel cells disconnected from the main buses. This allowed the crew to recover and reset the fuel cell connections to the main buses, reference 2.

Lesson: Redundant systems for critical functions should be dissimilar and/or located in different parts of the vehicle.

2.3. Skylab 3 (July 1973): propellant leak and rescue mission

During the rendezvous on July 28, 1973 Commander Alan Bean saw the first indication of a problem: the attitude was off about 25° in yaw. According to fellow astronaut Owen Garriott, Jack Lousma "suddenly announced, 'Owen, there goes one of our thrusters floating by the window.'"[1]

The CSM RCS Quad B forward-firing positive-yaw engine oxidizer valve had leaked, and the nitrogen tetroxide had frozen into the shape of the thruster exhaust cone. The crew also reported a "snow storm" on the right side (Quad B side) of the spacecraft at the same time. The Quad was isolated, and the rendezvous had to be completed with only three of the four SM Quads. The crew members had not trained for this, but were able to complete the docking to Skylab.

Five days later after docking to Skylab, the Quad D engine package temperature had decreased, causing an alarm. The crew saw a "snow storm" blowing by the window and knew that something was leaking from Skylab. With guidance from the ground, the RCS engines were inhibited, and the isolation valves closed about 1 h and 20 min later. This second oxidizer leak prompted a concern that the oxidizer portion of the SM RCS had a problem.

If this were true the remaining SM RCS Quads could fail, and preparations began at Kennedy Space Center (KSC) for a rescue mission. By eliminating subsystem tests the spacecraft CSM-119 could be mated with its Saturn 1B launch vehicle the next week. Removing storage lockers allowed for two more crew couches to be installed under the existing three, see Fig. 1. Deleting the Countdown Demonstration Test meant the spacecraft and launch vehicle could be ready in early September. The Skylab Multiple Docking Adapter had been designed with a spare radial docking port, in case a rescue spacecraft might have to dock. Commander Vance Brand and Pilot Don Lind were assigned as the rescue crew.

While rescue preparations were underway, Engineering concluded that the SM Quads did not share a common problem. The only condition that would match the timeline of events and leakage rates experienced on Quad D was an improperly torqued Dynatube connector. A contingency procedure was developed so that if another SM RCS Quad failed, the CM RCS could be used to de-orbit and for control during entry. The mission was allowed to continue, and the Skylab rescue capability was available if needed, reference 3.

The innermost and primary seal of a Dynatube connector is a mirror-finished, metal-to-metal seal. The second seal is a butyl O-ring. Tests showed that a finger-tight Dynatube connection would pass a preflight helium leak test. However, when the RCS system is pressurized prior to liftoff, the oxidizer penetrates the unmated primary metallic seal, and the O-rings alone seal the connection. Tests showed the butyl O-ring degrades and can start leaking when exposed to nitrogen tetroxide at flight temperatures and pressures. The butyl O-ring seal was considered a redundant seal, but did not actually provide redundancy, creating a false sense of security.

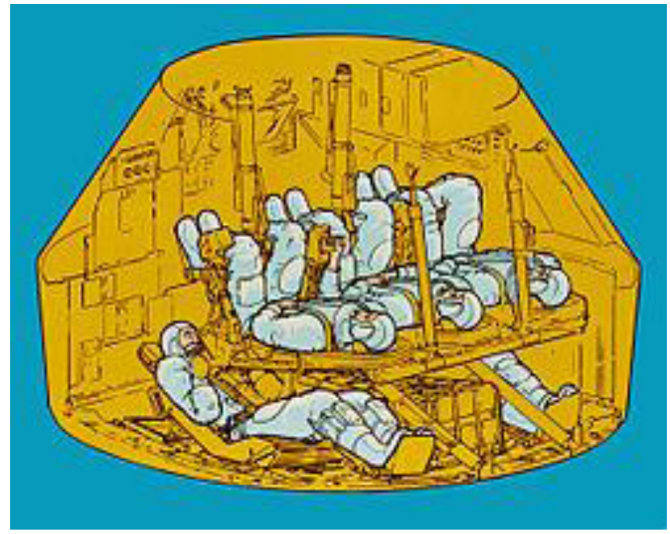


Fig. 1. Skylab 3 rescue mission crew location.

Lesson: Redundant systems must use compatible materials to be effective. Flight-critical fluid connections should have controls in place to insure proper torquing of connections, and seals should be compatible with the fluid they are trying to seal.

The program's early planning to develop a Skylab rescue capability allowed a rescue mission to be developed quickly.

2.4. Space shuttle orbiter electrical system

During the Space Shuttle Program, the orbiter avionics were required to have an additional level of redundancy (three levels) compared with what was required for the Apollo Program (two levels). This was because the new digital avionics system had no proven flight experience. This also applied to other systems, so that launches could continue even if one system failed. (This supported not only safety, but the goal of the shuttle launching repeatedly with a quick turnaround, like an airline.)

The orbiter had three hydraulic systems and was powered by three auxiliary power units, three fuel cells, and three main electrical buses. Early in the design phase the orbiter contractor's manager for electrical systems and the NASA electrical power distribution engineers recommended not to require the extra redundancy (three levels) to the main DC bus system, since it was very reliable and had been flight proven. This recommendation was not adopted, and the orbiter power system was designed with three redundant systems. The additional level of redundancy added cost and complexity to the electrical power distribution systems. While redundancy is often beneficial, increasing the complexity of a system introduces the possibility for more problems and greater risk. Unnecessary redundancy can hinder a program's development by adding cost and introducing new issues.

Lesson: To minimize spacecraft complexity, weight, cost, and schedule the level of redundancy should depend on the criticality, flight experience, and technology maturity of the hardware. Unnecessary redundancy adds complexity, which increases cost and risk.

3. Documenting and sharing information: communication is key

In a complex organization with many teams made up of many people, documentation enables information to be shared. Sharing

[1] Hitt, Garriott, and Kerwin, *Homesteading Space: The Skylab Story*, page 239.

information is key when many groups will be working on different aspects of one project, like a spacecraft. Any modifications to spacecraft design or installation need to be documented and communicated to the other systems and personnel. When significant anomalies happen, the results of formal board investigations should be shared widely to avoid repetition of the same mistakes. Mission reports should include significant mission anomalies and should be updated if a problem comes to light after the report is written. Flight operation changes to a crew checklist system procedure need to be reviewed by the engineers responsible for that spacecraft system. Drawings from a previous spacecraft program or mission must include all engineering changes to that drawing. A spacecraft project engineer, besides understanding his system, should know and work with the design, manufacturing, test, and operations personnel for that system, even though they may work under another organization, vendor, or launch center.

