BEA
Bureau d'Enquêtes et d'Analyses
pour la sécurité de l'aviation civile
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TECHNICAL REPORT

Accident on 23 July 2020 at Bâle-Mulhouse airport (Haut-Rhin) to the PIPER - PA-28RT-201T registered HB-PNP

Examination of wreckage

RÉPUBLIQUE FRANÇAISE
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Photo: Marcel Moser

Foreword

This document and the photographs and technical information contained herein are subject to the laws relating to communication and confidentiality embodied in European Regulation 996 of 20 October 2010.

The conclusions of this document are based on the work undertaken by the BEA. They should not be used to prejudge the final conclusions of the safety investigation.

SPECIAL FOREWORD TO ENGLISH EDITION

This is a courtesy translation by the BEA. As accurate as the translation may be, the original text in French is the work of reference.

Contents

Glossary

NTSB	*National Transportation Safety Board* The National Transportation Safety Board is an independent federal agency charged by Congress with investigating every civil aviation accident in the United States and significant occurrences in other modes of transportation: rail, transit, highway, marine, pipeline, and commercial air.
CEA	Commissariat de l'Energie Atomique
STC	*Supplemental Type Certificate* A Supplemental Type Certificate is a document issued by the United States Aeronautics Administration confirming authorization to modify an aircraft or aircraft equipment.
EPI	First Information Investigator
ABS	Acrylonitrile Butadiène Styrène ABS is an industrial thermoplastic and amorphous polymer combining three different monomers: Acrylonitrile, Butadiene and Styrene.
INRS	Institut National de Recherche et de Sécurité
Cold Cranking Amps	Ability of a battery to start an engine in cold weather
Rated Capacity C1	A rate of 1C means that the discharge current will discharge the entire battery in 1 hour. For a battery with a capacity of 100 amp hours, this equates to a discharge current of 100 amps.
Parts Catalog	A Parts Catalog is a book published by a manufacturer that contains illustrations, part numbers, and other relevant data for their products or parts thereof.
EFTE	Ethylène Tétrafluoroéthylène ETFE is a thermoplastic fluoropolymer.
Batterie AGM	*Absorbed Glass Mat* This battery technology does not contain free liquid electrolyte. The electrolyte is contained in fiberglass blotters, located between the electrodes.
EDS	*Energy Dispersive Spectrometry* This method consists of analyzing, using a spectrometer, the X-rays produced by a sample subjected to the bombardment of an electron beam. The energy spectrum of this radiation is characteristic of the differences between the electronic energy levels of the emitting atoms, and it therefore makes it possible to identify the atoms present in the sample.
SOH	*State of Health* Battery "health" status
SOC	*State of Charge* Ratio of battery charge to nominal capacity
DOD	*Depth of Discharge* Ratio between the already discharged capacity and the nominal capacity of the accumulator

1 - AEROPLANE INFORMATION

	PIPER – PA-28RT-201T registered HB-PNP
	 Source: https://www.jetphotos.com/registration/HB-PNP
Manufacturer	Piper PA-28
Type number	PA-28RT-201T
Serial number	28R-8131037

2 - WORK PERFORMED

2.1. INTRODUCTION AND DESCRIPTION OF INVESTIGATION PROCESS

The BEA represents France as the state of occurrence of the accident. Investigators from the NTSB representing the United States of America as the state of manufacture of the aircraft, the aeroplane manufacturer, PIPER, the aeroplane battery manufacturer, Concorde Aircraft Batteries, the CEA laboratory in Grenoble, along with an airport firefighter and fire analysis expert took part in this investigation.

The coordination between the BEA and the American manufacturers (Piper and Concorde) was facilitated by the NTSB.

2.1.1. Preliminary examinations on site

The examinations started on the accident site at Bâle Mulhouse airport. They consisted of a general visual inspection of the wreckage (see paragraph **2.2** for details). The wreckage was then prepared in order to transport the aircraft fuselage to the BEA premises.

The examinations carried out on the BEA premises were broken down into four phases described below and identified A to D. Phases B, C and D were carried out in parallel, phase B being the longest and most complex.

2.1.2. PHASE A: location of start of fire

Based on the visual examination of the damage observed on the wreckage and the characterization of its severity (see paragraph **2.3** for details about the work), the start of the fire was defined as being aft of the luggage hold, in the area containing various systems associated with the general electrical system supplying different aircraft equipment, including the battery and various contactors.

To characterize the damage observed, the BEA relied on the expertise of a firefighter specialized in this type of investigation procedure.

2.1.3. PHASE B: examination of general electrical system supplying aircraft equipment

After it had been determined where the fire started, the equipment and associated systems present in the zone where the fire started were examined. This phase consisted of assessing the general electrical system supplying various aircraft equipment items, which included energy sources and various components supplied with electricity (see paragraph **2.4** for details).

The aim was to:

- compare the aeroplane’s electrical system with that specified by Piper;
- assess the condition of the various electrical power sources used during the accident flight: battery and alternator;
- identify the damaged areas and systems;
- characterize this damage by defining the damage resulting from the accident and the damage prior to the accident.

On completion of this work, it was observed that several STCs had been embodied outside the specifications initially defined by Piper.

This work found that the alternator and its regulator ware functional, undamaged and that the abnormally high current seen by the pilot during the flight corresponded to the maximum performance level of the alternator.

The most singular damage was observed on the battery and an associated diode device which underwent more in-depth work (see paragraph **2.4.4** and **2.4.5**).

2.1.4. PHASE C: engine start-up system

During the examination work of the general electrical system (phase B), the BEA exchanged with Piper and the manufacturers responsible for the STCs. Thought was especially given to the probable causes of the abnormally high current in the general electrical system observed by the pilot during the flight, before the first signs of a fire were identified. Among the hypotheses examined, a fault on the engine start-up system appeared to be a possible cause. This fault had already been examined by the manufacturer responsible for the STC associated with the aeroplane’s battery.

The aim of the work on the engine start-up system (see paragraph **2.5**) was to:

- compare the aeroplane start-up system with that specified by Piper;
- assess the condition of the various components;
- identify the damaged areas and systems;
- characterize this damage by defining the damage resulting from the accident and the damage prior to the accident.

This work found that the start-up system was functional and undamaged prior to the accident.

2.1.5. PHASE D: autopilot system

The accident flight was the first flight performed by the aeroplane following work to install a modern avionics suite and a new autopilot. These systems were specifically examined in detail (see paragraph **2.6**) in addition to the work carried out in phase B.

This work did not find damage prior to the accident.

2.2. PRELIMINARY EXAMINATIONS: visual observations and preservation of wreckage

2.2.1. Visual observations on accident site

A BEA field investigator was present during the initial observations at Bâle Mulhouse airport (**Figure 1**)

Once these initial observations had been completed, the aeroplane was moved to the entrance of a hangar used by the ARFF (**Figure 2**).

Figure 1: Aeroplane at the edge of the runway at Bâle Mulhouse airport on the day of the accident
Source: BEA - photo taken day of accident

Figure 2: Aeroplane transferred to entrance of an ARFF hangar
Source: BEA - photo taken day of accident

On the day of the accident, at the edge of the runway, the following observations were made.

- The aircraft was on its landing gear. No major part had been lost in flight. Despite the damage, the airframe was practically complete.

- The outer surface damage was concentrated on the right side of the aeroplane, on the rear section of the luggage hold and around the rear right passenger window (**Figure 1**).

- The interior of the cockpit was covered with soot, extending from the instrument panel to behind the luggage hold. The seats were partially burnt (**Figure 3** and **Figure 4**).

Figure 3: Interior of cockpit, 3/4 view from rear right
Source: BEA - photo taken day of accident

Figure 4: Instrument panel
Source: BEA - photo taken day of accident

- The left side windows and the right side rear windows were broken or had melted.

- On the instrument panel, the controls were found in the following positions (**Figure 5**):

<u>Note:</u> The positions below are those identified after the operations carried out by the rescue services. The hypothesis that these controls were modified between the accident and the photo below being taken is unlikely but it cannot be excluded.

Item	Controls	Position of control
1	Master BAT (battery)	ON
2	Master ALT (alternator)	OFF
3	AUX FUEL PUMP	LOW
4	LANDING LIGHT	OFF
5	LIGHT (*)	OFF
6	PITOT HEAT	OFF

** Indication not visible making identification impossible*

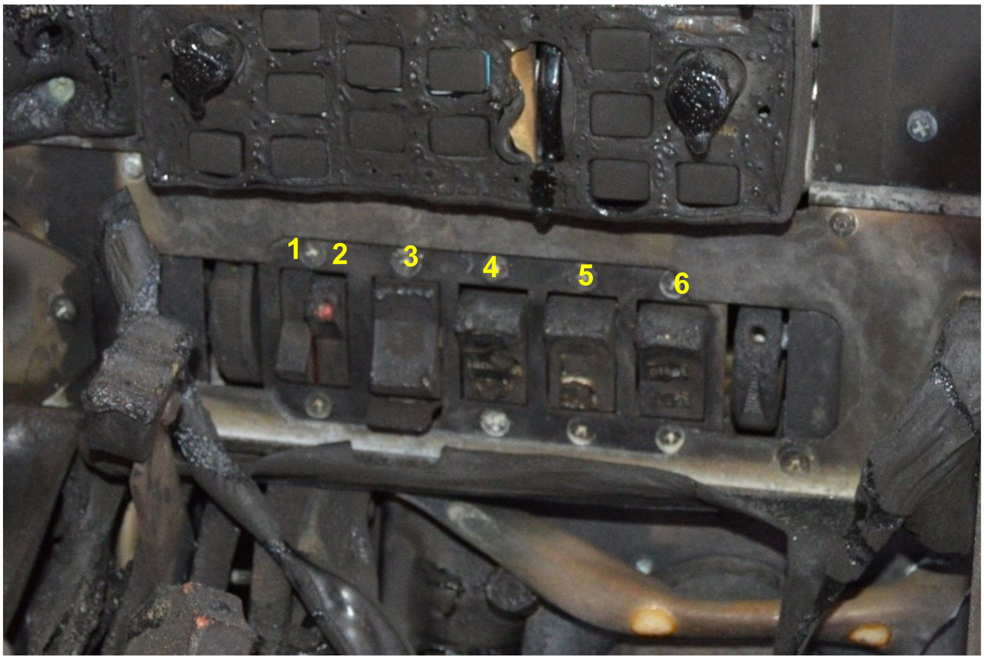


Figure 5: Instrument panel controls
Source: BEA

No circuit breaker was found open on the instrument panel (**Figure 6**).

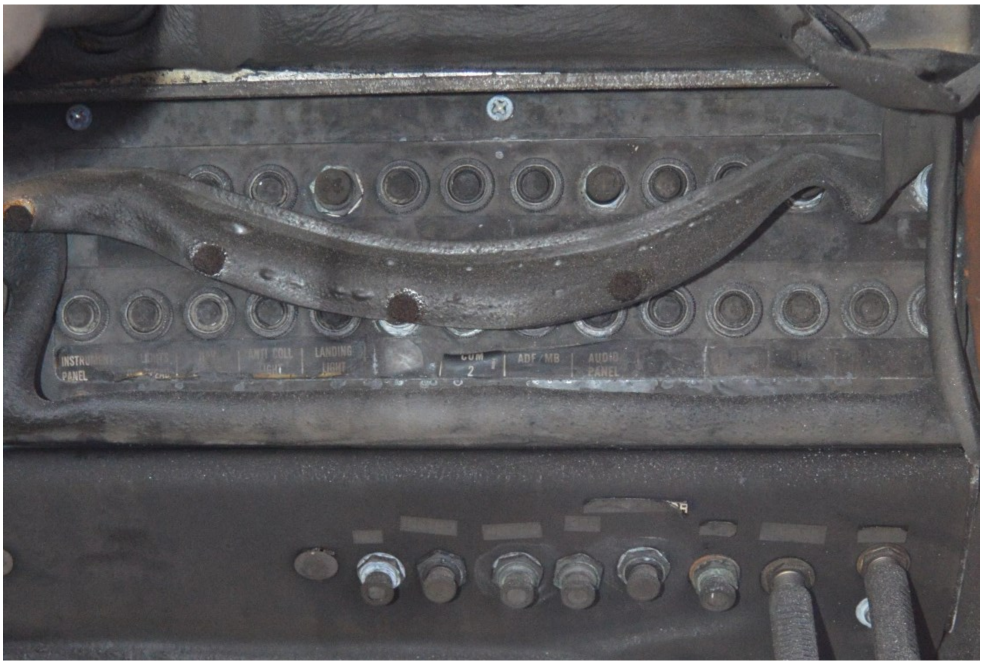


Figure 6: Circuit breakers on instrument panel
Source: BEA

The cockpit fan control switch was set to OFF.

2.2.2. Preservation of wreckage and transportation to BEA hangar

In order to facilitate the transportation of the wreckage to the BEA's premises, the two wings were cut off at their root. The decision to cut off the two wings was taken once it had been validated that these parts of the aircraft were not implicated in the fire phenomenon.

In order to preserve the fuselage of the aeroplane during its transportation, it was completely covered in several layers of thick plastic sheet (**Figure 7**). The aim of this protection was to preserve the fuselage in its original state without altering it.

The protected fuselage was transported by BEA personnel into the BEA investigation hangar.

Figure 7: Fuselage protection for its transportation to the BEA premises
Source: BEA

2.3. PHASE A: location of start of fire

The purpose of phase A was to define where the fire started and to describe the systems present in this area.

2.3.1. Detailed description of the outer surface of the wreckage

Two areas of distinctive damage were identified. These were:

- **Area of damage 1 (Figure 8):**

  This area was situated on the right side, just aft of the luggage hold.

  This area was characterised by the destruction of the aluminium alloy sheet making up the aeroplane's fuselage. It had taken a V-shape which is typical of damage which occurs after the release of intense heat. The closed end of the V probably corresponds to the area of maximum heat. At the open end of the V, the fuselage sheet was laid bare, its outer coating (layers of paint) having been destroyed.

- **Area of damage 2 (Figure 8):**

  The area is located level with the luggage hold, in the immediate vicinity of the rear bench. There is no particular equipment in this area.

