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Battery Failure on the Electric Propulsion Space Experiment (ESEX)

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14. ABSTRACT

The anomaly experienced on the USAF Electric Propulsion Space Experiment (ESEX) mission related to the silver-zinc battery is described. The on-orbit observations of the battery behavior in relationship to the thruster firings are summarized, as well as a description of the catastrophic failure which ended the ESEX mission. Although a final cause of the battery failure was not definitized, the data collected on-orbit, from ground tests, and from previous programs suggest a combination of effects that ultimately led to electrolyte leakage, and a subsequent short circuit between the battery cells and the case. The contributing effects likely include degradation of the mechanical interconnections between the cells, inadequate charge current, and excessive discharge current requirements. Lessons learned from the ESEX experience focus on the need for more realistic simulations of the on-orbit operations during ground test including the use of actual flight hardware and the associated timelines as much as possible.

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NOTICE

FOREWORD

This special technical report, entitled "Battery Failure on the Electric Propulsion Space Experiment (ESEX)," presents the results of an in-house study performed under JON 6340RH65 by AFRL/PRSS, Edwards AFB, CA. The Principal Investigator/Project Manager for the Air Force Research Laboratory was Daron R. Bromaghim.

This report has been reviewed and is approved for release and distribution in accordance with the distribution statement on the cover and on the SF Form 298.

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GLOSSARY

A	ampere
AFB	Air Force Base
ARGOS	Advanced Research and Global Observation Satellite
Ag-Zn	silver-zinc
A-hr	amp-hour
EMI	electromagnetic interference
ESEX	Electric Propulsion Space Experiment
Ga-As	gallium-arsenide
IMS	integrated mission simulation
kW	kilowatt
LLC	Limited Liability Company
mA	milliampere
mΩ	milliohm
NiH	nickel hydride
nmi	nautical mile
OSC	Orbital Sciences Corporation
PAC	Primex Aerospace Company
PCU	power conditioning unit
psi	pounds (force) per square inch
RDT&E	research, development, test and evaluation
RSC	RDT&E Support Complex

GLOSSARY (cont.)

TQCM	thermo-electrically-cooled quartz crystal microbalance
USAF	United States Air Force
Vdc	volts direct current / direct current voltage
Ω	ohms

Battery Failure on the Electric Propulsion Space Experiment (ESEX)

1.0 HARDWARE DESCRIPTION

1.1 ESEX Description

The Electric Propulsion Space Experiment (ESEX) was a United States Air Force (USAF) Research Laboratory space demonstration of a 30 kW ammonia arcjet designed, fabricated, and tested by TRW, Primex Aerospace Company (PAC) and Orbital Sciences Corporation (OSC).[1] The experiment objectives were to demonstrate the feasibility and compatibility of a high power arcjet system, as well as measure and record flight data for comparison to ground results. The flight diagnostics included four thermo-electrically-cooled quartz crystal microbalance (TQCM) sensors, four radiometers, a section of eight gallium-arsenide (Ga-As) solar array cells, electromagnetic interference (EMI) antennas, a video camera, and an accelerometer.[2] Data from these sensors addressed all of the potential integration issues including contamination, EMI, and thruster performance on an integrated spacecraft, and successfully demonstrated the feasibility of this system for future applications.[3-8] An exploded view of the ESEX flight unit is shown in Figure 1.

ESEX was one of nine experiments on the USAF's Advanced Research and Global Observation Satellite (ARGOS).[9,10] ARGOS was launched on 23 Feb 99 from Vandenberg AFB, CA, on a Delta II into its nominal orbit of approximately 457 nmi (846 km) at 98.7° inclination. The

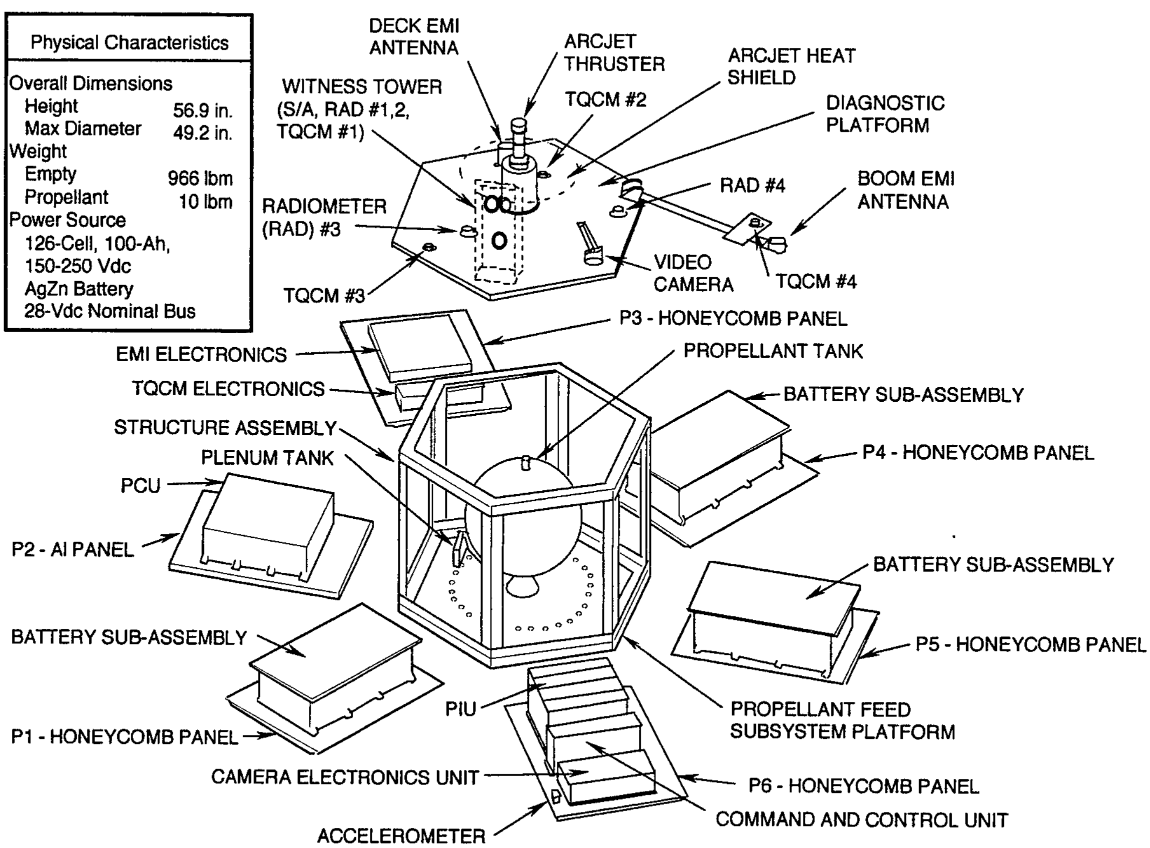

Figure 1. ESEX Exploded View Showing All of the Flight Diagnostics, the Arcjet, and the Supporting Electronic Boxes (Including the Battery Sub-Assemblies)

satellite was controlled from the Research, Development, Test and Evaluation (RDT&E) Support Complex (RSC) at the USAF Space and Missile Test and Evaluation Directorate at Kirtland AFB, NM.