3.1. Apollo-Saturn mission 201 (February 1966): loss of reaction control system

During the uncrewed Apollo-Saturn (AS-201) mission after CSM separation, a short to power occurred causing the RCS commands to be transferred from the CM to the SM, resulting in the loss of RCS control. This transfer of command circuits to the shorted umbilical due to entry heating caused a large short and low voltage on Main Bus A and B till the circuit breakers opened. After the loss of the RCS, the CM went into a stable roll and did a ballistic entry instead of the planned lifting reentry.

The cause of all this was a non-functional circuit that was routed through the CSM umbilical that was not deadfaced prior to separation, and it shorted, tripping the circuit breaker powering the Sequential Events Control Subsystem System B circuit causing loss of the redundant Earth Landing System (ELS). The non-functional circuit had been dropped from the drawings, but the wiring was left in the spacecraft and was not disconnected from power. Since it was not on the drawing for the powered wiring going through the CSM umbilical, it was not deadfaced prior to guillotine of the umbilical.

Redundant NASA Standard Initiators, one powered by System A and one by System B (this source failed), were used on all critical pyrotechnic functions, including the ELS, and this allowed for recovery of the CM, reference 4.

Lesson: Drawings should include all design changes, even disconnected or unused circuits. The lack of documentation in this case meant that the design drawings did not match the spacecraft, which made it more difficult to identify the problem.

Also, all non-functional circuits left in the spacecraft should be disconnected from both power and return. Careful review must be made to ensure all powered wiring through an umbilical to be guillotined must be deadfaced prior to cutting with the guillotine.

3.2. Apollo environmental control system (April 1966): fire

On April 28, 1966 a fire occurred at the AiResearch Torrance Facility in California in the altitude chamber used to simulate the interior environment of the Apollo CM (100% oxygen at five psi). The Apollo Block 1 Environmental Control System (ECS) was undergoing a 500-h mission-duration qualification test. The fire severely damaged the ECS and test setup equipment, but the damage was confined within the test chamber. The test was later repeated successfully and without incident, both with a new set of Block I hardware and then again with an ECS of the Block II configuration.

An MSC Fire Investigation Board was appointed on May 3, 1966 by Dr. Robert R. Gilruth, Director of the MSC, to independently investigate the cause of the fire. At the request of the Apollo Spacecraft Program Office, I was assigned to participate in the fire investigation to determine the ignition cause. I was working at the time as the CSM electrical power distribution project engineer.

I investigated the test setup wiring and the sequence of events leading to the start of the fire and interviewed the AiResearch instrumentation engineer in charge of the test. When the fire occurred, the chamber had been evacuated for several days to simulate a depressed cabin environment, which was part of the test profile, and was being repressurized with oxygen to five psi in preparation for the next test phase. The last change in configuration before the start of the fire was increasing the voltage on the steam duct heater, and 48 s later the chamber dome lifted due to the increased pressure from the fire.

The most probable cause of the fire was a failure of the commercial quality strip heater used to add heat to the steam duct (the line from the suit heat exchanger and glycol cooling evaporator to the vacuum source) to preclude freezing of water in the duct. This duct was wrapped with two strips of the heater tape and overwrapped with asbestos tape. During the interview the instrumentation engineer stated that the electrical technician had purchased the heater tape from Sears. This was the same type used to wrap a house's outside water line to prevent freezing. I was told not to put in my report that the electrical technician had installed Sears heater tape for the duct heater. No analysis had been done to see if the commercial tape wire insulation could withstand the temperature it would be exposed to. The steam duct heater wire had previously shorted outside the dome near a splice like the splice to the heater tape in the dome, where the wire was open and looked like it had arced.

I never saw the official board's report, because it was not distributed even to those that participated in the investigation. I had made a recommendation about a possible ignition source on the test setup wiring, but never found out if my recommendation was included, reference 5.

Lesson: Information about mishaps should be shared, so that NASA can learn from its mistakes to possibly prevent future accidents. The investigation board's official report was classified, limiting knowledge of the incident and preventing the lessons learned from being widely understood.

This event should have been a wake-up call. It revealed issues that would appear again during the Apollo 1 fire (no engineering assessment of materials at 16 psi pure oxygen and no protection of Teflon wiring from physical damage to prevent arcs from shorts). Materials must be compatible with the environment (temperatures and pure oxygen exposure), but this commercial hardware had undergone no engineering analysis to show it was compatible with the test environment.

3.3. Apollo 8 (December 1968): launch pad electrical test

The Apollo Program Manager stated that if everything went well on the Apollo 7 mission, we would plan for Apollo 8 to be the first launch to go to the moon. NASA also thought the Russians would do a circumlunar flight before we did, so this launch and flight had particular pressure and concern. The AC system had electrical shorts on Apollo 7, but the cause had been identified and was thought to be fixed. However, the Program Manager directed that we run a test on the Apollo 8 spacecraft to carefully check out and test every AC load and component on the spacecraft, because he wanted to be absolutely certain nothing would go wrong with the AC system.

I was assigned to develop and run tests on the CSM AC electrical

system and record the voltage, current, power factor, and wattage of every AC-powered load. I worked for about a week in Houston planning what needed to be done. I already knew and had been working closely with flight operations at the North American Rockwell Space Division (NARSD) Downey, California design and test division, as well as the KSC and NARSD launch processing CSM power distribution personnel. The CSM 2TV-1 and Spacecraft-008 vehicles (full-up vehicles) had been tested in the large vacuum chamber at JSC, so I ran my planned tests of the AC system on 2TV-1.

Then I went to KSC, and Apollo 8 was on the pad. I spent a week at KSC writing the test procedure. I had known and previously worked with the NASA and contractor electrical power distribution engineering personnel, so they were very helpful in getting me the information I needed on the launch pad, such as spacecraft interfaces and required paperwork to develop the test on the AC system. I was very nervous at that time, even though I had run the test at JSC, because this was the first time I'd been responsible for a vehicle being tested at KSC on the pad for an upcoming moon mission, and making sure it worked with all the KSC ground support and connections for spacecraft checkout at the pad.

The test procedure was many pages. Even though I was very careful and worked closely with the NASA KSC and contractor engineers, I was sure they would carefully check my work as I was going through the required engineering and management approval signatures for the test. I assumed that the KSC engineers would carefully scrutinize the work of a young, visiting engineer from JSC. However, when this lengthy procedure started making its way through the signature chain, people signed off very quickly. They told me it seemed like I had checked with the right people and done the right things. I went through all the required signature chains for the test procedure, and no one had thoroughly checked my work. That made me even more nervous.