  This area was characterised by the destruction of the sheet of the aeroplane's fuselage.

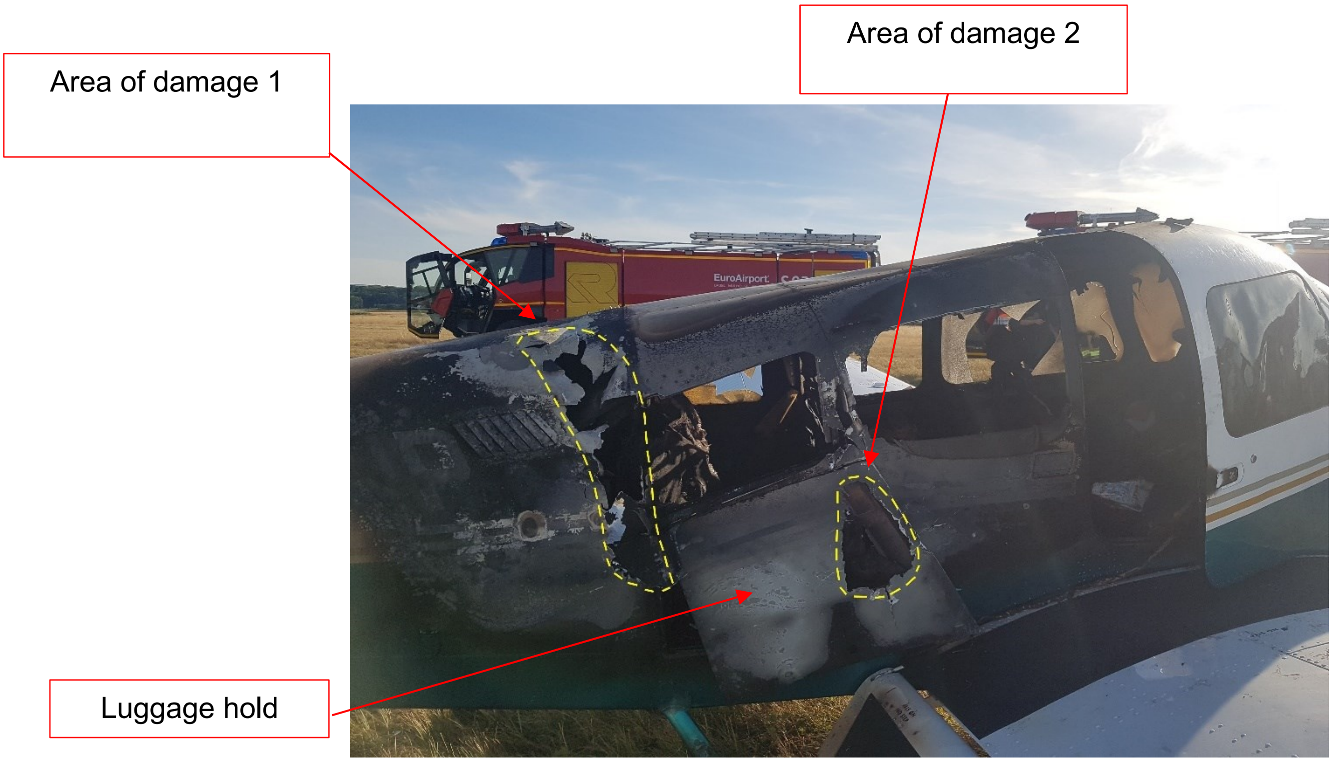


Figure 8: Right side of aeroplane
Photo taken day of accident

Furthermore, the external paint was damaged on the rear right side and part of the rear upper surface of the fuselage. The degree of damage varied according to the area of the airframe. This variation in damage reflects the spread of the hot smoke in the cockpit. The paint was more damaged at the rear, aft of the luggage hold and on the right side.

The extent of this area of damage is shown in the diagrams below (**Figure 11**).

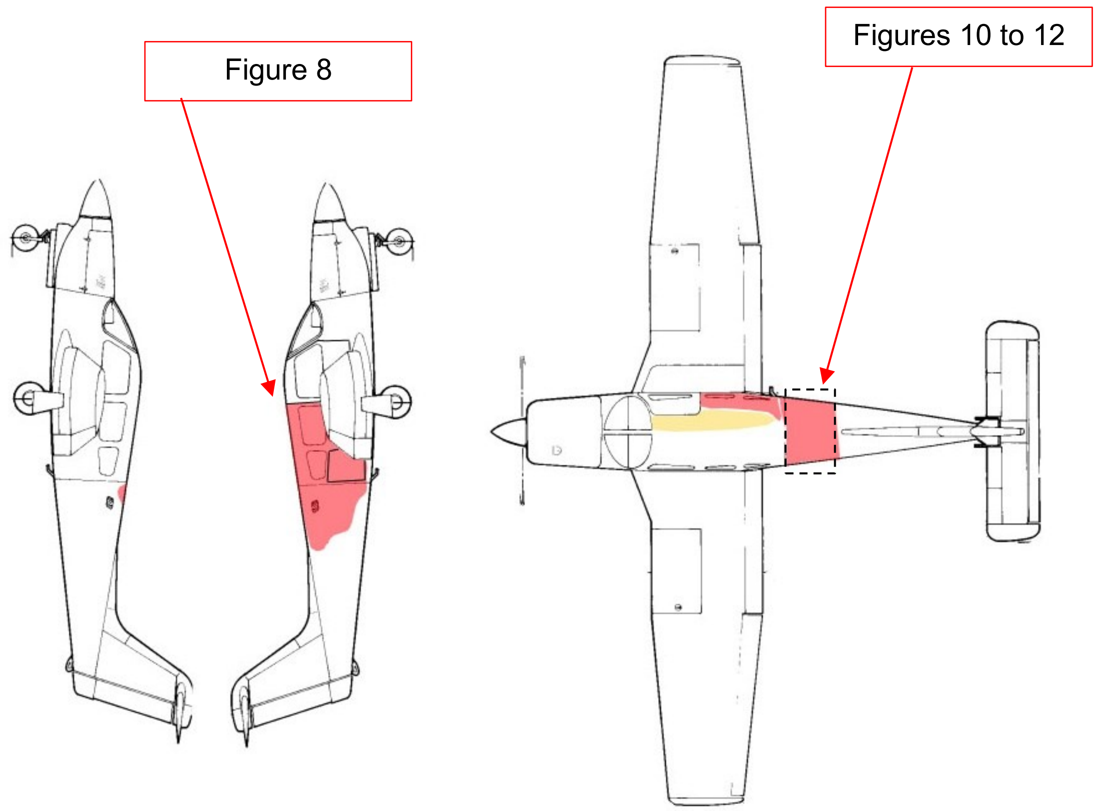


Figure 9: Identification of areas where the outer skin was destroyed or damaged
Source of original diagram: Piper Maintenance Manual

Key:

(red)	Area where the external paint on the sheet was damaged or destroyed
(orange)	Area where the external paint on the sheet was just covered by soot

Figure 10: Area where the external paint on the sheet was damaged or destroyed (upper part of the aeroplane)
Source: BEA

Figure 11: Area where the external paint on the sheet was damaged or destroyed (upper part of the aeroplane)
Source: BEA

Figure 12: Detail of condition of external paint
Source: BEA

The glass panels on the left side were broken (**Figure 13**). These ruptures had a curved profile. According to the expert consulted, this type of profile suggests that this damage was the consequence of a high temperature and not a mechanical action.

Figure 13: Glass panels on left side
Source: BEA

On the right side, the two rear glass panels had melted (**Figure 14**). The glass panel of the right door was complete but deformed.

Figure 14: Glass panels on right side
Source: BEA

The aeroplane's windshield was complete and not deformed (**Figure 15**).

Figure 15: Windshield
Source: BEA

2.3.2. Detailed description of the inner surface of the wreckage

The various components were in position in the cockpit. They were damaged to varying degrees.

The four seats showed substantial damage (**Figure 16**), in particular the destruction by combustion of most of the non-metal components. Only the front right seat cushion and backrest showed little damage. This can be explained by the seat having been folded forward by the rear passenger when he exited the aircraft. This observation indicates that the damage in the cockpit continued after the persons on board had exited the aircraft. The condition of the aeroplane, as described above, is thus not that on touchdown or landing.

Figure 16: Seats removed from cockpit at the BEA's premises
Source: BEA

The cockpit was characterized by substantial damage to the side skin panels, the overhead skin panels and the instrument panel. This damage was the result of contact with smoke (**Figure 17** to **Figure 19**), and reflected the movement of the smoke in the aircraft due to the circulation of air. This damage particularly consisted of the melting or deformation of the plastic panels.

On the floor, the carpet at the base of the seats still had its normal colour (**Figure 17**). This carpet had heat damage at the base of the rear right seat, next to the door (**Figure 17**).

The outer surface of the overhead skin panels was blackened but was not destroyed (**Figure 19**).

The inner surface of the tail was covered in soot (very black).

Figure 17: Cockpit after cutting the fuselage and removing the seats
Source: BEA

Figure 18: Cockpit after cutting the fuselage and removing the seats
Source: BEA

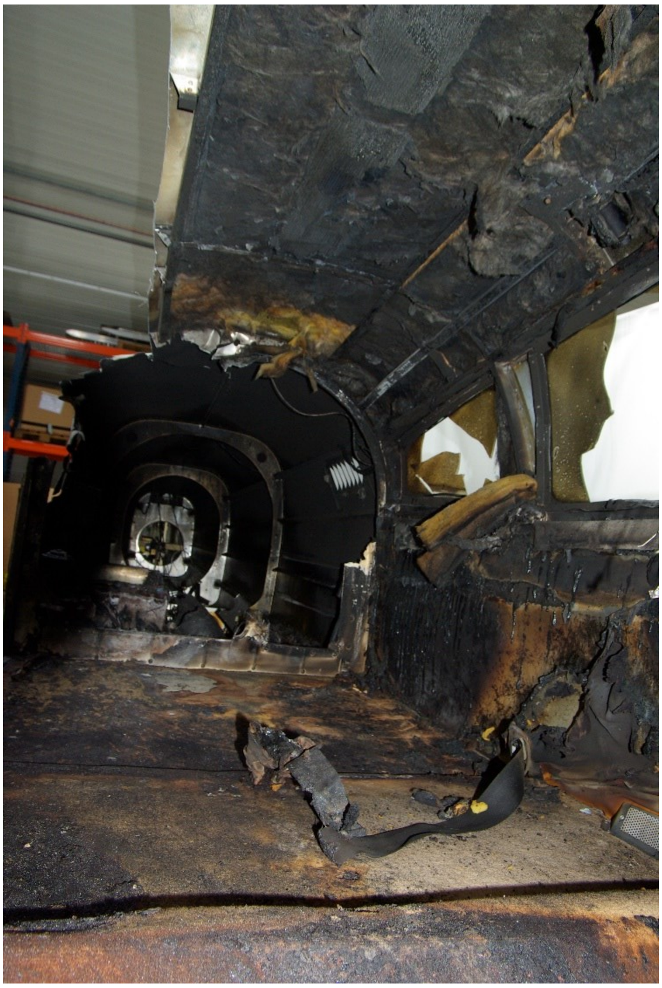


Figure 19: Cockpit after cutting the fuselage and removing the seats
Source: BEA

The plastic skin on the instrument panel had partially melted and was deformed. The instrument faces were covered with soot (**Figure 20**).

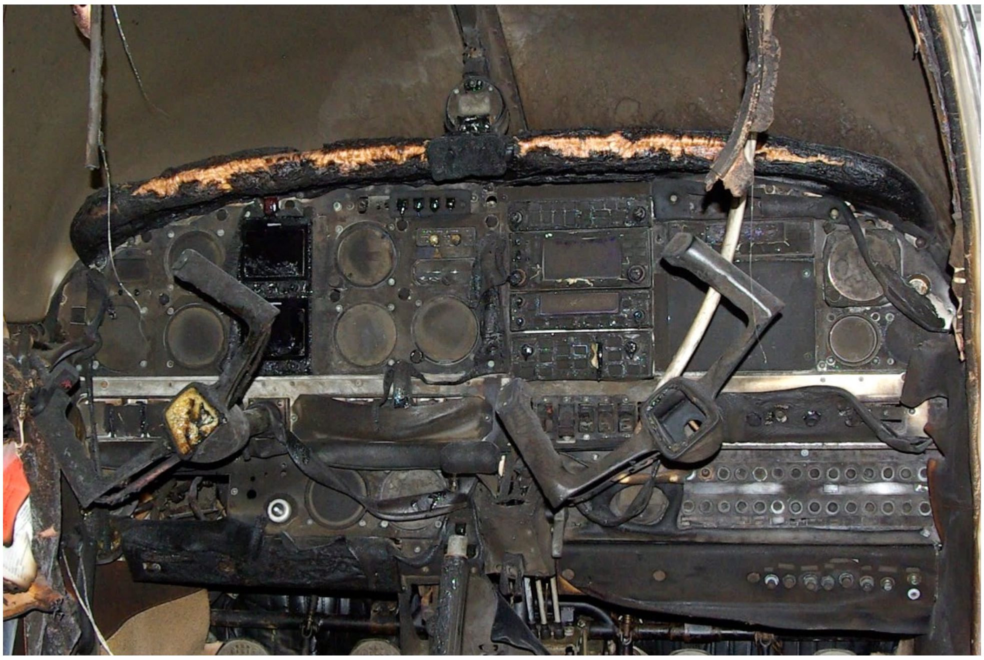


Figure 20: Instrument panel
Source: BEA

2.3.3. Detail of zone situated aft of luggage hold

2.3.3.1 - Composition of area in question

The description of the area situated aft of the luggage hold is based on observations made on an identical aeroplane registered HB-PPB. This aircraft and damaged HB-PNP were maintained by the same workshop.

The luggage hold is accessed through a small door on the right side of the aircraft (**Figure 21**).

Figure 21: Access door to luggage hold
Source: BEA

This luggage hold has two windows situated on each side. It is not closed off horizontally by a cover. The hold can therefore be accessed from the rear seats.