The planned ESEX mission profile consisted of 10-15 firings of varying length, from 5-15 minutes.[11] In order to accomplish the 30 kW firings (since no current space-based platform can generate a steady-state power level of that magnitude), rechargeable silver-zinc (Ag-Zn) batteries were used.[12] These 80 A-hr batteries were charged from an on-board charging circuit that was designed to supply 0.67 A of charging current to the battery. For a typical operational cycle, the charging circuit would have been turned on approximately 24 hours after an arcjet firing, and allowed to shut off autonomously prior to initiating the next firing via a software-controlled battery voltage trip point. This planned operational flow was not attained as a result of the anomalous battery behavior.

1.2 Battery Design Description

The ESEX Ag-Zn battery was designed and fabricated by Eagle-Picher Technologies, LLC, and is made up of 126 cells connected in series, divided evenly between three sub-assemblies. As shown in Figure 2, each of the battery sub-assemblies has 42 cells interconnected with a multi-

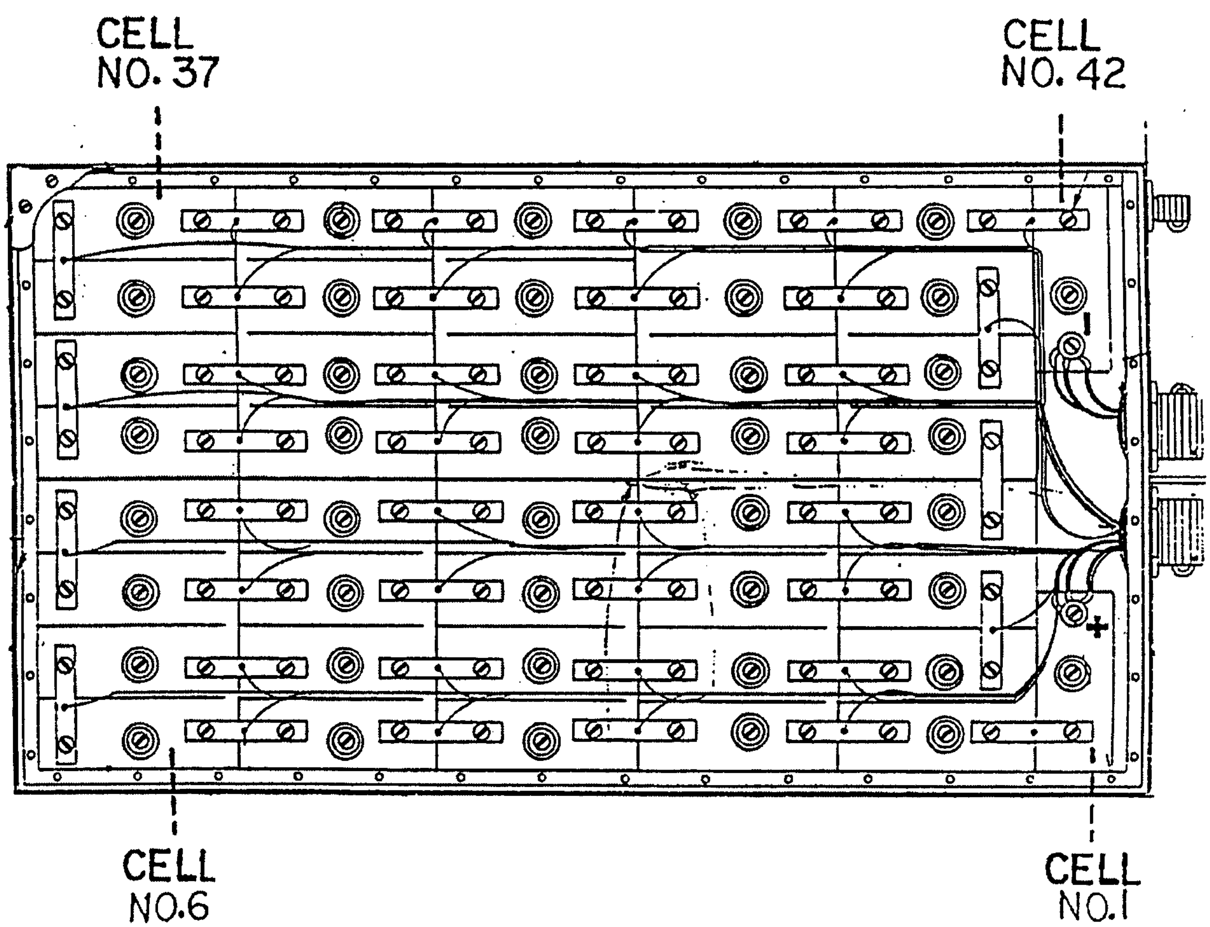

Figure 2. ESEX Battery Sub-Assembly Drawing Showing the Layout for the 42 Cells

layer copper strap (128 straps total, each strap had a resistance of 0.047 mΩ), bolted to the cell terminals. The completed sub-assembly then had a layer of potting encapsulate (FR-88 epoxy filled with microscopic glass beads) over the cells, covering the cell interconnects, leaving only the cell vents open to the interior of the battery sub-assembly chassis. Each sub-assembly also has a relief valve (cracking pressure of 3-8 psi) to vent over-pressure through its mounting panel to the outside of the flight unit.

The Ag-Zn cells and the packaging design were similar to other Eagle-Picher products; however, the application of this type of battery to the requirements ESEX imposed was very unique. As a result, one of the seven sub-assemblies was subjected to a limited qualification test series including vibration, thermal cycling, and a destructive evaluation. This testing was only performed on one of the seven in order to minimize cost. Additionally, a fully integrated functional test (performed as a part of the integrated mission simulation [IMS] test) with the engineering model arcjet power conditioning unit (PCU), and power cable was conducted with three of the sub-assemblies to further validate the design approach. Unfortunately, a test was never performed which charged the battery to the software shut-off point with the flight charger or at the flight charge rate. Segmented tests were performed that demonstrated the capability of the charger to operate at different battery voltages (i.e., 200 Vdc and 243 Vdc), but the primary charging was typically accomplished with an external power supply. This was a restriction resulting from the thermal constraints on operating the charger in ambient conditions. No anomalous results were ever observed during the battery development or during any of the tests to indicate a design problem. These cells are used in many other applications, including launch vehicle power systems, but typically for a primary (single) firing and at much lower discharge rates and higher charge rates than those used for the ESEX mission.

The ESEX mission requirement was to launch at an intentionally low state of charge, and then charge under less than optimal conditions on-orbit, since the orbit average power available from the spacecraft was limited. The effects of this approach were not clearly understood during the design phase, and ground tests were not adequate to uncover the ultimate shortfall. Additionally, the ESEX mission requirements to fire numerous times at unusually high currents were not a part of the battery design heritage, and presented several other design issues. The matting designed to absorb expelled electrolyte had never been proven over numerous high current firings. All of the ground firings were done with the batteries in an upright position in atmosphere, which would tend to reduce venting since the electrolyte would be positioned away from the vent ports. The unplated copper straps used on the cell interconnects are also prone to degradation of electrical contact with time under adverse conditions. This degradation would not be a problem for a mission profile that was controlled and short duration, but could become problematic under extended adverse conditions such as ESEX.