The test was scheduled for the third shift, meaning in the evening. I was out at the Saturn V launch pad in the White Room, sitting just outside the open crew hatch, with my test procedure and headset on to monitor the test, and the ground test crew was in the spacecraft. About halfway through the test, suddenly everything went black, and the lights in the White Room went off. I almost had a heart attack. I thought, “My goodness, what have I done?” Then I heard the control center say we had lost facility power, and I looked inside the hatch and saw all the spacecraft lights were still on, and everything was fine. The Florida Power and Light Company (which in those days we called “Florida Light and Flicker”) had experienced a power failure, so the lights in the White Room had gone off, which caused everything to go dark. The spacecraft itself was on backup emergency battery power, so it was fine. It turned out there was nothing wrong with my test, but at the time all I could think was, “You're responsible for scrubbing the Apollo 8 mission.” That experience stuck with me a long time, reference 6.

Lesson: On a critical task make sure your work is correct. Do not depend on someone else reviewing and checking your work. In this case, information was thoroughly documented and shared among organizations, but only one person was truly ensuring that the information was correct. Take the time to do your homework, and don't assume someone else will double check your work.

3.4. Apollo 11 (July 1969): service module entry

About five minutes after CM/SM separation, the crew reported seeing the SM fly by to the right and a little above them, straight ahead. It was first visible in window number 4, then later in window number 2, and spinning. The CM should never have been close enough to see the SM after separation. During lunar return, if the SM contacted the CM in the entry corridor the result would be catastrophic, so this triggered an anomaly investigation. Photographs obtained by aircraft showed the SM entering Earth's atmosphere and disintegrating in the vicinity of the CM entry corridor. Radar tracking confirmed what the photographs had shown. The radar tracking data for the previous Apollo 8 and 10 lunar return missions were similar to Apollo 11, with the SM entering in the same corridor as the CM.

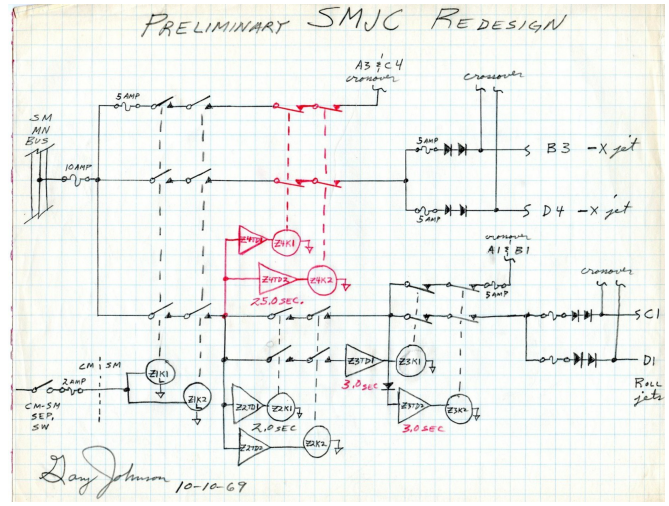


Fig. 2. Preliminary service module jettison controller redesign.

To prevent SM re-contact with the CM during entry, a System A and redundant System B SM Jettison Controller (SMJC) were located in the SM. As the Apollo Sequential Events Control System NASA Subsystem Manager, I was responsible for the design and development of the SMJC. Redundant signals from the crew operated a guarded CM/SM separation switch, which would initiate a sequence in each SMJC, firing all four –X SM RCS jets. Two seconds later it would fire the four RCS roll jets, and 3 s later would terminate the four RCS roll jets, while the –X RCS jets continued to fire. Electrical power for the SMJCs and the RCS came from the still-active fuel cells in the SM, and the RCS used the residual SM RCS propellant.

The investigation analysis showed that under certain conditions, such as propellant slosh after separation, re-contact could occur. We were lucky, as analysis showed that if the –X RCS jets were turned off after 25 s it would prevent any chance of re-contact. That change was then made to the SMJCs for the later Apollo missions, see Fig. 2.

This anomaly was not in any of the Apollo 11 mission reports, and I had forgotten about this close call prompting a design change to the SMJC. However, in December 2016 when looking back through my files, I found MSC-03466 “Apollo 11 Anomaly Report No. 3 Service Module Entry,” dated November 1970. Since the date is long after the Apollo 11 mission in July 1969, this is probably the reason the anomaly is not mentioned in any of the Apollo 11 mission reports, reference 7.

I had previously been concerned that the Orion spacecraft SM currently being designed and built by the European Space Agency for NASA's Orion Program did not have a requirement for an active controller after CM/SM separation. When I mentioned this, the Orion Program personnel said their analysis indicated that the present design met their requirements. The CM/SM separation force is provided by springs with the CM RCS firing after separation. No safety hazard report had been written to address the hazard of SM re-contact with CM during

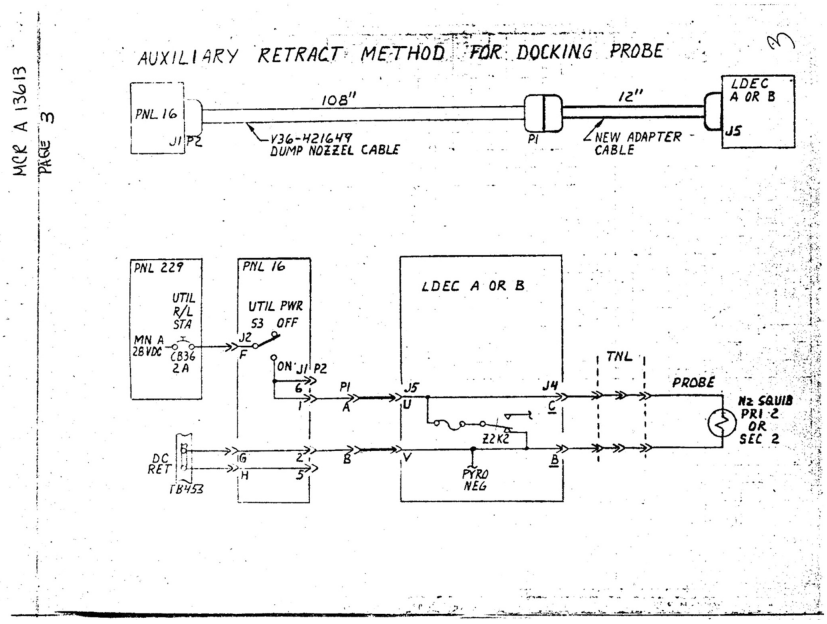


Fig. 3. Docking probe retract schematic.

entry. When I rediscovered the Apollo 11 anomaly report, I passed the information along to the Orion Program. So far it has generated an action to develop a safety hazard report to cover SM re-contact with CM during entry.