The rear of the hold is closed off by a composite partition (**Figure 22**).

On the left side of this partition (pilot side looking forward), there is a screen associated with the ventilation system.

On the right side, there is a door to access equipment situated behind the luggage hold: this is the area with the most damage on HB-PNP.

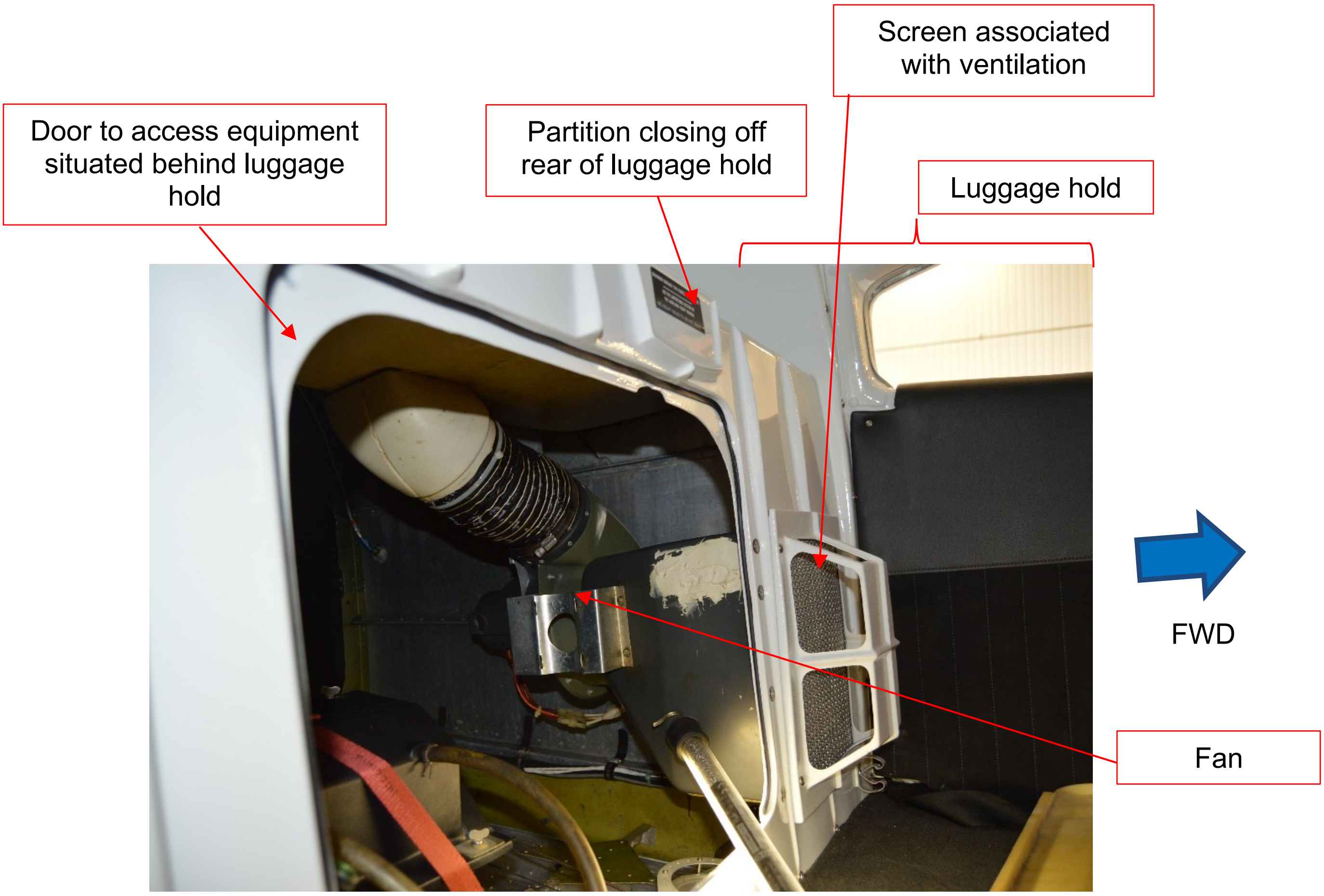
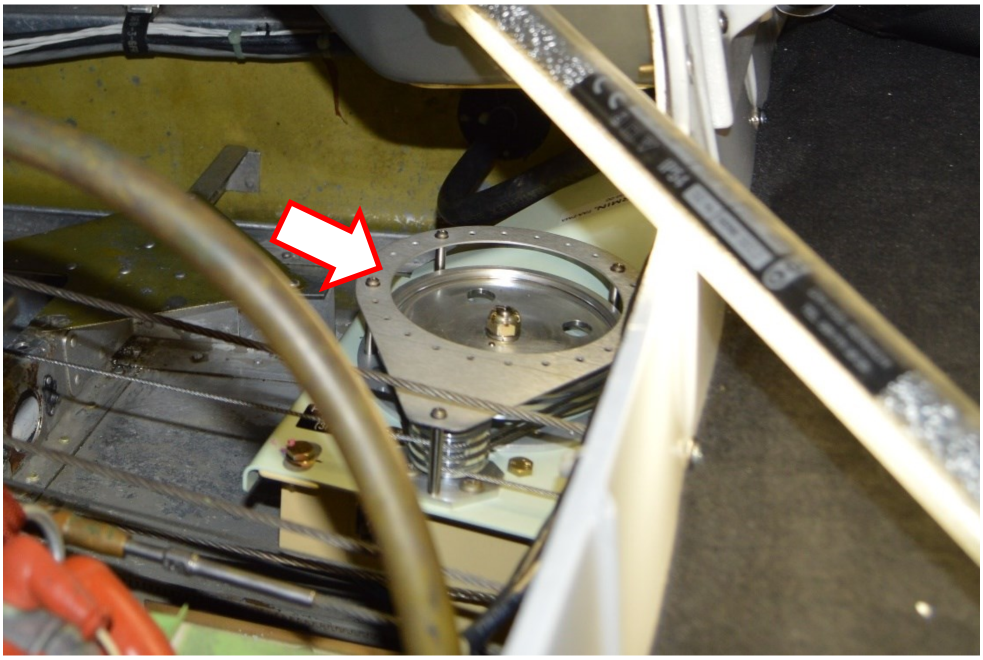


Figure 22: Inside of luggage hold (view from right to left) and compartment situated behind luggage hold
Source: BEA

The following equipment is located behind the luggage hold.

- On the left side, at approximately mid-height, there is the electrically-powered fan (**Figure 22**). This fan is attached to a flexible metal hose which is itself attached to composite ducts to evacuate air to the exterior via two lateral outlets (one on each side of the airframe).

- On the floor, on the left side, there is the autopilot servocontrol (**Figure 23**). This servocontrol is also electrically powered, the electric motor being located below the servocontrol.



FWD

Figure 23: Servocontrol on left side of floor
Source: BEA

- On the right side, the hydraulic pump is attached along the partition shared with the hold (**Figure 24**). This pump is electrically powered.

FWD

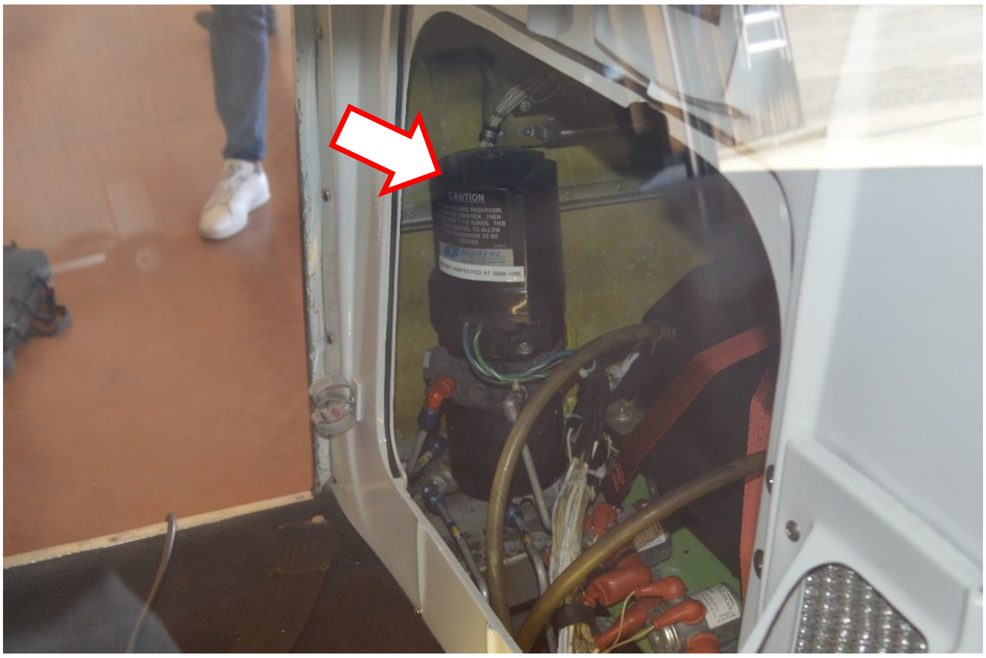


Figure 24: Hydraulic pump
Source: BEA

- Next to the hydraulic pump, there is a battery and several contactors attached to the floor (**Figure 25**).

  The battery is enclosed in a composite box. This box is fastened to the floor with fabric straps. The battery box is equipped with three vent hoses leading to the exterior of the aircraft: one hose connected to the base of the box and two hoses connected to the top part of the box.

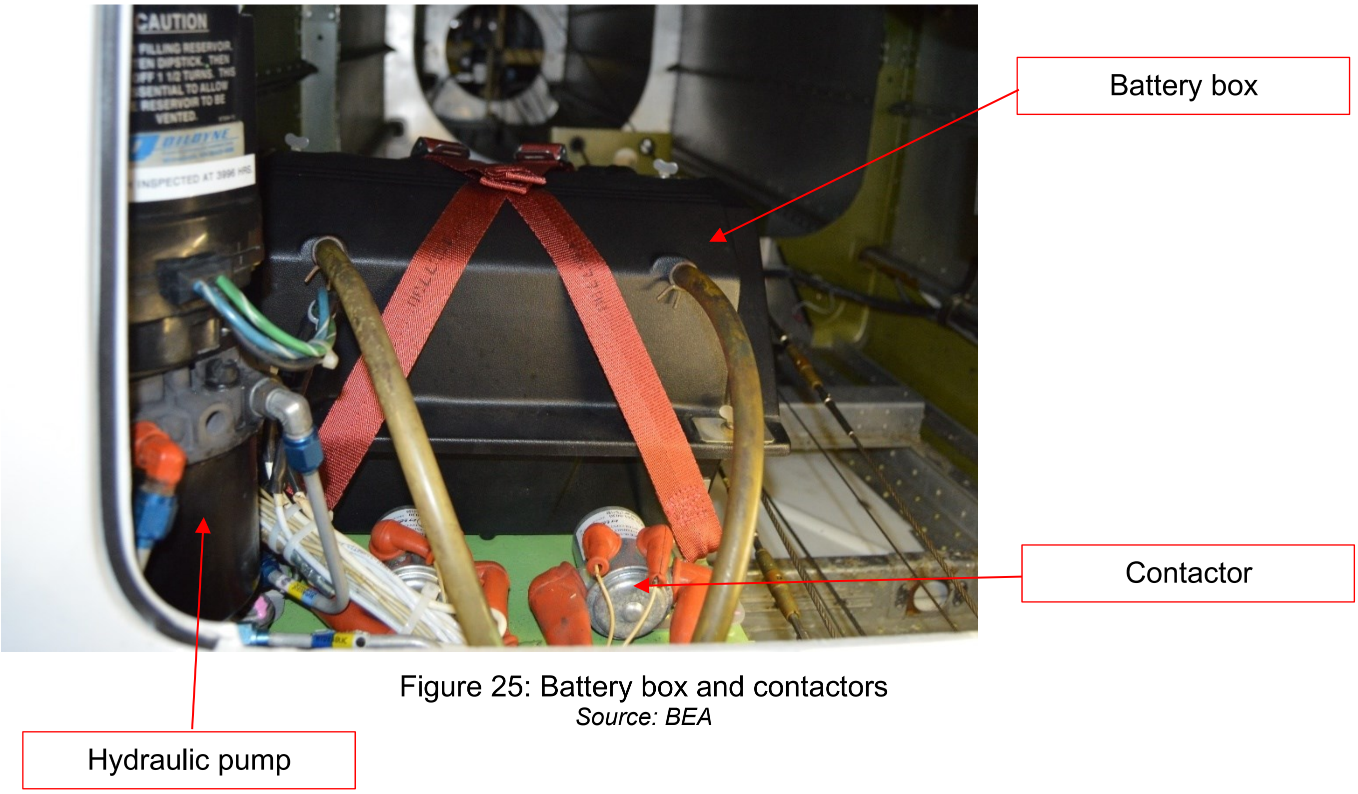


Figure 25: Battery box and contactors
Source: BEA

- On the right partition, just behind the hydraulic pump, there is an assembly comprising two relays and a variable resistor (**Figure 26**).

Figure 26: Electrical assembly situated on right partition
Source: BEA

- Several equipment items are positioned behind the battery (**Figure 38**). They are not shown on the intact aeroplane:
    - a 12 V – 24 V converter (manufactured by FLITE-TRONICS CO. INC / S/N: 875), situated above the hydraulic pump;
    - a contactor associated with the ground power connector;
    - a diode device installed on a metal cooler (device associated with the power system, paragraph 2.4.1). This device had a cover made of ABS[1];
    - a regulated power booster (manufactured by KCS ELECTRONICS, model: RB-125, S/N: 3524);
    - a circuit breaker situated behind the battery[2].

The area behind the luggage hold chiefly contains the electrical equipment of the general electrical power system supplying various equipment items of the aeroplane.

2.3.3.2 - Detailed description of area situated behind luggage hold on damaged aircraft

The partition closing off the rear part of the luggage hold was partially destroyed (**Figure 27**).