2.0 NARRATIVE DESCRIPTION OF THE ANOMALY

2.1 Battery Charger Instabilities

The first signs of anomalous behavior in the battery were observed during the first charging cycle, which was initiated approximately 60 hours after launch.[13] As indicated above, the charger was designed to provide an output current of 0.67 A to the battery. Within nine hours of

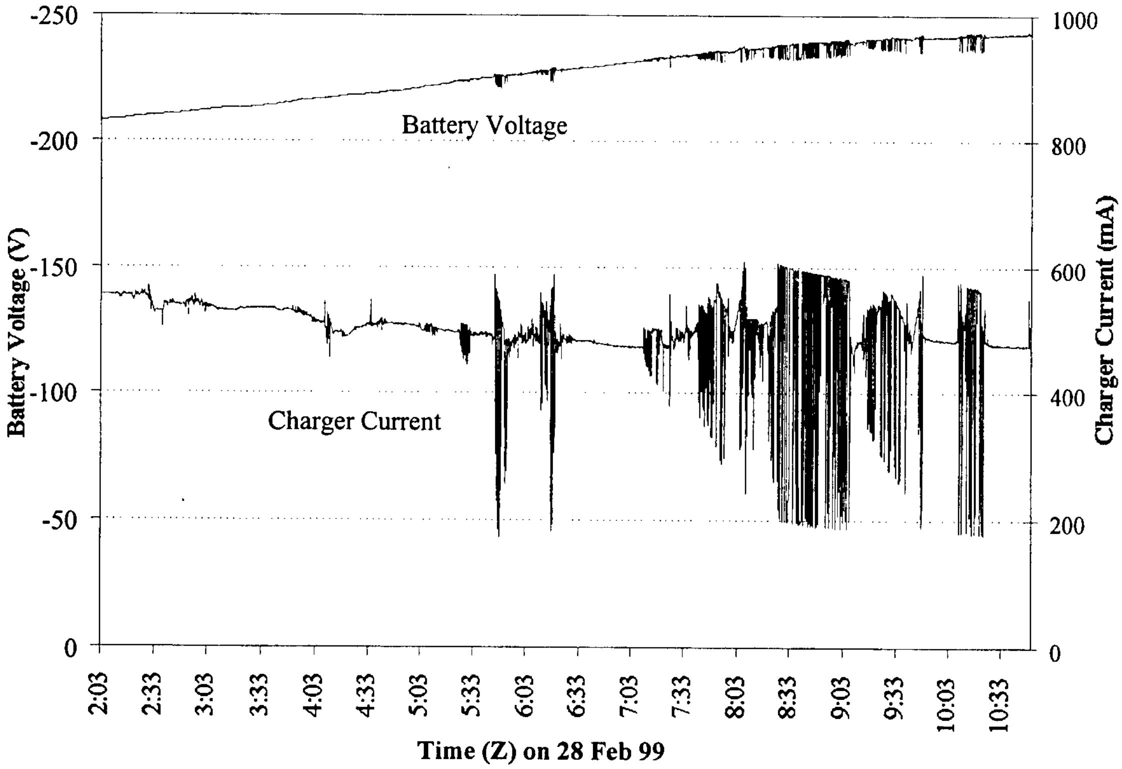

Figure 3. Typical Battery Charging Circuit Instability

turn-on, however, the charger output was less than 0.6 A, and decreasing. Although the charging circuit was operating at this lower output current, the charging proceeded nominally and was continued while the data was being monitored. As charging progressed over the next 48 hours, the output current remained stable around 0.57 A until the battery voltage approached -225 Vdc. At this point, as shown in Figure 3, the output current from the charging circuit began cycling on and off, resulting in oscillations of the open circuit battery voltage, charger output current, and even on the ARGOS 28 Vdc main bus current and voltage (not shown here). Initially, this was thought to be a result of higher-than-expected internal battery impedance as the charging progressed and the battery cooled – exacerbated by a low rate of charge. As the battery impedance is increased, the current loop control circuit in the battery charger becomes unstable, and results in the charger cycling on and off. This impedance was determined by taking the ratio of the change in battery voltage to the change in battery charge current. Near the first charging cycle, this impedance was calculated to be approximately 13 Ω – lending further credence to this proposed source of charging instability.

In an attempt to offset this problem and lower the effective charging circuit impedance, high capacitance filters were switched into the circuit via high voltage relays connecting the battery with the PCU.[12] This increased capacitance improved the charger performance, but did not eliminate the fluctuations. Since the output current instability was not detrimental to the ESEX battery or the spacecraft bus, charging was performed through this region. The total charging time was then increased to compensate for the charging inefficiencies resulting from the cycling output from the charger.

During the subsequent charging cycles, further degradation was observed as the battery impedance appeared to increase to a point where the charger would shut off prior to attaining a full state of charge. When attempting to restart charging at a battery voltage above -225 Vdc, for instance, the battery voltage would spike up to the software or hardware limit, and immediately shut off. The impedance required to cause this behavior was calculated in the same way to be approximately 58 Ω. When the battery voltage was allowed to discharge to below -220 Vdc, however, the charger successfully turned on and continued to charge to near the -247 Vdc turnoff point. Subsequent analyses indicate these instabilities might have actually been symptomatic of the ultimate problem, and may not have improved with either a more robust charger circuit design, or a higher output current.

2.2 Shortened Firing Durations

For the first three firings, no anomalous data were observed and the battery discharge voltage appeared normal as the power was ramped up and the firing progressed. The behavior replicated the experience observed during ground tests[12,14] where the voltage decreased approximately -65 Vdc following arcjet ignition, followed by a slight increase in output voltage. As the ground test firings progressed, and the battery temperature increased, this rise in voltage continued until the firing was terminated – at which point the battery voltage increased to approximately -200 Vdc.