Lesson: Lessons learned should be freely available and shared. Mission reports should be updated to include anomaly and failure investigations that occur after the mission. Also, because the crew had reported seeing the SM during the Technical Crew Debriefing, this information was stamped "confidential" and was prevented from appearing in the Apollo 11 mission reports. This example shows that the quality of the documentation and sharing of information from past programs can have a direct impact on future programs, which need to be aware of all the potential hazards they may face.

3.5. Skylab 2 (May 1973): emergency docking procedure

After Skylab rendezvous and approach, the first objective was a "soft docking" at Skylab's forward port, engaging capture latches but not retracting the CM's docking probe to obtain hard dock. Then the crew released the capture latches, backed away from Skylab, flew around to a stuck solar panel with the side hatch open, and tried to free the solar panel. Unable to free the solar panel, the crew closed the side hatch and proceeded with the final docking to Skylab.

After numerous attempts, the crew was unable to achieve soft dock. Movement after the previous soft dock may have damaged the capture latches. There was one more procedure in the checklist labeled "Final Docking Attempt." Following the checklist, the crew members donned the pressure suits, depressed the cabin, opened the tunnel hatch, removed the probe cover, and cut the wires. They also connected the emergency retract cable to the Utility Power Outlet and the other end of the cable to the Lunar Docking Events Controller connector J5, see Fig. 3. After firing the probe retract pyro using utility outlet power, the commander was able to make direct contact, triggering the 12 structural latches and achieving hard dock, see Attachment for location on the chart.

The procedure for achieving hard dock when capture had failed was developed during the Apollo 14 mission (see "Expect the unexpected, and never stop learning"). After Apollo 14 a special cable was developed and stowed in the CM and the right-hand equipment bay panel was modified to allow quick access to the J5 connector. The procedure did not require removing the probe retract cover to cut wires, for which the crew had to be suited and the cabin depressed. Later, without coordinating with Engineering, the Flight Operations personnel changed the crew checklist to require the cutting of the probe wires, reference 8.

The culture at the time was that only Flight Operations and the crew were involved in developing the crew checklist procedure, and Engineering did not have the opportunity to review the crew checklist. The official Engineering interface with Flight Operations on crew procedures was via the Crew Procedure Change Request.

Lesson: Organizational culture should encourage interdepartmental communication. In this case, because organizations outside of Flight Operations could not review and approve crew procedures, the Skylab 2 crew used a much riskier procedure, which required donning suits and depressurizing the cabin. Engineering's procedure could have been performed using only the emergency probe retract cable without requiring access to the docking probe.

Sharing information across organizations can enable better, more informed decisions. A process change with regard to reviewing crew procedures occurred after the Space Shuttle *Challenger* accident when, over the objections of Flight Operations, the Space Shuttle Program approved the request from SR&QA to allow safety engineers to review the checklist to ensure the operational hazard controls were properly implemented.

3.6. Apollo-Soyuz test project (Spring 1975): service module inspection

The later Apollo missions had a scientific instrument bay in the SM for conducting experiments in lunar orbit. Some experiments had booms extending out into space, and limit switches indicated whether the boom had been retracted far enough away to fire the SPS engine

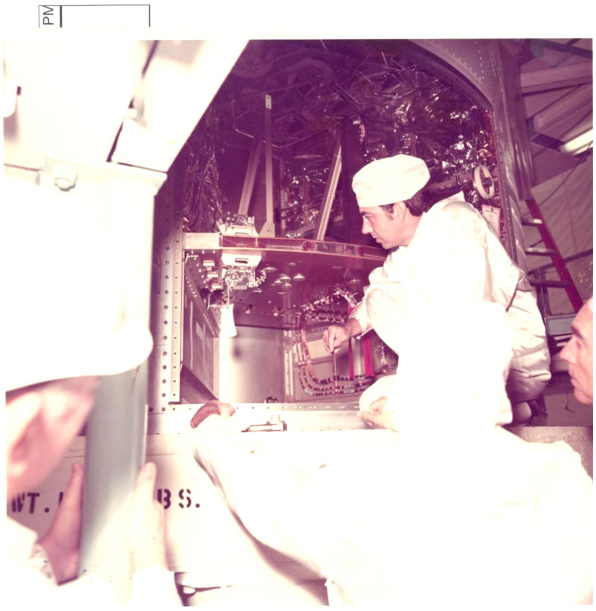


Fig. 4. Photo of Gary Johnson identification of faulty micro switch installation in ASTP service module.

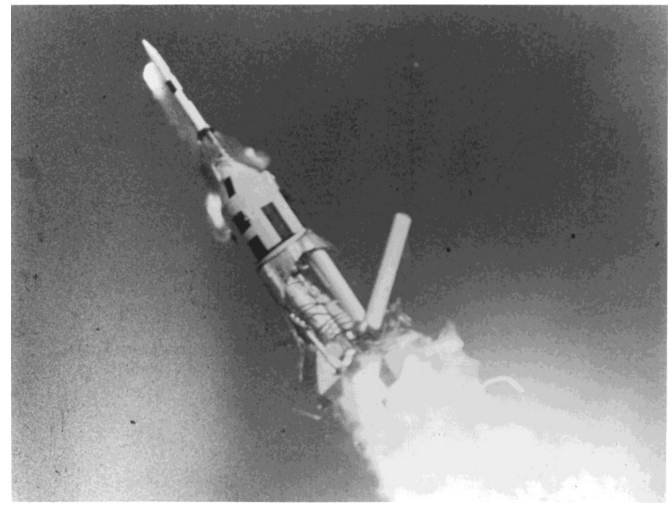


Fig. 5. Apollo BP-22 Little Joe II breakup.

without that boom coming back and colliding with the SM. On one of those missions the limit switches were not sensing properly, which was found to be an installation problem and was later corrected.

When I was working on the ASTP doing the wiring inspection for the SM scientific instrument bay at the North American Aviation, Inc. plant at Downey, California, I was looking carefully at those same limit switches because of the problem that occurred on the Apollo mission. I noticed that the little lever arm that's supposed to trip the switch was not making good contact with the boom piece, so I flagged that during the walk-around inspection, see Fig. 4. Then we checked to see how that issue had been missed. The quality team always inspected the vehicle to make sure it was built per the installation drawing, and it turned out that this was indeed per the installation drawing.

The explanation was the long time lag between the last Apollo flight, Apollo 17 in 1972, and ASTP in 1975. In response to the limit switch problem during the Apollo mission, an engineering order was attached to the drawing to correct the limit switch instillation, instead of changing the actual drawing, which was more expensive. Often, many engineering orders were attached to a drawing. When those installation drawings were taken from the files to be used in ASTP, the engineering orders were not included.