Figure 27: Rear limit of luggage hold
Source: BEA

[1] Acrylonitrile butadiene styrene, or ABS, is a thermoplastic and amorphous industrial polymer combining three different monomers: Acrylonitrile, Butadiene and Styrene.
Concorde Aircraft Batteries states that this ABS cover has the following characteristics:
- softening point: 103-128°C;
- flash-ignition temperature: 349°C;
- self-ignition temperature: 508°C.
These characteristics are taken from the SDS available at the following link:
https://www.acplasticsinc.com/media/SDS%20for%20ABS.pdf. This document also states that when exposed to an external source of ignition, ABS will burn, releasing intense heat and dense black smoke.

The INRS also provides a product sheet specifying that this material degrades from 276°C, with 92% degradation at a temperature of 450°C. Link to the INRS data sheet:
https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&cad=rja&uact=8&ved=2ahUKEwj65PGG5d-EAxXBdqQEHc8iD3c4ChAWegQIAhAB&url=https%3A%2F%2Fwww.inrs.fr%2Fdms%2Fplastiques%2FPolymerePlastiques%2FPLASTIQUES_polymere_4-25%2FPOLYMERE_ABS.pdf&usg=AOvVaw044DN1soQoEs5Kb4Y5Vj8n&opi=89978449
[2]: This equipment item does not have an identification plate.

On the left side, the fan was found lying on the floor. The flexible metal hose and the composite ducts were destroyed.

The servocontrol was still in position on the floor (**Figure 28**). It showed no substantial damage. Its electrical connections and the associated conductors were whole, no singularity was observed (**Figure 29**).

Figure 28: Servocontrol in position
Source: BEA

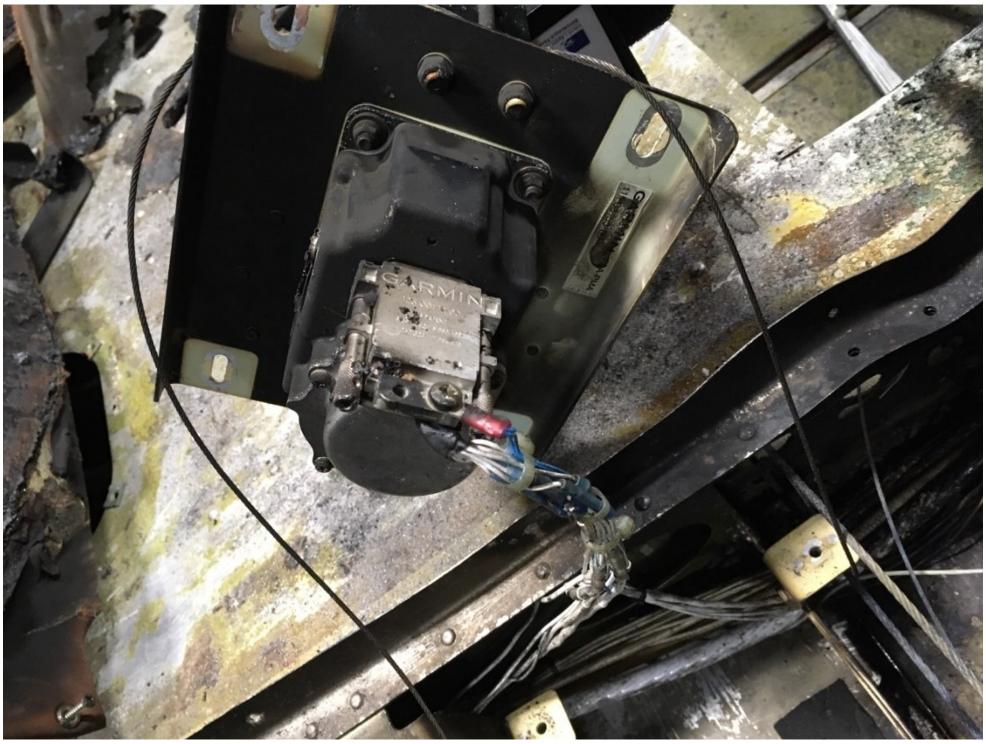


Figure 29: Electrical connections of servocontrol
Source: BEA

On the right side, the hydraulic pump was in position and nominally screwed to the floor (**Figure 30**). Three conductors were associated with the hydraulic pump. These conductors were continuous (an earthed conductor secured to the aeroplane's floor and two conductors attached to adjacent contactors). Their insulation was destroyed, very probably by the high temperature. Despite the insulation being destroyed, the core of these conductors remained flexible. These observations did not indicate an electrical problem on these components.

Figure 30: Hydraulic pump
Source: BEA

The two contactors, situated in front of the battery, were still in position (**Figure 31**) with their conductors still nominally attached and electrically continuous. The insulation of some of these conductors was destroyed, very probably by the high temperature. Despite the insulation being destroyed, the core of these conductors remained flexible. These observations did not indicate an electrical problem on these components.

It was observed that the outer surface of one of the relays was corroded, very probably due to the degradation of the adjacent battery.

Figure 31: Contactors situated in front of battery
Source: BEA

The two components situated on the right partition had the following damage:

- The electrical assembly, comprised of the two relays and the variable resistor, was in position. The associated conductors were in place (**Figure 32**), their insulation had been destroyed by the high temperature. Despite the insulation being destroyed, the core of these conductors remained flexible. These observations did not indicate an electrical problem on these components[3].

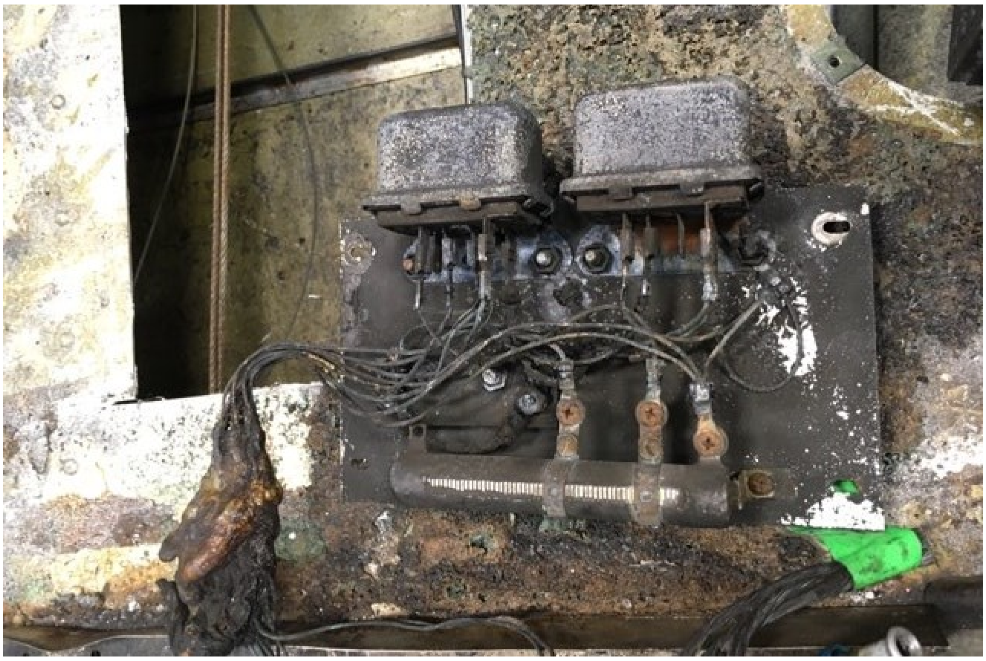


Figure 32: Electrical assembly removed from right partition
Source: BEA

[3] This property was shared by the expert consulted by the BEA.

- The 12 V – 24 V converter was in position. No conductor was associated with this assembly (**Figure 33**). These conductors were found in the area around the battery. The insulation of these conductors had been destroyed by the high temperature. The core of these conductors remained flexible.

Figure 33: 12 V - 24 V converter removed from right partition
Source: BEA

A smell of burning was perceived when this converter was opened. The components were damaged by exposure to a source of high heat. The casings did not show any singular area which stood out from the rest of the equipment[4].

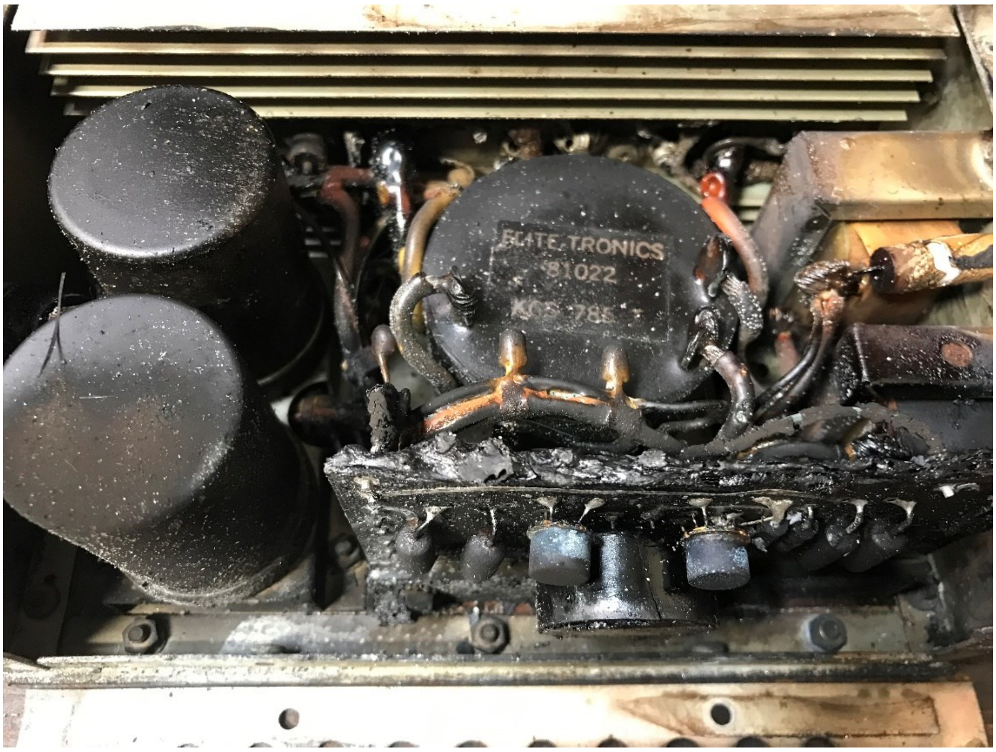


Figure 34: Components of 12 V - 24 V converter
Source: BEA

[4] When there is overheating due to the degradation of a component inside the unit, signs of local overheating can generally be observed on the inner surface of the unit in line with the component (discolouration, soot, melting, etc.).

Figure 35: Inner surface of 12 V - 24 V converter casing
Source: BEA

- The circuit breaker (different from the circuit breakers equipping the rest of the electrical system) was in position on the aircraft floor (**Figure 36**). The equipment had damage on its upper surface due to its exposure to an excessive temperature (**Figure 37**). This circuit breaker was not electrically connected to the aircraft.

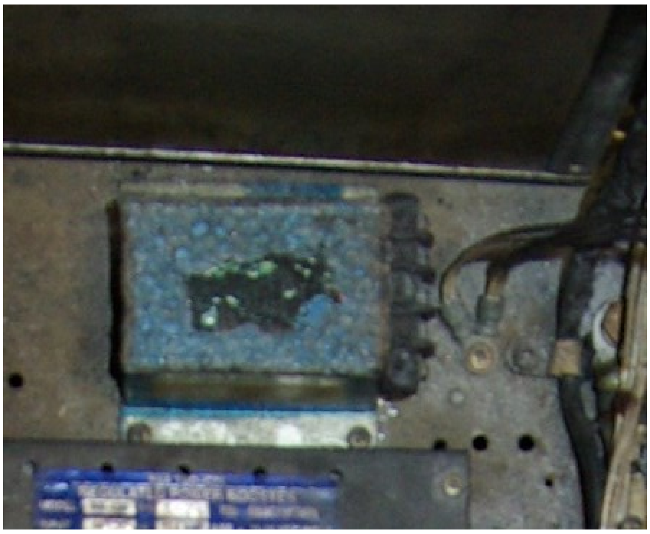


Figure 36: Circuit breaker in position on the cockpit floor and not electrically connected
Source: BEA

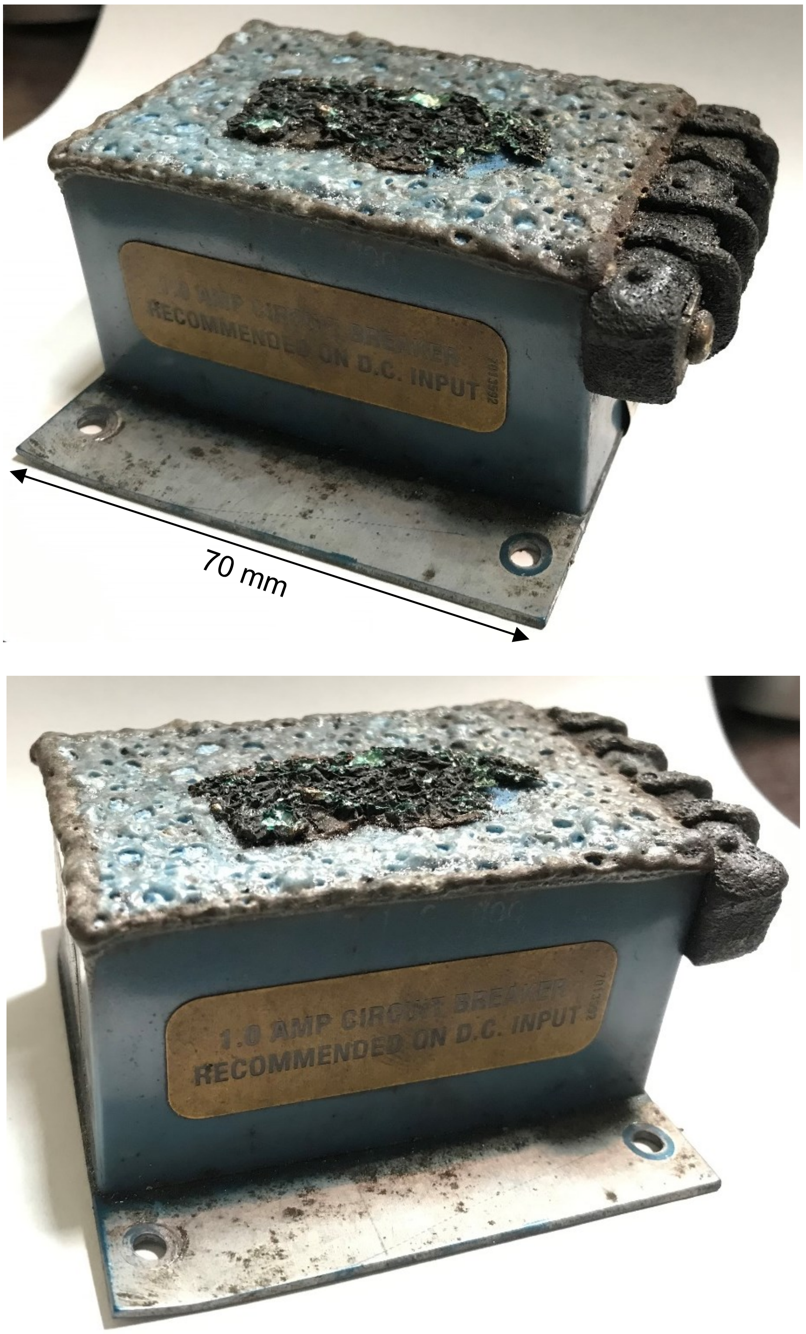


Figure 37: *Circuit breaker*
Source: BEA

Various components were present behind the battery. They are described in detail in the following chapters.