Beginning on firing #4, however, a decrease in the battery output voltage was observed that limited the total firing duration. As shown in Figure 4, this drop in output voltage resulted in

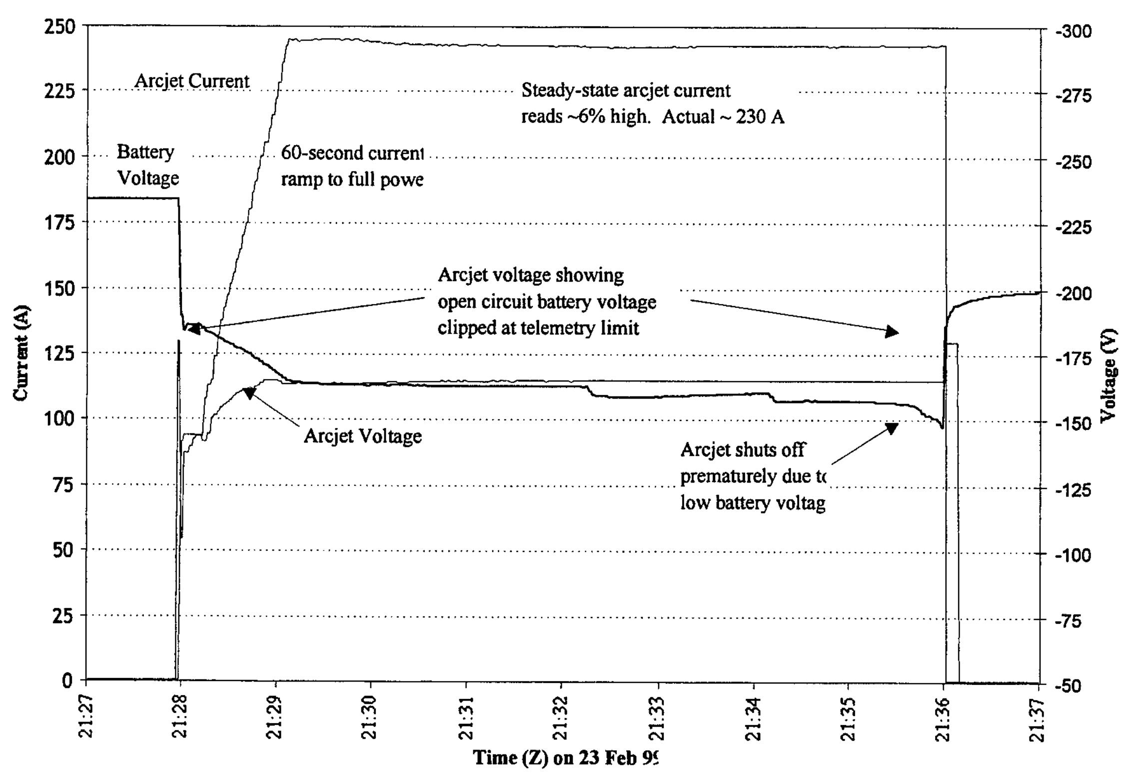

Figure 4. Example of a Premature Arcjet Firing Shut-off as a Result of the Low Battery Output Voltage. Data is Taken from Firing #4.

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unstable PCU and arcjet operation, and eventually extinguished the thruster. The duration of each firing from #4 through #8 steadily decreased from 482 seconds to 42 seconds, as the battery performance deteriorated. After firing #7, an attempt to recondition the battery was made by performing a deep discharge through the existing bleed resistors, and restarting the charge. The initial plan was to wait until the battery was at a full state of charge (indicated by the charger circuit shutting off at the upper voltage limit) before attempting the next firing. After more than 19 days, however, the charger was commanded off and a firing was attempted. As indicated above, the duration of firing #8 showed the reconditioning did not have the desired effect.

2.3 Short Circuit and Catastrophic Failure

Following firing #8, the battery voltage fluctuated erratically between -175 and -200 Vdc with periodic drops as low as -30 Vdc – where it eventually stabilized, as shown in Figure 5. This behavior lasted approximately 24 hours until the battery sub-assembly on panel #1 had a catastrophic failure as a result of a short circuit. As the energy in the cell was discharging internally through the short circuit, there was a dramatic increase in the battery temperature and pressure as hydrogen gas was being generated from electrolyte decomposition. This process continued until there was a breach of the battery case and a release of this super-heated gas internal to the ESEX flight unit on 22 Apr 99 at 1501 Z. This gas release was likely directed at panel #3, since a 40 °F increase in temperature on this panel was observed over the course of a few seconds. This gas was eventually vented into space, causing a dramatic attitude disturbance

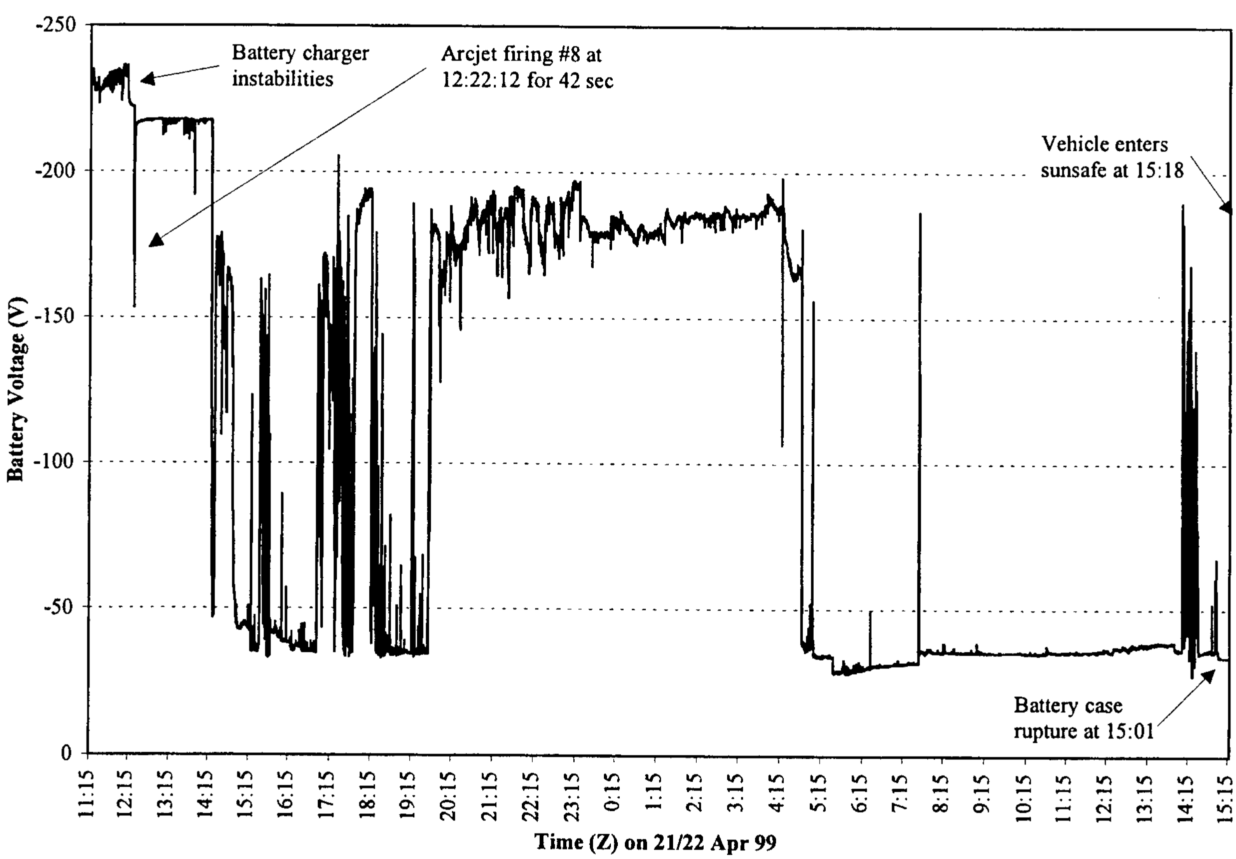

Figure 5. Battery Voltage Fluctuations Following the Final Arcjet Firing (#8) Ultimately Resulting in the Short Circuit

on the vehicle just prior to an umbra crossing. The disturbance was violent enough to saturate the ARGOS reaction wheels and cause the vehicle to spin about the yaw axis at 1/8 rpm through the following eclipse period. This spin resulted in the vehicle automatically switching to a safe mode known as "sunsafe" which sheds non-essential power loads and points the solar arrays at the sun to maximize chances of survival. Once ARGOS exited eclipse, the vehicle was recovered through normal recovery procedures, and no long-term impacts have been observed.