The quality assurance personnel could only verify that the installation matched the drawing, not that it matched the intended design. However, the designer could see if the drawing was wrong. That was the beauty of both NASA and contractor design personnel performing vehicle inspections. After the Apollo 1 fire, NASA required a "Management Walk-Around Inspection" of all spacecraft prior to shipment to KSC, and this continued in the Space Shuttle Program, reference 9.

Lesson: Design drawings should be updated following a design change. Engineering orders (attachments which show a change to a design) should not be overused. When an older design is being used, ensure that all of the subsequent modifications and changes have been documented and included.

Also, involving personnel that participated in previous missions or investigating past lessons learned can help avoid the repetition of past mission anomalies. For critical systems, the designers should be able to inspect the installation, to make sure it accurately follows the intended design.

4. Hazard analysis is critical

Hazard analysis was not routinely performed during the Apollo Program and was not fully accepted in the broader NASA community until the *Challenger* accident. Engineering safety hazard analysis should not only be performed for flight systems, but also for critical spacecraft ground tests. Also, the hazard analysis should be repeated when there is a subsequent design change. Pessimistic thinking can help reveal risk areas and motivate preparations for emergencies.

4.1. Apollo mission A-003 (1965): Little Joe II booster fin failure

Apollo Mission A-003 used a boilerplate spacecraft (BP-22) launched from Launch Complex 36 at White Sands Missile Range on May 19, 1965. The mission was a high-altitude test of the abort system, and the Little Joe II had six Algol solid rocket motors, making this the largest solid rocket motor launch at that time. Concern about the reliability of the abort signal from the launch vehicle led to using a loss of signal/open circuit rather than an electrical signal being sent. Three abort signals were to be sent in two-of-three voting in the spacecraft sequencer to initiate the abort. A radio frequency command was sent to the launch vehicle for the abort, which powered a relay opening the normally closed contacts, causing loss of signal to the spacecraft.

Shortly after liftoff one of the Little Joe fins failed hard-over, and the faster it went forward, the faster it spun around till centrifugal force caused the launch vehicle to structurally fail approximately 25 s after liftoff. The launch vehicle breakup, Fig. 5 caused the open circuit/loss of signal, and the spacecraft sequencer initiated an automatic abort of the BP-22 spacecraft, see Fig. 6. The spacecraft separated from the launch vehicle at a high roll rate. The drogue chute had steel cables as risers, which was beneficial, since the spacecraft upper deck was

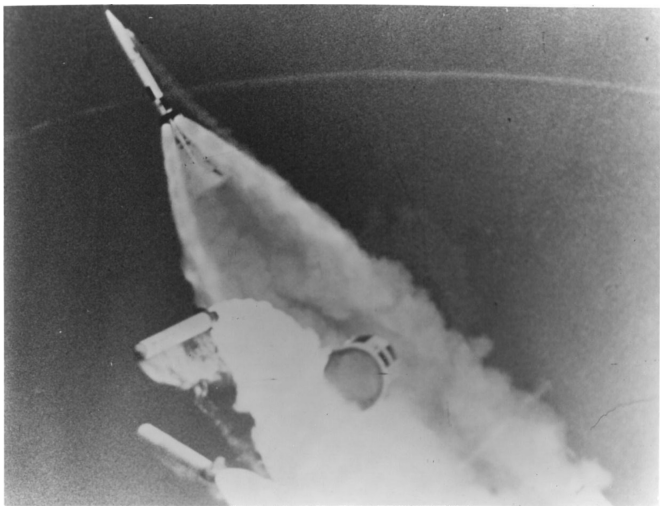


Fig. 6. Apollo BP-22 Abort.

damaged from the risers lashing around in the process of damping out the roll rate. The lower part of the main risers had a fine steel overwrap. Concern about spacecraft dynamics during parachute deployment causing damage to the parachute risers from upper deck sharp edges resulted in the changes to the parachute risers.

The abort command from the launch vehicle implemented as an open circuit allowed this signal to also be an indication of launch vehicle structural integrity. This concept was carried forward to the Saturn 1B and the Saturn V launchers. The three Emergency Detection System abort circuits to the CM were routed 120° apart along the inside of the outer structure, with open circuit being the abort command. Two out of three of the circuits being open would initiate an automatic abort. The change to steel cables for the drogue chute risers is probably the reason the spacecraft was safely recovered, reference 10.

Lesson: Engineering safety analysis should indicate what the abort signal to spacecraft should be if the launch vehicle structurally fails. Parachute risers should be designed for the worst case environment.

4.2. Apollo-Saturn mission 201 (February 1966): loss of reaction control system

After CSM separation, entry heating led to an electrical short, causing the RCS commands to be transferred from the CM to the SM, resulting in the loss of RCS control. After the loss of the RCS the CM went into a stable roll and did a ballistic entry instead of the planned lifting re-entry.

The cause of all this was a non-functional circuit (Criticality 3) that was routed through the CSM umbilical that was not deadfaced prior to separation, and it was powered from the Sequential Events Control Subsystem System B circuit breaker, causing loss of the redundant Earth Landing System (Criticality 1). The non-functional circuit had been dropped from the drawings, but the wiring was left in the spacecraft and was not disconnected from power. Since it was not on the drawing for the powered wiring going through the CSM umbilical, it was not deadfaced prior to guillotine of the umbilical. The redundant Earth Landing System allowed for recovery of the CM.

Lesson: Hazard analysis should be performed on critical systems. All non-functional circuits, if left in the spacecraft, should be disconnected from both power and return and should be identified on spacecraft drawings. Careful review should ensure all powered wiring through an umbilical to be guillotined is deadfaced prior to cutting with the guillotine.

Also, Criticality 3 functions (which are not essential for crew or vehicle survival) should not be allowed to affect Criticality 1 functions (which are essential for the operation of the vehicle and/or the survival of the crew).

4.3. Apollo environmental control system (April 1966): fire during test

See the full summary of this event in “Documenting and sharing information: communication is key.” The hardware used in the test had undergone no engineering analysis to show it was compatible with the test environment. Hazard analysis would probably have revealed that the commercial heater tape insulation could not withstand the temperature to which it would be exposed.

Lesson: Hazard analysis should be performed on critical test configurations. The compatibility of materials with the environment (temperatures and pure oxygen exposure) must be assessed. This error should have led to greater awareness of this issue, but similar problems occurred the following year during the Apollo 1 fire.