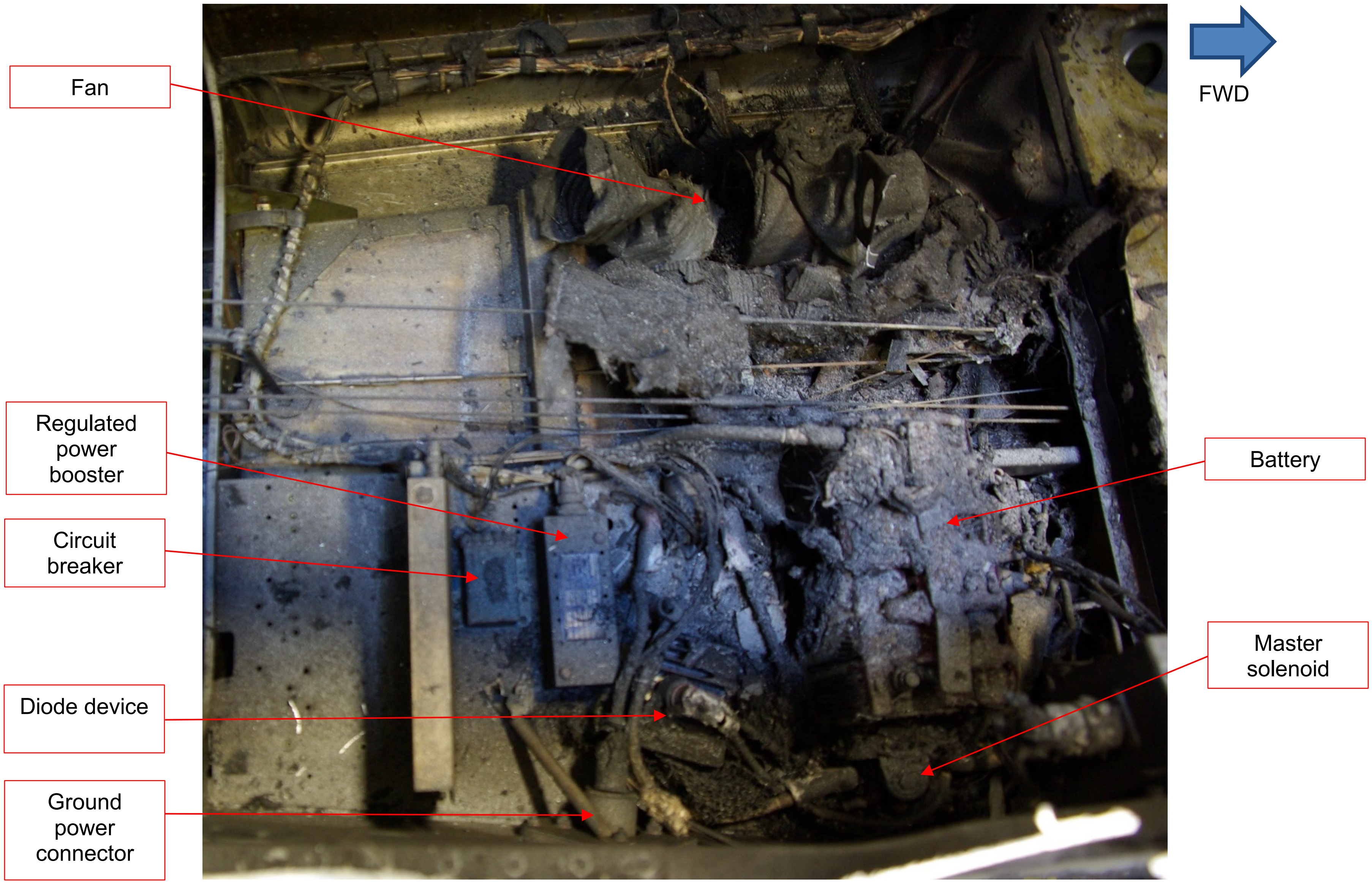


Figure 38: Area around battery
Source: BEA

2.3.4. Conclusion of phase A

The fire started in the compartment behind the luggage hold. This zone includes the ventilation system, the hydraulic pump for the landing gear, and numerous components making up the general electrical system supplying the aeroplane's various equipment items.

The damage observed on the ventilation system and the hydraulic pump was not prior damage on this equipment. The work was therefore concentrated on the aeroplane's electrical power system.

2.4. PHASE B: aircraft's general electrical power supply system

The purpose of phase B was to assess the aircraft's general electrical power supply system, from the power sources to the various electrically-powered components.

2.4.1. Composition of system specified by Piper

The damaged aeroplane had a 12 VDC electrical power system.

The following diagram (**Figure 39**) shows the general electrical power system supplying the various components, specified by Piper for the aeroplane in question.

On this diagram:

- the parts of the system located aft of the luggage hold are highlighted in blue;
- the parts of the system located in the engine compartment are highlighted in green;
- the parts of the system located in line with the instrument panel are highlighted in orange.

Note: It was observed that in this system, there was no circuit breaker between the alternator and the battery.

A large number of aircraft (e.g. Daher Socata TB20, Cirrus SR22, Cessna 206, Cessna 172, Beechcraft 58, DR400-180, etc.) have an electrical power system equipped with a circuit breaker between the alternator and the aircraft power system[5]. On the electrical power systems consulted, this circuit breaker had a tripping threshold between 60 and 100 amps.

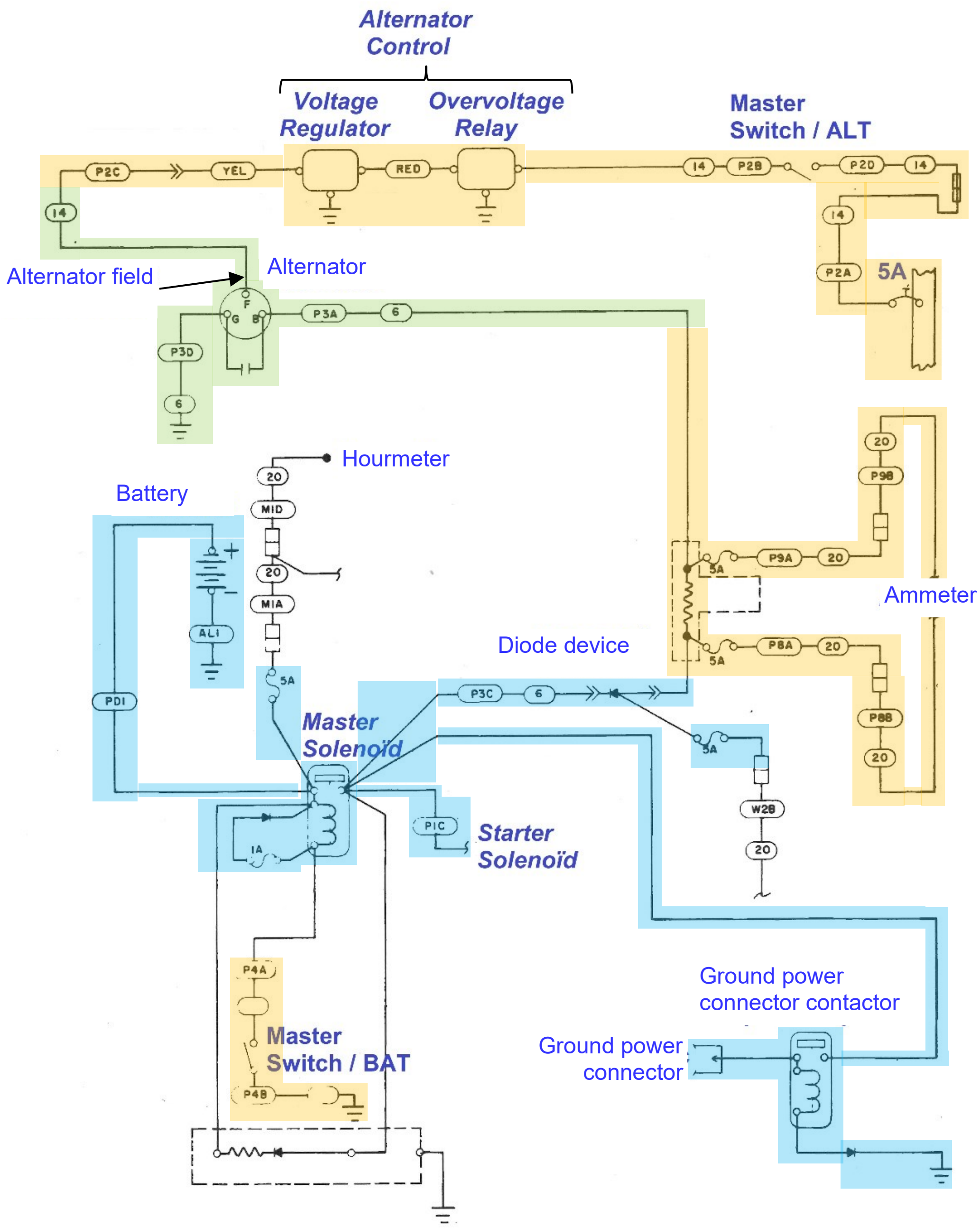
According to the manufacturer, Piper, consulted via the NTSB, the use of such a circuit breaker was not required during the certification of the aircraft type in question.

[5] This type of circuit breaker is triggered by two conditions: an abnormally high amperage and/or an abnormally high temperature.

The BEA carried out examinations on a circuit breaker of this type in 2019. This circuit breaker had a 60 A rating and equipped a Cessna 206G. The circuit breaker specifications were the following:

- The circuit breaker had to continually withstand 60 A at an ambient temperature of 25° C +/-3° C.
- When a 120 A charge is applied, the circuit breaker must trip in less than 30 s.

Furthermore, the BEA showed that this type of circuit breaker tripped at a temperature of around 160°C.



	Parts of the system located aft of the luggage hold
	Parts of the system located in the engine compartment
	Parts of the system located in line with the instrument panel

Figure 39: Aircraft's electrical power supply system

Source: PA-28RT-201 / 201 T Maintenance Manual, published 17 August 1982, Figure 91-10. Alternator System/External Power, PA-28RT-201 SN's: 28R-8118006 and up, PA-28RT-201T SN's: 28R-8131006 and up

2.4.2. HB-PNP's electrical power supply system

The electrical power supply system identified on HB-PNP corresponded to the system specified by the manufacturer.

The exact designation of the components of the damaged aircraft's system is specified in the following tables: the components are those specified for the aeroplane in question.

Components	Name/P/N/S/N
Alternator	• Manufacturer: Hartzell Engine Tech • P/N: ALX-9525BR • OEM P/N: ALX-9525B 12V/70A • S/N: H-R052153
Battery	• Manufacturer: Concorde Aircraft Batteries • Model: RG-35AXC • S/N: 40764283 • Volts: 12 V • Rated Capacity C1 = 1 h (quantity of electricity in Ah that the battery is capable of providing in 1 h): 33 A • Cold Cranking Amps (capacity of a battery to start up an engine in cold weather): 440 A • Polypropylene battery housing with a melting point of 160°C and a flash ignition temperature of 329°C  Figure 40: Concorde Aircraft Batteries, battery model RG-35AXC *Source: https://www.concordebattery.com/search-by-your-aircraft/battery_detail/159* *Note: Installation of this type of battery on a Piper. PA-28RT-201T complies with the specifications of STC SA01147WI (https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&cad=rja&uact=8&ved=2ahUKEwieidmLxqD6AhULKhoKHaoMAwAQFnoECA0QAQ&url=https%3A%2F%2Frgl.faa.gov%2FRegulatory_and_Guidance_Library%2FrgSTC.nsf%2F0%2F524C9D62448A72CE86257A79004D6EB8%3FOpenDocument&usg=AOvVaw0TVVD0czyFSQMiU15zUz9S). Concorde Aircraft Batteries[6] informed the BEA that STC SA01147WI had been embodied on around 3,000 aircraft and that no in-service difficulty or anomaly had been reported.*
Alternator Control (Voltage Regulator + Overvoltage Relay)	• Manufacturer: Lamar Technology • P/N: B-00371-1 • S/N: 80U05309 • 14.0 V 5 A • O/V CUT OFF: 16.0 V • FAA-PMA • Date of Mfg: FEB-2020 *Note: According to the information provided by Piper, P/N B-00371-1 replaces the initially specified P/N B-00331-2 which is no longer available.* *Note: This equipment was installed in July 2020 during the work to modify the aeroplane's avionics.*

[6] *The exchanges with Concorde Aircraft Batteries were carried out via the NTSB.*

<table>
<tr><th>Components</th><th>Name/P/N/S/N</th></tr>
<tr><td>Master solenoid</td><td>

Manufacturer: Eaton Cutler Hammer
TYPE II COIL 12 VDC
No. 6041H105A
The master solenoid is positioned on the right side of the battery box.