The on-orbit anomaly was actually the second anomaly experienced by the ESEX flight battery, as the first was during final integration just prior to satellite shipment. During the final closeout of panel #5, the harness between the battery and the PCU was pinched between the panel and the ESEX structure – causing a short of the battery to ground (since the flight unit was grounded). Although it was at a very low state of charge, there was very high current drawn out of the battery. As a result, four of the 126 individual cells were discharged to less than 0.5 Vdc (from 1.5 Vdc) – one on the sub-assembly on panel #1, and three on the sub-assembly on panel #5. These cells were charged back up with an external charger and stabilized (cell #4 on panel #5 had to be recharged twice). It is not clear whether this event had an effect on the battery resilience, but there was almost certainly some stress induced from this event. Furthermore, the engineering model battery (the same one used in the IMS test) had a catastrophic failure just prior to the ESEX thermal-vacuum test at TRW.[1] After a series of bench-level tests and analyses, this failure was attributed to exceeding the specified life of the battery, and did not suggest any design flaws, or integration issues. This event did not have significant bearing on the troubleshooting of the on-orbit anomaly, nor does it have an impact on the outcome of this study since the cause was clearly a result of use beyond the specified life.

2.4 Post-Short Circuit Voltage Rise

Following the short circuit and the vehicle recovery, the battery voltage was stable at -39 Vdc, which suggested the battery sub-assembly on panel #1 was shorted to chassis around cell #39 and the interconnect between cells #39 and #40 was open as a result of the catastrophic failure. Since the ESEX mission was concluded, the temperatures of each of the three sub-assemblies were reduced to enable thermostatically controlled switches for the discharge resistors. When the switches closed, however, the indicated battery voltage slowly increased to -63 Vdc. The short-circuited cell in the battery sub-assembly on panel #1 established a charge path through its discharge bleed resistor from the battery sub-assemblies on panels #4 and #5 of approximately 0.057 A. The voltage of the cells above the short would slowly increase as the top 39 cells were charged – in this case to approximately -1.6 Vdc per cell, or -63 Vdc total (assuming 39 cells). Since the integrity of the battery was somewhat unknown, and in order to avoid any further incidents, the discharge resistor on the sub-assembly on panel #1 was opened to eliminate the charge path. The panel #1 resistor was closed again once the panel #4 and #5 sub-assemblies were discharged below the voltage of the 39 cells above the short (less than 63 Vdc) on panel #1, and the remaining energy was depleted.

3.0 FAILURE SCENARIO

The ultimate failure of the ESEX battery was a result of electrolyte leakage, causing a short circuit from one of the cells on panel #1. This anomaly initially caused an intermittent short, and ultimately a hard short circuit to the battery case – a scenario that has been observed on other programs. The anomalous behavior, including the activity before the short circuit, was a result of at least one of two problems. First, the battery impedance increased as the mission progressed, reducing the ability of the flight charger to adequately charge the battery. Second, some effect was responsible for rupturing at least one of the battery cells, causing an electrolyte leak that resulted in the short circuit. There are two potential sources for these effects:

- As a result of the low on-orbit charge current, the internal battery impedance degraded over time as a result of gas buildup. This increased impedance resulted in a degraded state of charge.

- The mechanical cell interconnects degraded, resulting in increasing battery impedance and localized heating – ultimately rupturing a cell.

Either of these scenarios would fit some of the telemetry data, but neither can exclusively account for all of the behavior observed. It is likely that a combination of effects resulted in the battery failure, and it is possible that the ultimate problem was exacerbated by an external phenomenon. These scenarios are discussed below.

3.1 <u>Increase in Internal Battery Impedance</u>

The first proposed failure mechanism is a chemical change in the battery that resulted in increased internal impedance. This change is likely to have occurred as a result of the on-orbit charging profile, especially since the charger output current was so low, but could also have been a result of the high discharge currents. Typical uses for Ag-Zn batteries are single-use, relatively low discharge current applications, with fairly high recharge rates (1-10 A). The ESEX battery was repeatedly discharged at 180 A, and recharged at a rate of only 0.55-0.69 A. This combination could produce some detrimental chemistry inside the battery cell that would lead to increased impedance as the mission progressed.

To further support this possibility, recent data on Ag-Zn batteries show a distinct increase of internal cell impedance as the battery approaches full charge when charged at low rates. This behavior was not observed during the ground testing because the majority of the battery charging was accomplished at a high rate with an external charger in order to expedite schedule. The battery was never charged for longer than 30 minute increments at the low rate supplied by the flight charger. The degraded impedance would explain the charger instabilities and result in a degraded state of charge. The higher impedance, however, would probably not have resulted in the catastrophic failure seen after firing #8.

3.2 <u>Mechanical Interconnect Failure</u>

The increased impedance could also be explained by a degradation of the cell interconnections, resulting from either a design flaw or from higher-than-expected launch loads. Since the mechanical interconnections were encapsulated in the potting epoxy, any stress buildup (from thermal or dynamic loads) could have caused the interconnections to loosen, leading to increased impedance. Furthermore, this design could have been subject to cold flow of the interconnections, or encapsulate wicking into the mechanical interface to the cell terminals or between the layers of the interconnecting straps. This type of failure will characteristically be unstable and the impedance will vary inversely with temperature. As the battery and terminal temperatures cool, the mechanical pressure relaxes and the impedance increases. During times of increased temperature, such as localized ohmic heating from high charge or discharge rates, the impedance will decrease. The electrical connection will degrade with each subsequent cycle – especially at these high discharge current rates – as a result of cold flow of the interconnection-to-cell interface and oxidation buildup from the localized heating. The problem is compounded on each cycle until the contact impedance is so high that it prevents any energy transfer into or out of the battery – preventing charging or firing. In this scenario, the ohmic heating would eventually be enough to rupture the cell, causing electrolyte leakage, and a short circuit to the battery case. The fluctuating behavior of the battery voltage is characteristic of an electrolytic "connection" to the case which is intermittent as electrolyte is boiled off and replaced by the leaking cell. The proposed failure location is near cell #39 in the sub-assembly on panel #1.

With an average charge rate of approximately 0.5 A and an impedance of 58 Ω (calculated from the occurrences when the charger tripped off), the dissipation at the cell terminal would be 14.5 W. This would result in localized heating of the terminal, temporarily improving (reducing) the contact impedance to allow continued charging, but not allow a true charge of the battery because of the voltage drop and the power lost in the contact impedance. On each subsequent discharge cycle, the contact impedance would degrade due to localized heating of the terminal from the high discharge current rates. Assuming a degraded contact impedance of 55 mΩ, the battery output voltage would be reduced by 10 Vdc and the power dissipation at the terminal would approach 1800 W as the arcjet current ramped up to full power.