4.4. Apollo 1 (January 1967): fire and loss of crew

At 6:31 pm on Friday, January 27, 1967 at KSC during a simulated countdown for the AS-204 mission, a fire broke out in the CM and quickly caused the loss of the crew, Virgil Grissom, Ed White, and Roger Chaffee. I was in the JSC MCC on the Electrical Power System Console in the Staff Support Room alongside my flight controller counterpart monitoring the test at KSC. The test was running long due to numerous problems. Because the MCC was just monitoring the test being run from KSC, after 5:00 pm the flight control personnel in the Staff Support Room left for the day. I and the North American Space and Information Systems flight controller on the Environmental Control Systems Console were the only ones left.

We heard the scream of fire on the headset and the KSC personnel trying to get to the CM. Ground personnel grabbed the available gas masks, but passed out after entering the smoke-filled White Room around the CM, because the masks were for filtering hypergolic propellant fumes and were not closed, oxygen-providing masks. I was hopeful, thinking the crew members were in their spacesuits and would be okay, but I heard the KSC test conductor on the headset tell Dr. Christopher Kraft to go to a private phone. My heart sank, as I knew it meant the crew didn't survive. Dr. Kraft was monitoring the test at the Flight Director Console in the Mission Operations Control Room. He then announced on the Flight Director loop to lock the doors, that no one was to leave, and that we were allowed one call to our spouse to say we would not be home, but not to say anything else. He later came back to the Staff Support Room and told us they would be playing back the data and to concentrate on reviewing it.

I had noticed at about the time of the report of fire that we had a short on Main DC Bus A and B. That indicated that the short must have a load connected to both main DC buses. The following day I went over all the schematics to identify the wires dioded to both main buses. The next week I was asked to go to KSC for the investigation. I was assigned, along with a photographer, to go through the wiring in the CM to identify the ignition source, see Attachment for location on the chart.

A significant part of the left-hand lower equipment bay was gone (metal, plumbing, wiring, etc.). This was where the fire started and was

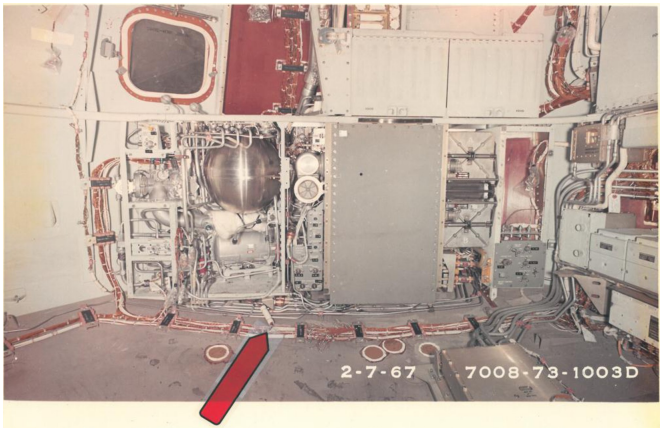


Fig. 7. Apollo Block 1 command module left hand lower equipment bay.

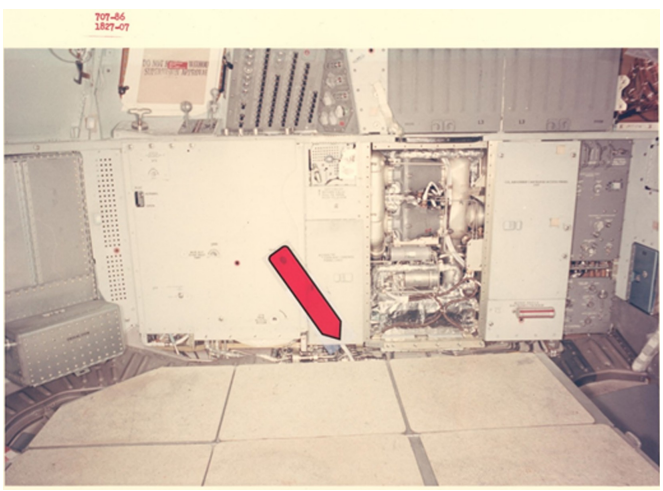


Fig. 8. Apollo Block 2 command module left hand lower equipment bay.

the hottest, because the most oxygen was present. The most probable initiator was an electrical arc occurring near the floor in the lower forward section of the bay. Here, instrumentation power from a Teflon-insulated wire powered by Main Bus A and B was routed over metal plumbing and under a coolant control access panel, just below the left crew couch.

The wire harness, a twisted pair of wires, and power and power return had an extra Teflon overwrap for protection from the panel door and the metal plumbing. However, the last closeout photo of that area before the fire showed that the extra Teflon wrap had slipped down, and was not keeping the wire away from the plumbing. The wire harness had no extra protection from the panel door opening/closing or from ground test personnel or crew at the time of the test. The wire looked like it was touching the bottom of the panel door. Spacecraft movement was noted during the test, as the commander was reconnecting the communications cable, and the damage could have happened at this time. A simulation showed that he would have had to put his foot down, off the crew couch, to reconnect the cable, and would have placed his foot near this wire, see Fig. 7. Fig. 8 shows protective changes, hard portable flooring for ground crew and metal covers over wire harness, shown at far left and right. Middle is open in this photo.

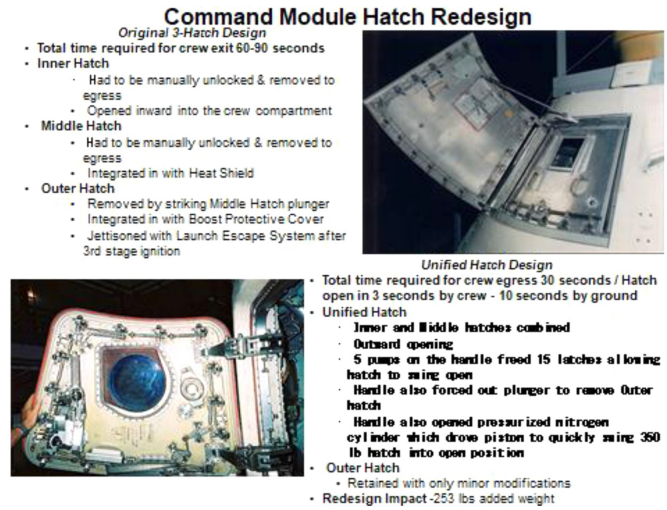


Fig. 9. Command module hatch redesign.

The crew cabin was pressurized to 16.7 psi with pure oxygen. The side hatch consisted of a two-piece, inward opening (pressure sealing) hatch. The fire increased the pressure, which made the hatch more difficult to open, so the crew was unable to open the hatch to escape. It took 90 s to remove the hatch from the outside and get them out.