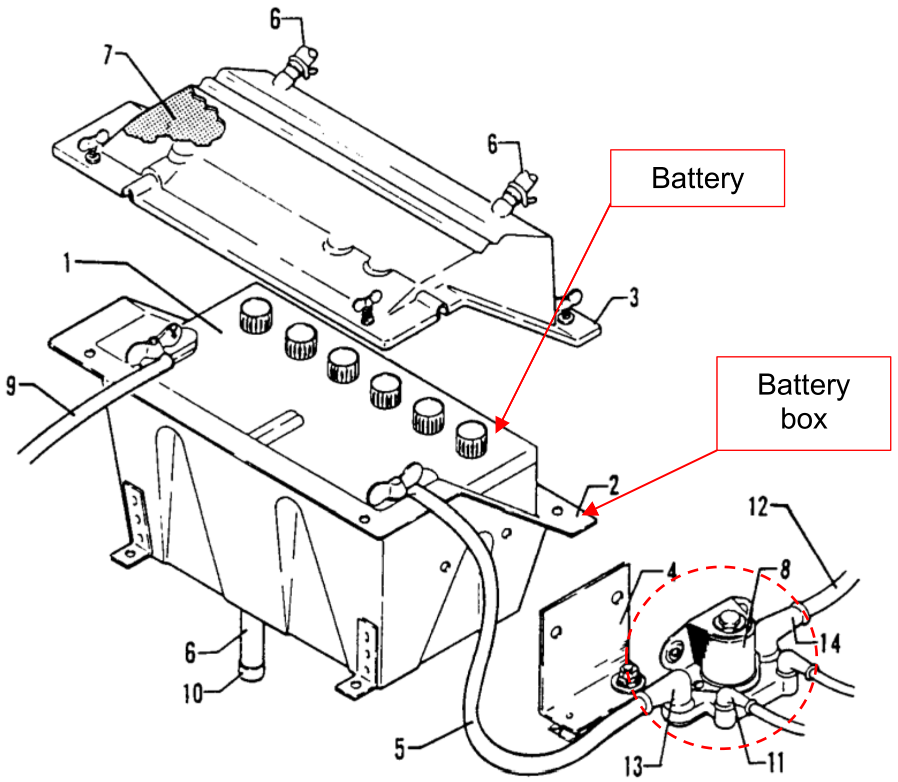
<u>Note:</u> Based on the information provided by Piper, the position of the master solenoid complies with the specifications (Parts Catalog, figure 70).


Figure 41: Diagram of battery box and master solenoid
Source: Piper
</td></tr>
<tr><td>Diode device between alternator and master solenoid</td><td>

Model: MR1215SLR

<u>Note:</u> Based on the information provided by Piper, this component forms part of an assembly P/N 79412-002, complying with the manufacturer’s specifications.
</td></tr>
</table>

2.4.3. STC SA3531NM

2.4.3.1 – Composition of STC

HB-PNP is equipped as per STC SA3531NM[7].
This STC is an installation specified by BOGERT AVIATION INC. This installation consists of replacing the cables between the battery and starter as illustrated in **Figure 42** below.

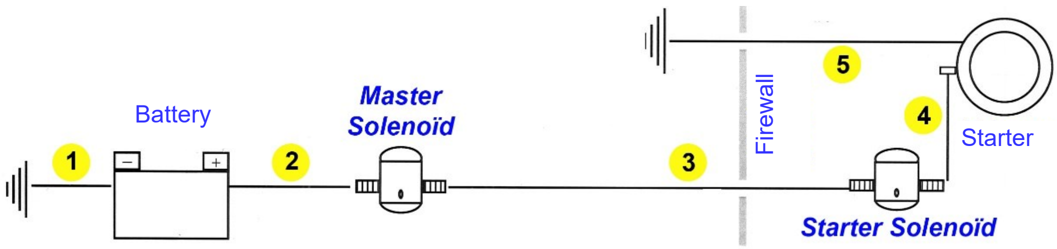


Figure 42: STC SA 3531NM

	Length
1	13'' (+2'' -0'') i.e. 330.2 mm (+50.8 mm -0 mm)
2	12'' (+2'' -0'') i.e. 304.8 mm (+50.8 mm -0 mm)
3	165.5'' (+2'' -0'') i.e. 4203.7 mm (+50.8 mm -0 mm)
4	35.5'' (+2'' -0'') i.e. 901.7 mm (+50.8 mm -0 mm)
5	50'' (+2'' -0'') i.e. 1270 mm (+50.8 mm -0 mm)

Each cable is manufactured from a copper cable P/N M22759\16-1-9 or another cable P/N M22759\16-2-9, M22759\34-1-9 or M81044\9-1-9.

Based on the information provided by Concorde Aircraft Batteries, this type of cable is associated with EFTE insulating material which has a melting temperature of between 254°C and 279°C.

Note: Two other STCs, listed below, can be associated with the STC above.
- *STC SA4008NM[8]: Battery box modification (**Figure 44**);*
- *STC SA01582SE[9]: New Bogert battery box (**Figure 43**).*

*STC SA4008NM specifies that it is compatible with STC SA3531NM. In this case, the adapters for the Concorde batteries (08M-1-1-6 FAA/PMA) are to be used (**Figure 45**).*

[7] https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&cad=rja&uact=8&ved=2ahUKEwjX8KnsyaD6AhURrxoKHbEfBUkQFnoECA8QAQ&url=https%3A%2F%2Frgl.faa.gov%2FRegulatory_and_Guidance_Library%2Frgstc.nsf%2F0%2FCAD966288C4ABC4E86257CEE0068782F%3FOpenDocument&usg=AOvVaw0PbUqiToRfbfoGUABM935x

[8] *https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&cad=rja&uact=8&ved=2ahUKEwjV94mzyqD6AhUCrxoKHQLaBswQFnoECAcQAQ&url=https%3A%2F%2Frgl.faa.gov%2FRegulatory_and_Guidance_Library%2Frgstc.nsf%2F0%2F03A248773F6458D4862570A7004B6C4B%3FOpenDocument&usg=AOvVaw1a0EbeSwMY_PsQ5MkS8nA4*

[9] *https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&cad=rja&uact=8&ved=2ahUKEwiq2seoy6D6AhUS_lUKHWofBbgQFnoECBIQAQ&url=https%3A%2F%2Frgl.faa.gov%2FRegulatory_and_Guidance_Library%2Frgstc.nsf%2F0%2F1390E2840759124386257ED5005C8EF3%3FOpenDocument&usg=AOvVaw0wF1A9q_NhdS4SG10zlMXb*

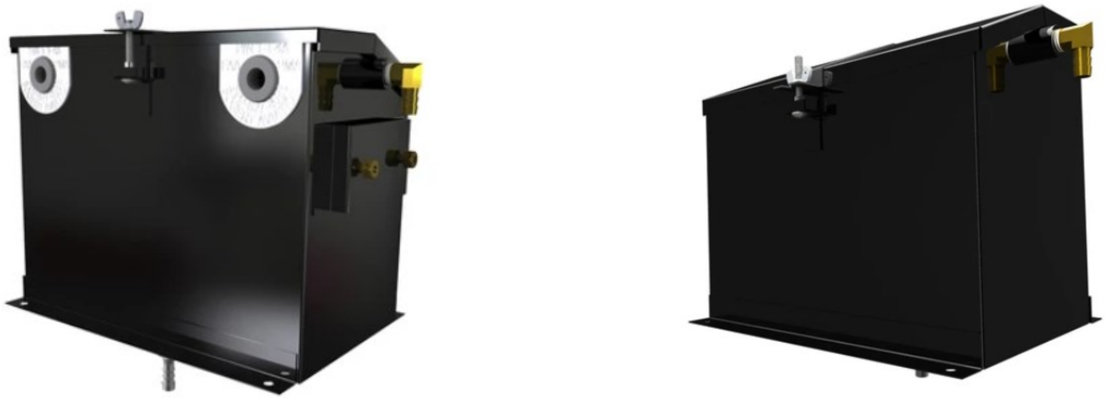


Figure 43: New Bogert battery box _ STC SA01582SE

Figure 44: Battery box modification _ STC SA4008NM

Figure 45: Adapters for Concorde batteries

Note:

- *The workshop which carried out the maintenance work on the aeroplane informed the BEA that the terminals of the Concorde battery (STC SA01147WI) were 8 mm lower than the terminals of the GILL battery normally specified for the aeroplane in question (**Figure 46**).*
- *This difference in height is eliminated by using the adapters for the Concorde batteries (08M-1-1-6 FAA/PMA) (**Figure 47**).*

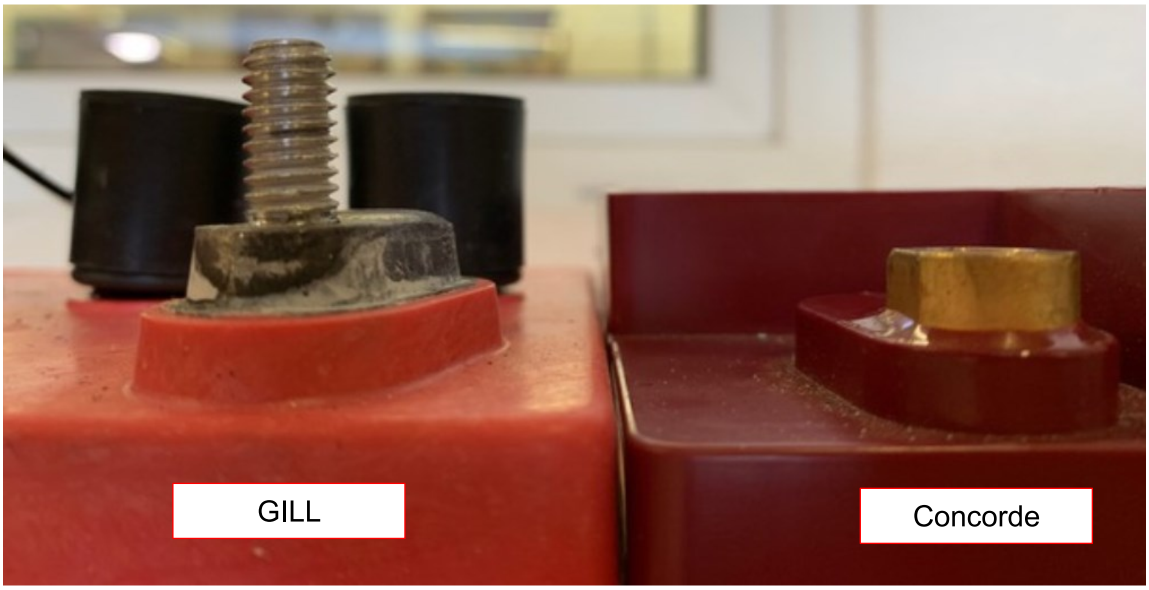


Figure 46: Comparison between a GILL battery terminal and a Concorde battery terminal
Source: Flugschule Basel AG

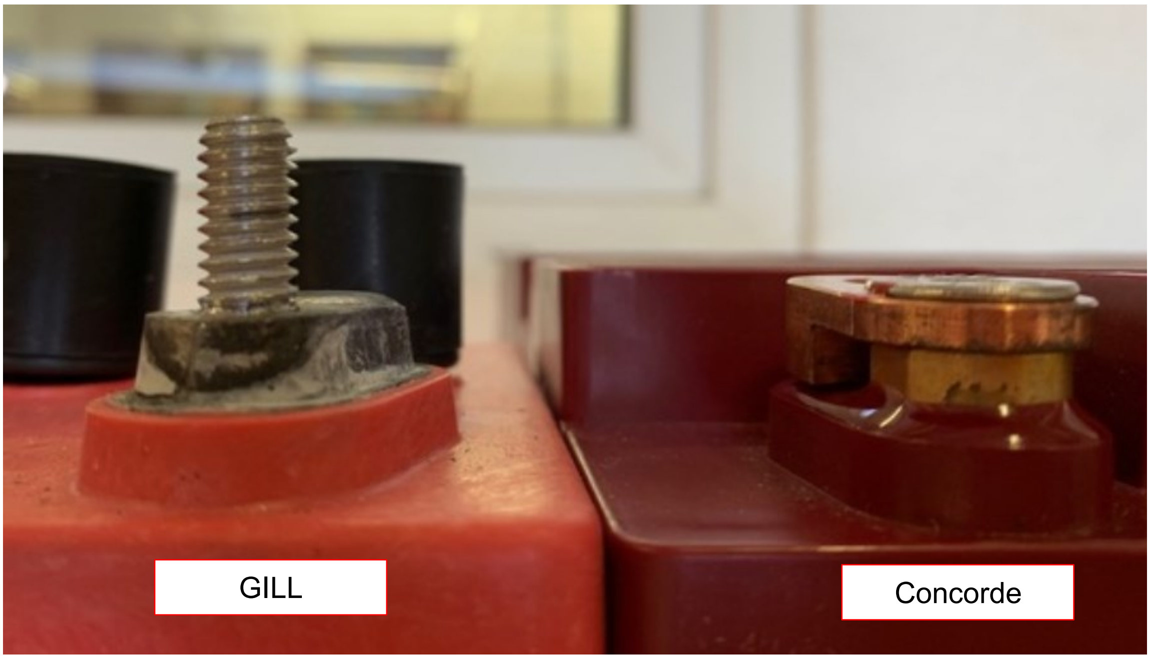


Figure 47: Comparison between a GILL battery terminal and
a Concorde battery terminal with adapter
Source: Flugschule Basel AG

2.4.3.2 – Embodiment of STC SA 3531NM on HB-PNP

The STC SA 3531NM work was carried out on 4 and 7 November 2019.