After the first three firings, the heat load from the degraded resistance would further deteriorate the contact until extended firings could no longer be sustained (i.e., firings #4-8). Ultimately, on firing #8, the localized heating on the terminal was enough to cause the cell to rupture and leak electrolyte, resulting in an intermittent low impedance path to the panel #1 sub-assembly chassis. This low impedance path would be present intermittently as electrolyte leaked out of the cell, was boiled off, and then replaced as more was drawn into the shorting site.

With a low impedance path to chassis, the battery voltage telemetry would indicate the voltage of the cells above the short circuit. Assuming a low state of charge at this point (approximately 1 Vdc per cell), the telemetry measurement of -39 Vdc would indicate the short circuit occurred around cell #39. The battery voltage became unstable due to the varying impedance of the terminal contact being heated by the "charging" into the short from the battery sub-assemblies on panels #4 and #5. As the unstable connection passed current from the panel #4 and #5 sub-assemblies into the short, the battery voltage increased toward 180 Vdc and the connection heated. As a result of this heating, the impedance dropped, resulting in lower thermal losses – which led to a colder connection, higher impedance, and a battery voltage drop. This continued at a cyclic rate as observed between 21 Apr 99 at 1300 Z and 22 Apr 99 at 0740 Z, as shown in Figure 5. During this time, battery pack 1 was slightly higher in temperature (10-20 °F), further verifying that some localized heating was occurring in the sub-assembly on panel #1. Eventually, this intermittent short would lead to a wholly metallic path as zinc or silver is plated out of the electrolyte, and result in a hard short circuit.

The source for the initial stress that caused a deteriorated contact is unknown, but seems to be unique to the flight configuration since this behavior was not observed during the IMS test of the engineering model arcjet, cable, and PCU. It is possible, however, that the IMS test configuration masked the problem since the engineering model battery was not subjected to the same environment seen on-orbit. All of the ground test firings were performed with the battery at atmospheric pressure in a horizontal position. Since the battery was not under vacuum during any of the IMS firings, for instance, a better heat path may have been present to accommodate any thermal losses from ohmic heating. Another possibility is that there was enough localized heat generated by on-stand charging cycles conducted prior to launch to affect the connection. This scenario, however, seems unlikely since the charging cycle duration was limited to 30 minutes separated by 60 minutes of down time. Further analyses of these charge cycles show that no obvious anomalous behavior was present. The charger output current fluctuates very little between 0.67 and 0.68 A and the battery panel temperatures remain essentially constant. This short charge duration, however, does not lend itself to a complete analysis, since on-orbit data show nominal behavior for the first 30 minutes in almost all cases. Excessive launch loads are another candidate source, but there were no indications of higher-than-normal loading on any of the launch vehicle sensors, nor on any of the other payloads.

3.3 Other Effects

The data seemed to suggest some further problem contributing to the final failure. One failure mode previously observed on vented Ag-Zn batteries is the release of electrolyte during launch depressurization and again during high rates of discharge as the cell heats rapidly and has an internal pressure increase (see Appendix). Since the ESEX battery cells lay on their side during launch, it is probable that some electrolyte was expelled during ascent into the cavity above the encapsulate and absorbed into the matting material. Over time and with additional high current firings (as more electrolyte was expelled), the electrolyte could have wicked to the wall of the battery case and into the porous encapsulate material around the terminals. If the electrolyte reached the copper terminal interconnects, the electrical connection would certainly degrade. This would most likely occur at cells closest to the battery housing (see Figure 2 for the cell layout), such as the proposed cell #39. Any electrolyte that bridged the cell terminals to the case initially would probably boil off due to the bulk power available from the batteries. As the cell interconnect degraded, however, localized heating of the terminal during the firings may have caused cell #39 to rupture and expel a large amount of electrolyte. The battery voltage data following the last firing suggest that the interconnect impedance between cell #39 and #40 increased to approximately 10-100 Ω, and a low impedance connection was established between the positive terminal of the cell and the battery case – presumably due to the expelled electrolyte. During this time, the panel #1 temperature was slightly higher than the other battery panels, probably caused by the battery sub-assemblies on panels #4 and #5 boiling off the electrolyte from cell #39.

As described above, the battery voltage stabilized near -39 Vdc six to nine hours prior to the catastrophic failure, which indicates a hard short from cell #39 to the battery case. Since this configuration remained stable for such a long time before the major anomaly, some other phenomenon likely caused the catastrophic failure. One possible explanation is that the expelled electrolyte from cell #39 slowly wicked across the matting material or the encapsulate to cell #34, which is next to cell #39 in the battery sub-assembly. As the electrolyte bridged the terminals, another short circuit developed across cells #34 through #39, and one of the cells violently erupted. This event, then, would have been the source of the battery case rupture and the release of the super- heated gases onto panel #3.

4.0 LESSONS LEARNED

During this review, two primary issues were identified as the ultimate source of the battery failure. The first was a lack of an adequate review of the battery design, and the second was an inadequate execution of the flight qualification testing. As indicated above, the battery used for the ESEX experiment was a similar design to other Eagle-Picher batteries, but was used for a more intensive application. Due to the cost constraints on the program, only a limited number of qualification tests were performed on a single test module. These tests were all completed successfully which led to a complacency about the battery capability. This complacency, in turn, resulted in a less-than-complete design review, which could have potentially identified the cell interconnections, for instance, as a potential problem area. Furthermore, this complacency resulted in a less-than-complete test program, such as a complete qualification test or a flight-like charging test with the flight charger circuit design, which could have possibly identified the root cause of this anomaly.

Recently, several reports have surfaced that identify experiences similar to the ESEX battery failure (see Appendix). This information, along with any further expertise from The Aerospace Corporation, would have been invaluable during the ESEX design phase, since several improvements in the design could have extended the battery life significantly. For example:

- The charger circuit would almost certainly have performed better at a higher charge rate, and would not have caused an increase in battery impedance. The low charger current was dictated by the available ARGOS power and the thermal dissipation requirements, but the high sensitivity of battery performance to charger current was not known. A higher output current could have been implemented while maintaining the average power and thermal constraints simply by pulsing the circuit.

- The stability of the charger circuit could have been addressed to some degree, if it was known the low charge rate would result in a higher battery impedance. While it is not clear that charger circuit instability had any direct effect on the battery anomaly, this issue could almost certainly have been addressed prior to the flight operations had it been discovered during ground test.

- An external compartment to hold expelled electrolyte away from the cells could have been implemented, which might have protected the battery from internal degradation.

- Improvement to the cell interconnects is a must for longer, adverse mission requirements such as that on the ESEX mission. Previous experience from TRW or The Aerospace Corporation almost certainly would have identified this as a risk item had a thorough design review been performed.

- Electrical isolation of the battery chassis from the structure may also have given an extra level of protection (a common practice on all TRW NiH battery designs).