No engineering safety hazard analysis was performed on the ground test configuration. If performed, it probably would have identified that the materials inside the cabin were not certified for the environment and that no protective gear or training was available for fighting a fire on the spacecraft level at the launch pad, reference 11.

Lesson: Hazard analysis should be performed on critical test configurations. All tests at 100% oxygen should be defined as hazardous. Emergency training should be required for all test support personnel, and test areas should be equipped with appropriate emergency, fire-fighting, or rescue equipment. The selection and placement of materials inside the spacecraft must be compatible with the spacecraft environment. The crew ingress/egress hatch should be a single, outward-opening hatch requiring only 5 s to open, see Fig. 9.

4.5. Apollo 13 (April 1970): oxygen tank explosion

See the full incident summary in the section “Expect the unexpected and never stop learning.” The thermostatic switches were rated for 30 V DC, but several years earlier the heater ground power supply voltage was raised to 65 V to reduce the pressurization time. As the temperature increased, the thermostatic switch opened and the higher voltage caused the contacts to weld closed. No engineering safety hazard analysis of the voltage change to the thermostatic switch was performed, reference 12. Note in the Attachment Apollo 13 location on the chart.

Lesson: Hazard analysis should be repeated following a design change.

4.6. Skylab 3 (July 1973): propellant leak and rescue mission

See the full incident summary in the section “Redundancy can help or hinder.”

Lesson: Hazard analysis should be performed on redundant systems, to ensure they provide the expected redundancy. Had the analysis been performed, it would have revealed that a system that appeared to be

single-fault tolerant was actually zero-fault tolerant in effect, because the backup system would not work.

The programs' early planning to develop a Skylab rescue capability is what allowed a rescue mission to be developed so quickly. This is a good example of reaping the benefits of planning ahead for something to go wrong.

5. Politics is the enemy of good design

The political interests of NASA centers can motivate decisions that are in the best interests of the center, but not in the best interests of the overall design or of safety. A spacecraft contractor that wants to use a particular hardware subcontractor may work to show that its hardware is the best, even if the customer has a cost concern about that subcontract, and even when it compromises the overall systems design and safety. The following examples from the early development and design of the space shuttle illustrate this problem.

5.1. Space shuttle main engine electrical design

Why was the orbiter designed for 117 V AC power instead of the aerospace standard of 115 V?

The Power Distribution and Control Branch in the Control Systems Development Division was responsible for the AC power and distribution for the orbiter being developed by Rockwell Space Division. As a redundancy improvement over Apollo, the AC inverter would be single phase 115 V, using three inverters to power a three-phase AC bus. The vendor's design was similar to the design used in commercial aviation. After the design was well underway, we were informed that the MSFC Space Shuttle Main Engine (SSME) controllers under contract were designed to receive 115 V, three-phase AC power. This was because the SSME contracts were awarded early when the proposed orbiter design called for AC generators in the aft to provide electrical power. Later when the orbiter contract was awarded, the design removed the AC generators and electrical DC power was provided by fuel cells.

We informed MSFC that the power in the aft compartment would be DC power. MSFC did not want to change the SSME controller design, due to cost and schedule, and insisted on receiving AC power, rather than the DC power located in the aft compartment. I don't know the actual figures, but I suspect the SSME controller cost and schedule change was greater than the cost and schedule change to the orbiter AC inverter. It was probably the correct decision for cost and schedule, but looking at it from an integrated power distribution design standpoint, the DC power would have been a more reliable power source and lower safety risk.

Rockwell had to change the inverter design to provide the non-aerospace standard 117 V, which was required because of the two-volt line drop from the forward compartment AC bus to the SSME controllers. All AC powered equipment in the orbiter had to change to accept 117 V instead of the standard 115 V. This was a less reliable source of power for the SSME controller due to the three small-gauge wires, long wire run with many connectors (forward avionics bay to aft compartment SSME controller), and need for each phase to be protected by a three-ampere circuit breaker. This compares to a design of two wires (power and return) from the aft avionics bay to the SSME controller. Electrical DC power to the aft avionics bay from the forward distribution assembly was via two very large-gauge wires, each protected by 200 A fuse. During ascent on STS-93 a short on AC bus 1 phase A caused the loss of SSME 1 Controller A and SSME 3 Controller B.

Lesson: NASA centers tend to compete with each other, and may not want to take each other's advice, especially if it involves redoing work. These factors may influence spacecraft design. Also, this example shows cost and schedule considerations taking precedence over design and safety.

To avoid this situation, NASA Headquarters would have to review both centers' proposals and insist on the design with less safety risk. This would be challenging to implement.

5.2. Space shuttle avionics computers

Why were the computers air cooled, while the avionics were on cold plates?

In the orbiter contract Rockwell had stated that IBM would provide the orbiter avionics computers. IBM was very expensive compared to the other bidders for the orbiter computers, but told NASA these would be the flight-proven IBM computers flying on the B-52s. The B-52 computers were air cooled, like all aircraft avionics. The orbiter avionics were on cold plates, which would not be affected if the orbiter crew cabin became depressurized, but because the IBM computers were the less efficient air-cooled design, they would fail if the cabin were depressurized. Also, the air inlet filters had to be periodically cleaned. IBM's orbiter computer design was nothing like its aircraft computers, so the argument that the design was flight proven was no longer valid.

After the shuttle had been flying for several years, NASA considered converting the computers to a cold plate design, which would involve removing the air ducts and adding cold plates, but the idea was rejected as too expensive.

Lesson: Contractors' business interests may influence design decisions, with detrimental effects. Also, either the hazard analysis in this situation was insufficient or the additional risks posed by the IBM design were not sufficiently communicated, resulting in a poor design decision.