The cables equipping the damaged aircraft all had the P/N M22759\16-1, effectively corresponding to one of the P/Ns specified in the STC.

All the cable lengths complied with the specification. No damage was found on these cables.

On the damaged aeroplane (HB-PNP), STC SA4008NM/STC SA01582SE and the adapters for the Concorde batteries (08M-1-1-6 FAA/PMA) were not installed.

2.4.4. Battery and associated cables

2.4.4.1 – Technology and history of battery

HB-PNP was equipped with a RG-35AXC battery manufactured by Concorde Aircraft Batteries.

The RG-35AXC is a valve-regulated lead battery, also known as an AGM (Absorbed Glass Mat) battery. In this construction, there is no free liquid electrolyte. The electrolyte is contained in glass-fibre mats situated between the electrodes.

The date of manufacture of the battery was 12 November 2015. The battery was installed on the damaged aeroplane on 18 October 2017.

Note: The battery manufacturer does not specify any time limit between the data of manufacture and the date at which it is first used. The storage and installation conditions are specified in the Component Maintenance Manual for RG Series Main Aircraft Batteries, No 5-0171, published by Concorde Aircraft Batteries[10].

The battery's capacity was routinely checked during each annual inspection. The workshop which carried out the maintenance work on the aeroplane specified that this check was carried out according to the recommendations of the Owner/Operator Manual No 5-0324, published by the battery manufacturer[11].

The battery manufacturer specified that the battery test procedure was set out in document No 5-0171 and not in document No 5-0324. However, the latter refers to the former.

The battery had been recharged the day before the accident. It had been removed from the aeroplane to do this.

The workshop which carried out the maintenance work on the aeroplane told the BEA that the tightening torque applied to each battery terminal (attachment of conductors to positive and negative terminals) was 70 in.lbs. The battery manufacturer confirmed that this torque value was nominal.

*Note: Concorde Aircraft Batteries examined the consequences of excessive tightening of the terminal screws. If the overtorque was "modest", this would lead to cracks in the non-metal part around the terminal. This type of damage can be easily seen during an inspection of the exterior. However, if the overtorque was "extreme", this would lead not only to cracks in the non-metal part around the terminal, but to the rupture of the underlying seal (see diagram in **Figure 56**), which could lead to an electrolyte leak.*

2.4.4.2 – Outer appearance of HB-PNP battery

The battery box[12] (**Figure 41**) was largely destroyed. Only the base of the box had its habitual shape. The hose was still connected to the base of the box. Residue from the other two hoses was found in the bottom part, the top part of these hoses had very probably melted.

The straps holding the battery box in position had been destroyed. Only strap fragments in the metal buckles on the aircraft floor were identified.

[10] https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&ved=2ahUKEwjtpb7-_Zz0AhVGzYUKHS2IAwsQFnoECAcQAQ&url=https%3A%2F%2Fbatterymanagement.concordebattery.com%2FBatteryDocs%2F5-0171.pdf&usg=AOvVaw2I-NsgVKJZ6uTns3wOp7MJ

[11] https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&ved=2ahUKEwis6ZKtgp30AhVBExoKHS9cDmUQFnoECAcQAQ&url=https%3A%2F%2Fbatterymanagement.concordebattery.com%2FBatteryDocs%2F5-0324-rg-manual.pdf&usg=AOvVaw3rkbnKr4a8ja6GIQTLJIu0

[12] Based on the information provided by Concorde Aircraft Batteries, this battery box was made of Kydex 100, a plastic material. This material has a flash point of 390°C.

The appearance of the upper part and lower part of the battery was not homogeneous (**Figure 48** and **Figure 49**):

- the upper part of the battery box had melted, with material running from top to bottom, locally revealing internal components (without identified damage). The battery caps could still be identified;
- the upper part of the box, made of plastic, could still be identified with its initial grey colour. No swelling was observed in the bottom two thirds of the battery.

In the lower part of the battery, a contrast was observed between the sector on the negative terminal side and the sector on the positive terminal side (**Figure 50** and **Figure 51**). There was more damage on the positive terminal side.

Figure 48: Battery
Source: BEA

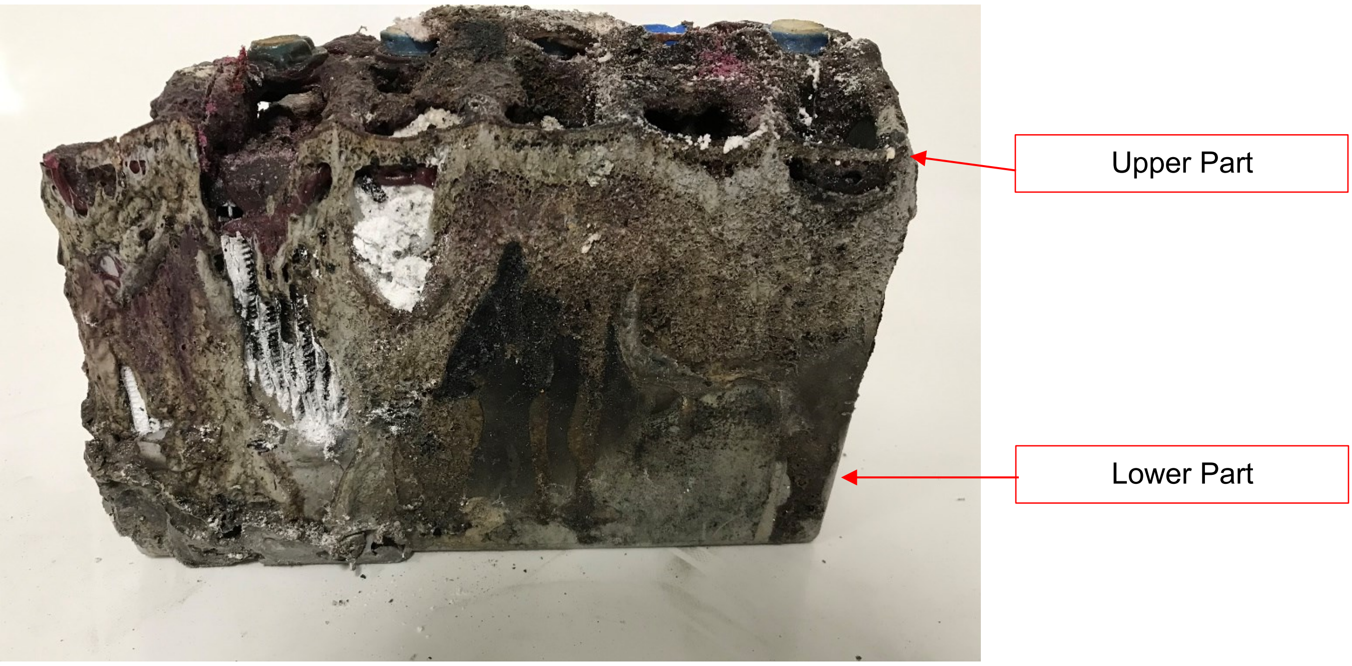


Figure 49: Lateral face of battery
Source: BEA

Figure 50: Lower part of battery, positive terminal side
Source: BEA

Figure 51: Lower part of battery, negative terminal side
Source: BEA

The negative terminal of the battery was still in position. The associated conductor was attached to this terminal (**Figure 52**). It was not possible to check the torque value. Due to the temperatures reached during the accident, this torque value would not have been usable and representative.

Figure 52: Negative terminal of battery
Source: BEA

The positive terminal of the battery was partially destroyed (**Figure 53**). Signs of melting were observed level with this terminal. The associated conductor was no longer connected. Part of the terminal was still integral with the end of the conductor and had therefore separated from the battery (**Figure 54**).

The O-ring ensuring the leaktightness of the battery was still in position and had not ruptured.

The insulation of the conductor normally connected to the positive terminal of the battery was damaged level with the lug. The electrical cable also seemed to have "swollen" and to be oxidized in this area (**Figure 54**).

When repositioning this conductor on the positive terminal of the battery, it was observed that substantial force was required and that the end of the conductor, on the battery side, was under high stress loadings (**Figure 55**). The stress loadings created can very probably be explained by the rigidity and length of the conductor (large section, diameter varying between 9.1 and 9.9 mm) and by the shape which had to be imposed on it to ensure its connection with the battery terminal.

Figure 53: Positive terminal of battery
Source: BEA

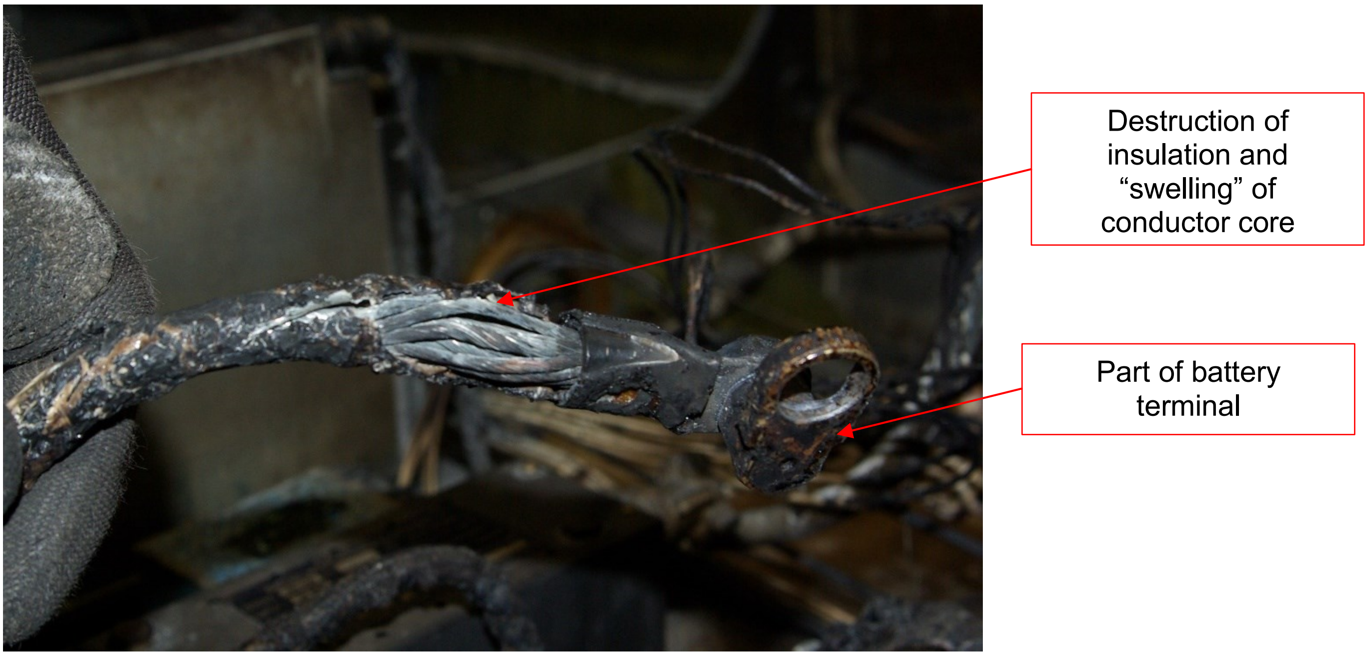


Figure 54: Conductor initially associated with positive terminal of battery
Source: BEA

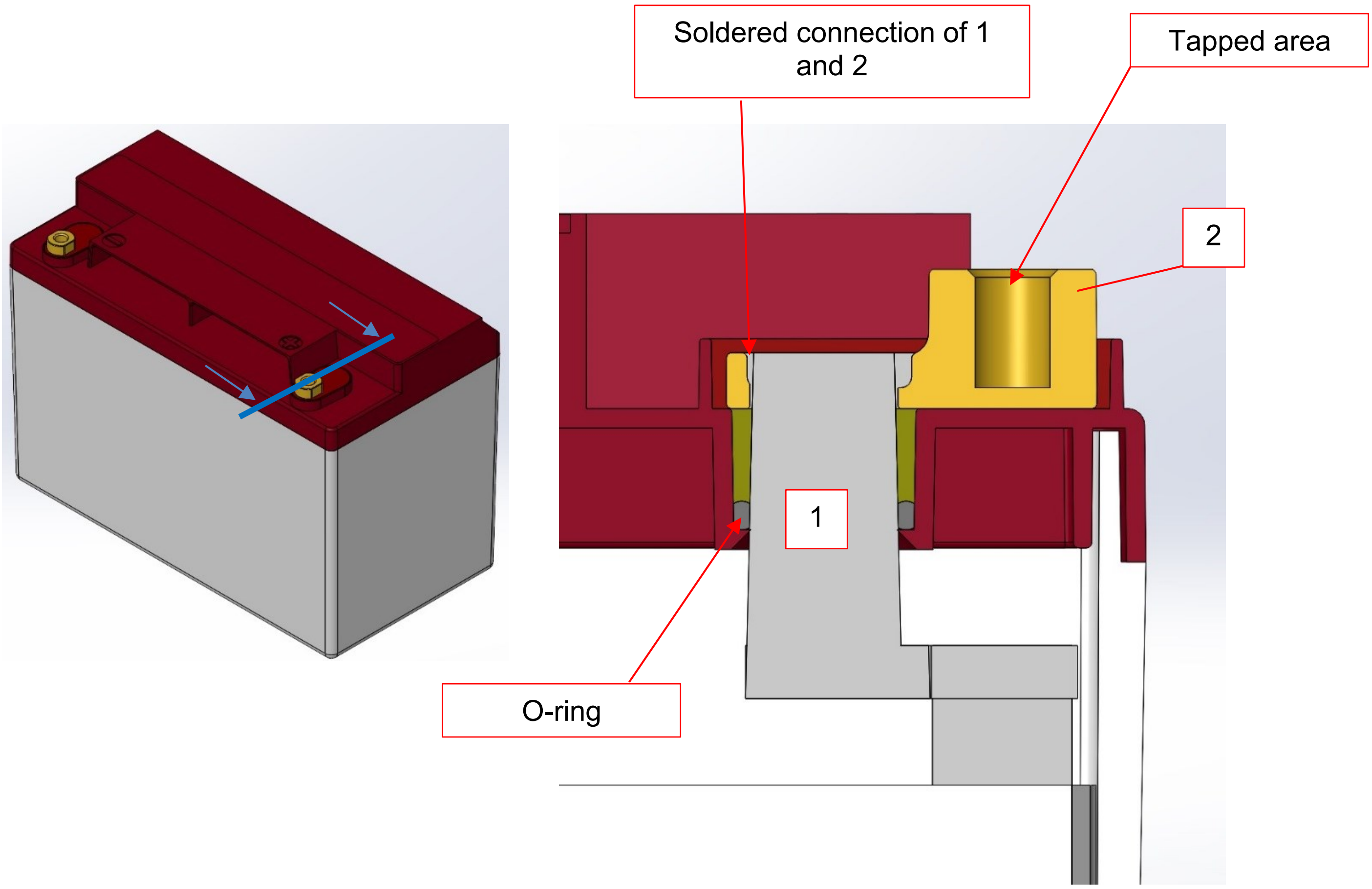
Figure 55: Repositioning of conductor on positive terminal of battery
Source: BEA

2.4.4.3 – Detailed description of battery terminals

2.4.4.3 – Composition of battery terminals

The battery has two terminals, a positive terminal and a negative terminal. The composition of each terminal is identical.