As this battery was not truly a heritage design for this mission, a full high-fidelity qualification test program with true orbital conditions would probably have uncovered many of these problems.

Ultimately, this anomaly occurred because of a programmatic philosophy to minimize cost. Of all of the failure scenarios presented here, all of them could have been ruled out if enough testing had been done to adequately describe the battery performance.

5.0 IMPACTS ON EXISTING OR FUTURE SYSTEMS

Since this application was so unique, there is no impact to any existing systems as a result of this anomaly. Most launch vehicles, for instance, only require 10-15 minutes of total battery life, and require discharge currents an order of magnitude less. The ESEX application of this battery was a high current (~180 A) discharge, a low current recharge, and a relatively large number of charge/discharge cycles at these rates. Future systems using any type of batteries at this level of discharge current must consider the implications of all mechanical interconnections. Furthermore, the lack of adequate testing in a flight configuration clearly exacerbated the problem. Had testing been accomplished to demonstrate the flight charger with the flight battery in flight-like conditions, this behavior would have almost certainly been discovered and resolved.

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APPENDIX

INTRODUCTION

The following contains additional information on similar anomalies associated with silver-zinc batteries, as well as a review of this article by personnel at The Aerospace Corporation. While not inherently extensive, these articles lend further credence to the conclusions reached here, as well as provide additional references for more detailed analyses.

DESCRIPTION OF PROBLEM AND CAUSE

Flight Manifestation of Problem

At some ten hours after lift-off and some four hours of post-deployment 14A current load, a 10-cell 145AH battery (one of two in series) began dropping in voltage, increased its rate of temperature rise, developed a noisy output voltage, and then went through numerous brief voltage drops, followed by loss of the power buss.

Root Cause at System Level

This problem appeared at a system level because the qualification testing conducted did not fully test the battery under flight conditions of usage: longest mission duration under vacuum, maximum anticipated temperature and current. Active thermal vacuum testing was conducted per MIL-STD-1540A. That testing, as defined in battery Critical Item specifications, did not include full or mission-profile discharge with a vacuum applied to the battery case vent valve under adiabatic conditions or with the mission-profile temperatures simulated.

Root Cause at Component Level

Root cause of the battery failure was electrolyte escaping from battery cells and forming a bridging path between battery cells and ground. No means of containing the escaped electrolyte was available and the epoxy coating contained thin spots and bubbles, which allowed a path to form between the cells and the uncoated grounded battery case.

Deposition of zinc from zincate in the electrolyte and of silver deplated from cell terminals created a spurious electrical path; conduction in that path generated heat, which was capable of boiling (or electrolyzing KOH in) the electrolyte and eventually producing zinc and silver tracks on top of the potting. It is believed that there was enough heat generated to carbonize and/or distort the cell casings and the epoxy potting.

Detailed Considerations

The Zn/AgO battery cells generate hydrogen gas at the zinc electrode on open circuit stand and on discharge. The anomaly investigation team reports that the rate of gas evolution increases significantly as temperature is increased. The rate of gas evolution is on the order of 1 to 10 standard cc/hour per cell.

The Zn/AgO battery is exothermic on discharge. Approximately 20 watts of heat is generated for each 100 watts discharged. The present battery design allows for use in an adiabatic installation; the heat generated on discharge increases the battery temperature, which increases the gas generation rate.

Each battery half contains 10 cells. The battery is not sealed. Valves are placed within the battery to prevent battery dry out due to water loss and to limit the expulsion of corrosive battery electrolyte. There is a battery case valve (8 to 3 psia specification range) and individual cell valves that have a 10 to 0.5 psia specification range.

On ascent, there is venting to equalize the battery and cell pressures as the mission progresses. Gassing is dependent on current and temperature, generally increasing with both. Entrained electrolyte may enter into the headspace of the battery as gases are expelled from the cells.

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Theory of flight failure

Cell and battery case venting during ascent depressurization probably occurred, but without significant ejection of electrolyte. Subsequent application of load for a continuous extended time period caused an increase in temperature and in the rate of hydrogen evolution. When the cell vent valves again opened, under zero gravity, electrolyte was entrained in the vented gas. Epoxy potting had defects in its coverage of the terminal nuts and the tin-nickel-plated magnesium battery case. The ejected electrolyte formed a current path from a cell terminal to the case, possibly aided in its migration by vehicle maneuvers.

Four hours post-deployment, but before the cell-to-case short, a battery internal sensor indicated 46 C. The heating due to the cell to case short added heat to both half batteries. Localized heating in the vicinity of the short would have increased the temperature of nearby cells in the upper half-battery, further increasing their ejection of electrolyte, thus perpetuating the short.

The first shorting path would be wholly electrolytic, and thus prone to periodic open-circuits due to electrolytic bubbling or ionization. Testing has shown that electrolyte is drawn into the shorting site, causing ready reestablishment of the short. Repetitions of these events would yield the noisy output voltage that was among the first in-flight manifestations of the problem.

Zinc, possibly dendritic, as well as some silver from the terminals, plated out in periods when an electrolytic, conductive path existed There may have been periods when the electrolyte film evaporated after which silver residue might be left on the epoxy. An increasing portion of the conductive path would then be metallic rather than electrolytic.

The first occurrence of a wholly metallic path, while causing larger voltage drops than the electrolytic path, would probably be of small cross section and likely to burn away. Again, electrolyte drawn into the gap would replate the burned-out part of the path, eventually leading to a sustained short.

Ground based testing has demonstrated these phenomena.

SPECIAL CONDITIONS APPLICABLE TO THIS PROGRAM

The length of the mission, adiabatic thermal conditions, with attendant current draw and exposure to vacuum and zero gravity for a much longer time than most launch vehicle batteries; the latter must operate for less than 15 minutes. Increased quantity of electrolyte, added to cells to achieve longer wet life, increased the likelihood of electrolyte ejection without attendant addition of electrolyte containment features external to the cell.

ACTIONS TAKEN OR PROPOSED FOR THIS PROGRAM

Rework of existing assets

Apply an insulating coating to the exposed portion of the battery case.

Increase depth of potting material to the tops of the cells to prevent escaping electrolyte from contacting the cell terminals or the connecting straps.

Add an absorbent pad between the tops of the cells and the lid of the battery box to contain any expelled electrolyte.

These steps will be followed by a requalification program to verify the effectiveness of the modifications.

Changes for new assets,

Apply an insulating coating to the entire inside surface of the battery.

Add a chimney to cell cover; this will allow for self-fixturing of epoxy operations, augmenting the potting dam now used.

Increase thickness of potting over cell terminals.

Add an absorbent pad as described above for existing assets.

RECOMMENDATIONS FOR OTHER PROGRAMS:

Present Batteries

Programs should review the procedures used during the thermal vacuum testing that was carried out as part of their qualification program for present batteries, verifying the validity of that testing to replicate the worst case mission scenarios.

Factors of particular importance include the following:

- Battery wet stand (with temperature) and vibration environment.
- The length of the period in which the entire battery is exposed to vacuum.
- The length of the period in which mission-profile load is applied to the battery.
- The flight thermal environment should be simulated in the qualification.