6. Conclusions

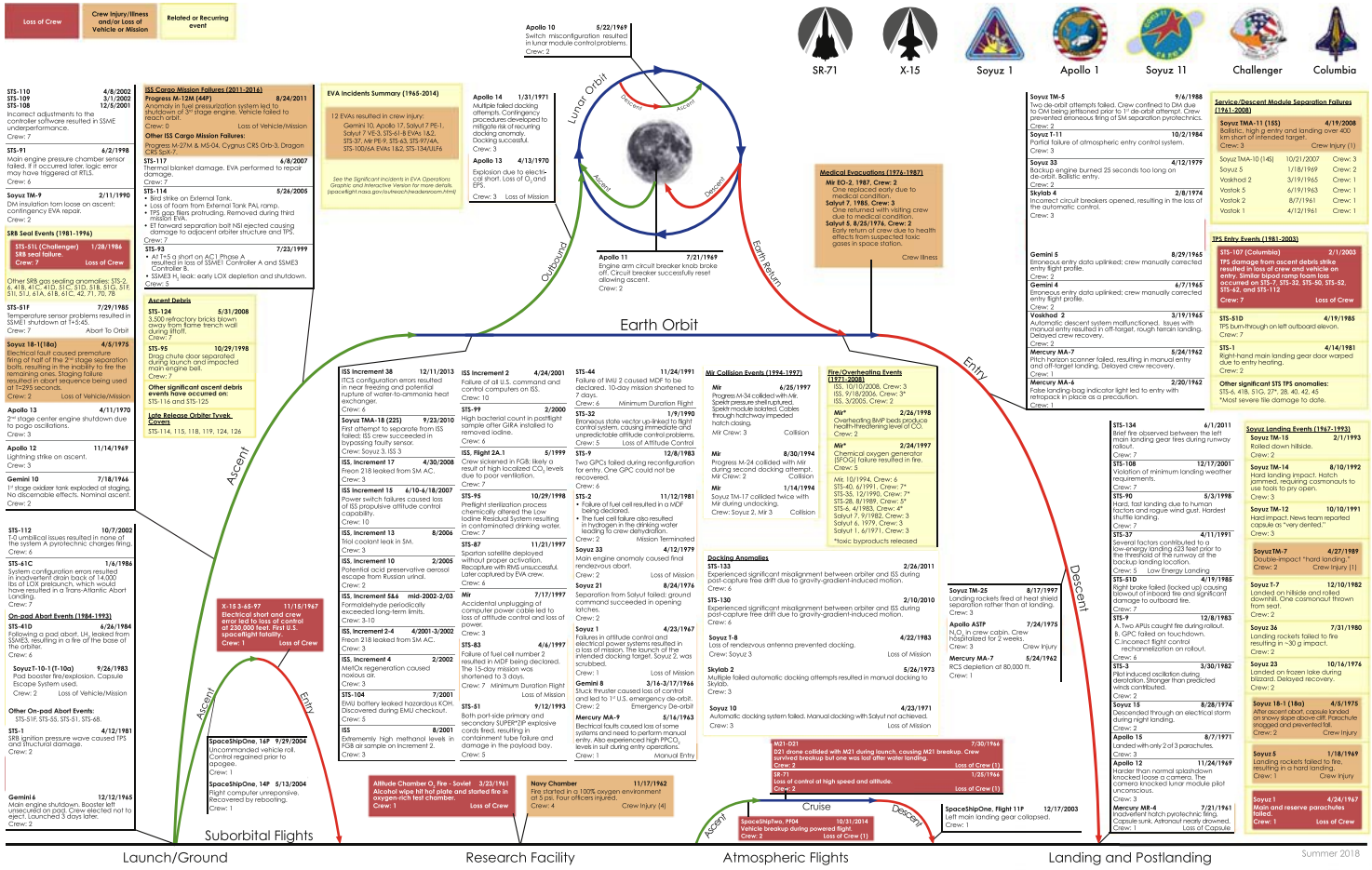
These lessons apply to future human spacecraft design, even though the technology has changed. The Apollo redundancy philosophy of no crew safety single-point-failures and amount of redundancy determined by the criticality, flight experience, and maturity of technology is just as valid now as it was then. For fluid systems, redundant seals must be compatible with the fluid. Communications is vital during the design process not only when failures occur but with the manufacturing, test, and operations organizations working as a subsystem team. Design must take into account the hazards in testing, flight, and environment during ground processing, as well as flight. 'Significant Incidents and Close Calls in Human Spaceflight' graphic to raise awareness of past incidents and lessons learned, such as the ones in this article, to encourage future designers to incorporate the lessons of the past into future spacecraft designs, operations, and processes.

Attachment

Significant incidents and close calls in human spaceflight.

Significant Incidents and Close Calls in Human Spaceflight

A Product of the JSC SMA Flight Safety Office



The Significant Incidents and Close Calls in Human Spaceflight graphic is primarily focused on human spaceflight incidents occurring with a crew aboard a space vehicle. It includes suborbital, orbital, and lunar missions. Selected non-spaceflight and uncrewed events are included if they have strong relevance to human spaceflight. For instance, the loss of the uncrewed Progress 44P is included because it has launch vehicle commonalities with the crewed Soyuz missions. The altitude chamber oxygen fire in Russia preceded the U.S. Navy oxygen chamber fire and the Apollo 1 fire, which occurred under similar circumstances. The SR-71 accident is the highest and fastest vehicle breakup on record that was survivable, and it represents the demonstrated limit of crew survival with current technology. The SpaceShipTwo accident represents the loss of a suborbital space vehicle during flight testing.

This document is a work in progress. It is continually under review and frequently updated. Please direct comments and questions to the JSC SMA Flight Safety Office.

Abbreviations and Acronyms

AC	Air Conditioner
APU	Auxiliary Power Unit
BMP	Microimpurities Removal System (Russian)
CDRA	Carbon Dioxide Removal System
CMG	Control Management Gyroscope
CO	Carbon Monoxide
CO_2	Carbon Dioxide
DM	Descent Module
EMU	Extravehicular Mobility Unit
EPS	Electrical Power System
EVA	Extravehicular Activity
FGB	Functional Cargo Block (Russian)
FSO	Flight Safety Office
GIRA	Galley Iodine Removal Assembly
GPC	General Purpose Computer
GPS	Global Positioning System
H_2	Hydrogen
IMU	Inertial Measurement Unit
ISS	International Space Station
ITCS	Internal Thermal Control System
KOH	Potassium Hydroxide
LH_2	Liquid Hydrogen
LOC	Loss of Crew
LOV	Loss of Vehicle
LOX	Liquid Oxygen
MDF	Minimum Duration Flight
MetOx	Metal Oxide
MMOD	Micro-Meteoroid Orbital Debris
N_2O_4	Nitrogen Tetroxide
NSI	NASA Standard Initiator
O_2	Oxygen
OM	Orbital Module
OSMA	Office of Safety & Mission Assurance
PAL	Protuberance Air Load
PASS	Primary Avionics Software System
$PPCO_2$	Partial Pressure of Carbon Dioxide
RCS	Reaction Control System/Subsystem
RMS	Remote Manipulator System
RTLS	Return to Launch Site
SFOG	Solid Fuel Oxygen Generator
SMA	Safety & Mission Assurance
SM	Service Module
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
SSP	Space Shuttle Program
STS	Space Transportation System
TPS	Thermal Protection System
U.S.	United States

Visit the NASA OFFICE OF SAFETY & MISSION ASSURANCE (https://sma.nasa.gov/sma-disciplines/mishap-investigation) for the latest PDF and interactive versions of the Significant Incidents and Close Calls in Human Spaceflight chart.

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