Future Battery Builds

Design future batteries to be more robust and fault-tolerant and test batteries in the flight configuration to worst-case mission profiles and environments anticipated. Programs should also review their cell and battery designs and evaluate the electrolyte expulsion characteristics of the cell-types used in their programs.

Program Offices may elect to make modifications to the cell vent now in use to prevent the expulsion of electrolyte during battery operation.

Consideration should be given to providing insulation of the battery case and terminals from inadvertent shorting paths.

Systems Issues

Avionics systems should be evaluated for robustness in respect to the intermittence of supply voltage that is likely during the first phase of Zn/AgO battery failure. The number and the durations of power interruptions that may occur before final failure are unpredictable. Special attention should be paid to the operation of systems that automatically switch modes or employ redundant resources.

THE AEROSPACE CORPORATION

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SUBJECT: Comments on the material related to the ESEX battery failure

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SUMMARY

The write-up covering the ESEX flight battery failure has been reviewed. The findings contained in this report prepared by the Air Force are in keeping with findings associated with other silver-zinc battery failures. The "Lessons Learned" section contains statements with which this author can thoroughly agree. Some of these suggestions have already been incorporated into flight batteries as a result of earlier problems. As an example, following the IUS failure, the internal surfaces of the battery boxes are now painted with a dielectric coating.

INTRODUCTION

The ESEX battery failure report as prepared by Mr. Bromaghim has been reviewed. The following comments are based on the author's association with some of the earlier discussions with Aerospace and TRW personnel during the April/May 1999 time frame when this issue arose. They also include background information based on earlier in-flight and on-the-pad failures of silver-zinc batteries caused by electrolyte being expelled from the cells and initiating destructive electrical short circuits. This background information supports the findings as presented in the Air Force report. Since an electrical short circuit is one of the suggested causes of the problem experienced aboard the ARGOS spacecraft, the results of the extensive failure analyses carried out during these earlier failure analyses will be summarized here. Probably the AF/Aerospace Program Office team is fully familiar with the earlier failure of the ESEX battery several years ago during some laboratory testing. The details of that failure will not be covered here, but do attest to the destructive consequences of silver-zinc battery failures.

ATLAS PYRO BATTERY FAILURE

In the 1995 time frame, there was an incident at Vandenberg during some prelaunch activities associated with an Atlas vehicle. The battery in question had already exceeded its wet life limit of 15 or 30 days so this was not a failure of a flight battery. But, since the battery was still onboard the vehicle and more ground testing was to be done prior to launch, the battery was left aboard the vehicle for several more days. Several days later, the battery voltage abruptly dropped to 0.0 volts. Upon removal from the vehicle and a closer inspection of the internal portions of the battery, it was found that a destructive fire had taken place within the battery in the vicinity of the power wires attached to the backside of the J1 connector.

As part of the failure investigation, the staff of the Energy Technology Department secured a spare battery of the same model. The going-in supposition was that a few drops of electrolyte had escaped from one or more of the cells and made its way via capillary action along the power wires going to the J1 connector. Once the electrolyte was able to bridge between the +28 volts of the battery and the grounded shell of the connector, electrochemical reactions occurred which led to a fire that resulted in the melting of the connector pins and some of the wires. This sequence was replicated first using connectors onto which electrolyte was dripped. This was followed by a series of experiments leading to a video-taped destructive connector fire at the J1 connector of the spare pyro battery. This experiment convinced most people that the Vandenberg incident had been replicated. Following this, an instructive teaching video was prepared and distributed to interested personnel showing the destructive consequences when small amounts of electrolyte contact the positive and negative portions of a connector.

IUS-13 IN-FLIGHT FAILURE

In the same general time frame as the Atlas pad failure, an in-flight failure occurred during the transfer orbit of IUS-13. Since these vehicles have a flight duration measured in hours rather than just a few minutes, like many launches, there is more time for things to happen. Telemetry gave indications that the temperature in the battery was rising at a significant rate. The bus voltage dropped from its normal level of 29.6 volts to 29.1 volts. This was indicative of about a 10-ampere short circuit. The bus voltage held steady for about a half-hour at a reduced level, and then began to drop down over the next half-hour until it reached the low voltage limit of the computer. At this point the "B" bus was lost. An in-flight failure creates much more concern compared to an on-pad failure. The symptoms of this failure pointed to a short circuit to ground. To investigate this more fully, the IUS Program Office had the battery manufacturer repeat the flight profile using another battery. The test was carried out in a vacuum chamber. The experiment replicated the failure as experienced during the actual flight. Photographs gave ample evidence of the short circuit that went from one of the electrical terminals of one of the cells to the grounded battery box. The creation of the shorting path was a consequence of electrolyte first being expelled from one or more of the cells out onto the tops of the cells. The electrolyte, being an excellent ionic conductor, set up a leakage path across which an ionic current was able to flow. This would not result in a battery failure since the current flowing through an electrolyte film would be insignificant relative to the capability of the battery. However, in a discharging cell, the electrolyte in this type of cell contains a soluble form of zinc – zincate ions. This zinc is plated out across the shorting path formed by the electrolyte film. When a path is completed, a metallic zinc bridge is formed which permits a larger current flow to occur. This larger current flow will burn out the zinc dendrite conduction path. As this happens, a portion of the epoxy potting material is charred in the process. Following many of these processes, a complete conduction path based on the carbonized pathway between the cell terminal and the grounded battery box is established. The carbonized epoxy has a significant higher current carrying capability compared to the zinc dendrites. The photographic evidence from the experiment set up by the IUS Program Office supported this scenario. Following this, a study within the Energy Technology Department culminated in a report TR-97 (1494)-1, entitled "Electrolyte Loss Tendencies of Manually Activated Silver-Zinc

Cells and Batteries" authored by L. Thaller and G. Juvinall. This report presented a technique for determining whether the cell design under consideration might expel electrolyte during the course of a flight to an altitude where the vent valves of the cell and the battery box would be expected to open.

THE ESEX FAILURE

The ESEX failure suggests an electrolyte expulsion problem as was concluded in the write-up. In Figure 3, the first indications of problems are evident. Between 5:33 and 6:03 Z, the battery voltage is seen to drop about 6.0 volts. The granularity of the data appears to be too course to be able to see a short circuit within a single cell. The 6-volt drop suggests a short circuit to ground. In addition, there could have been a short circuit within a cell that actually melted the cell case allowing electrolyte to leak out and form a shorting path to the grounded battery box. That is, the electrolyte path could come from a hole melted in the cell case or from electrolyte being expelled out through the cell vent. Since a hole in the cell case resulted in the failure of the ESEX battery several years ago, this same problem may have initiated the short to the battery box in the flight battery. In Figure 5, the short to ground has grown to a much more significant short circuit. Silver-zinc cells have the capability of extremely high rates of internal discharge during a short circuit inside a cell. They have been known to catch fire during these events. The cell cases are made from combustible plastic materials. At the elevated temperatures of a fire, the silver oxide material will disassociate liberating oxygen that will encourage the fire. All of this thermal energy can produce steam as the electrolyte is heated above its boiling point.

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