The diagrams below (**Figure 56**) show this composition, with an arbitrary parts list to facilitate understanding of the document.



Key	
1	Lead terminal connected to parts inside battery
2	Copper alloy component to which the conductor is attached by means of an attaching screw

Figure 56: Composition of battery terminals
Source: Concorde Aircraft Batteries

The parts designated 1 and 2 are connected by a solder. Energy-dispersive X-ray spectroscopy (EDS) coupled with scanning electron microscopy (SEM) analyses found a solder composed of (see following paragraph):

- Lead: base metal
- Tin: around 2%.

<u>Note:</u> According to the manufacturer Concorde Aircraft Batteries, the melting temperature of the soldered area is between 183 and 191°C.

2.4.4.3.2 – Examination of terminals

The positive and negative terminals of the battery were removed for a detailed examination in the BEA laboratory (**Figure 57** and **Figure 58**).

Figure 57: Positive terminal of battery
Source: BEA

Figure 58: Negative terminal of battery
Source: BEA

Each terminal was sliced into two parts.

- **Negative terminal**

There was no damage on the negative terminal. The various components were in position. The soldered connection was intact (**Figure 59**).

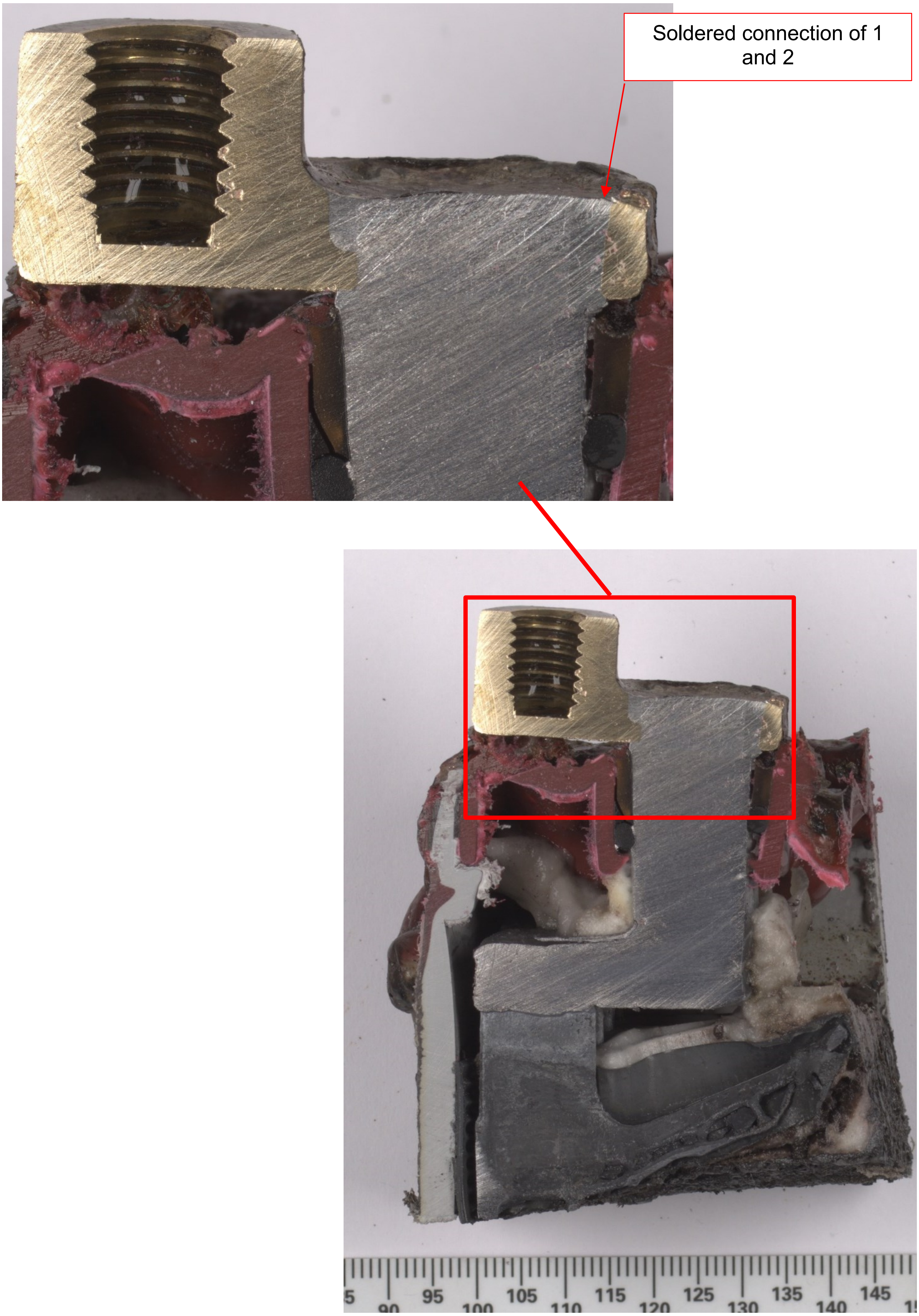


Figure 59: Cross-section of negative terminal
Source: BEA

- **Positive terminal**

The positive terminal had two singularities compared with the negative terminal:

- the solder which normally makes components 1 and 2 integral with each other (**Figure 56**) was not in its normal place. The solder seemed to have run downwards (**Figure 60**, **Figure 61** and **Figure 63**);
- the epoxy sealant was no longer present;
- arc type damage was observed in one of the section planes, level with the connection between the terminal and the battery cells (**Figure 62** and **Figure 64**);
- item 2 had stayed connected to the conductor.

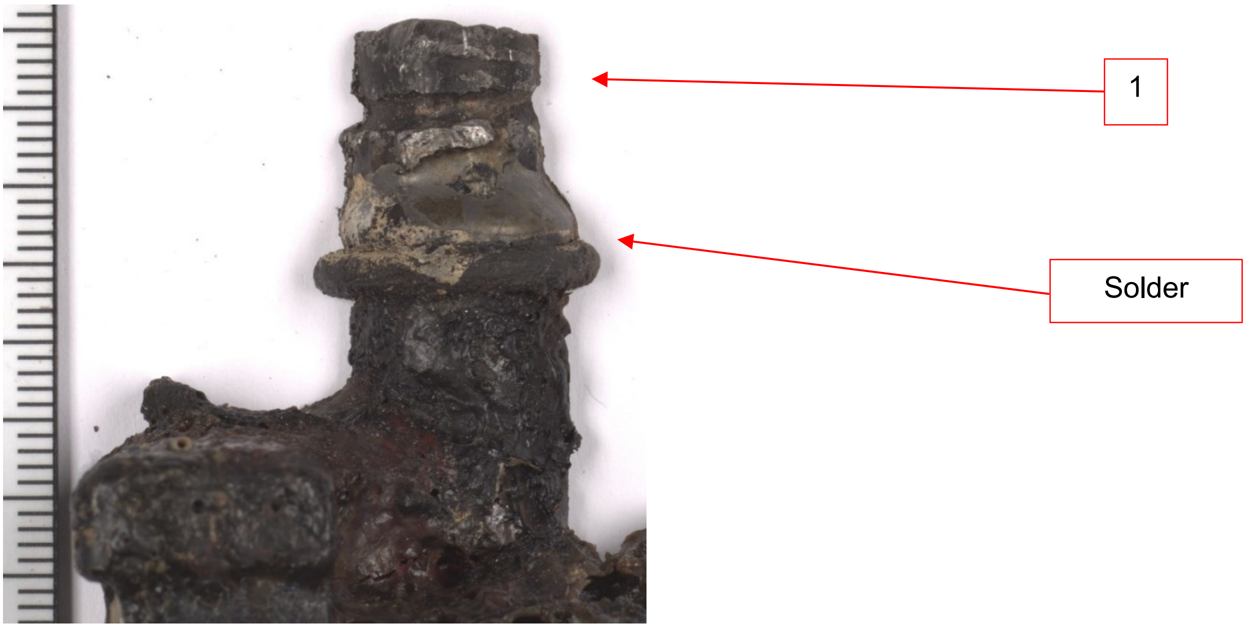
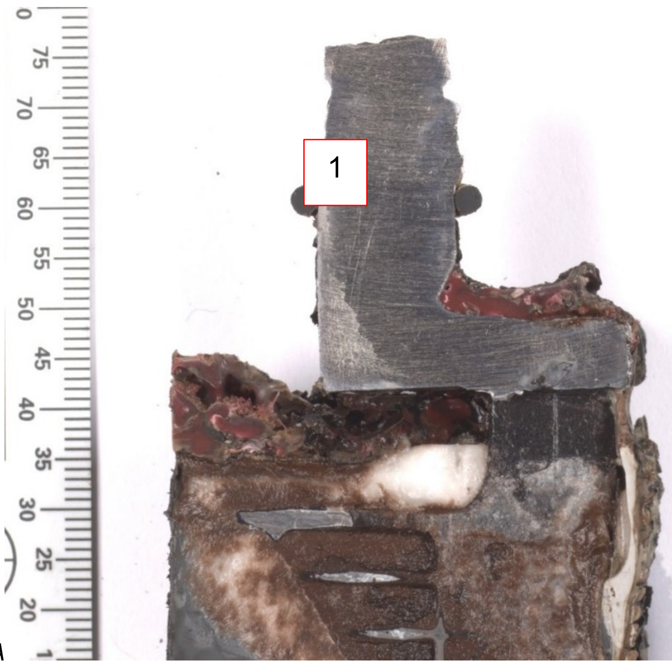


Figure 60: Positive terminal



Source: BEA

Figure 61: Cross-section of positive terminal
Source: BEA

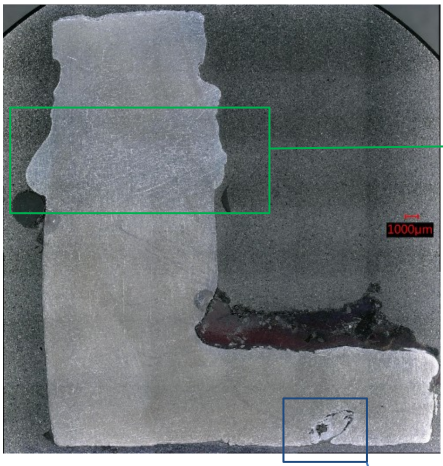


Figure 62: Overview of cross-section of positive terminal
Source: BEA

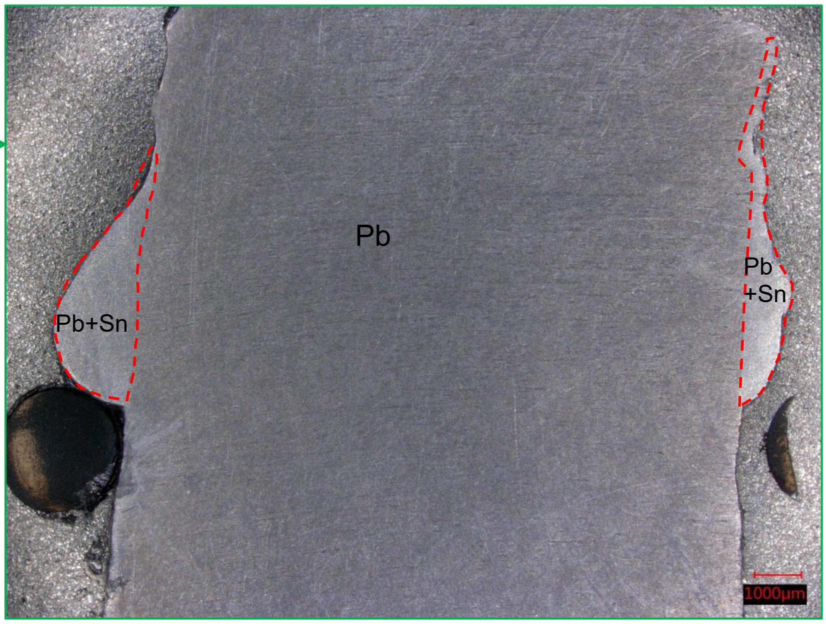


Figure 63: Detail of cross-section of positive terminal
Source: BEA

Figure 64: Arc type damage within positive terminal
Source: BEA

2.4.4.4 – Detailed description of conductor normally connected to positive terminal

The conductor showed signs of overheating, principally on its insulation (**Figure 65**). The destruction was greater on the part of the cable on the battery terminal side.

Figure 65: Conductor associated with positive terminal of battery
Source: BEA

Figure 66: Conductor associated with positive terminal of battery with connector
Source: BEA

Once the insulation had been removed, a bluish colouring could be seen on the conductor over a length of around 30 mm starting from the lug normally attached to the positive terminal (**Figure 67**). This was very probably a sign of overheating. In the same area, there was also a deformation of the strands making up the cable (**Figure 68**).

On an identical cable, the destruction of the insulation and this bluish colouring were reproduced after exposure to a temperature of between 250° and 300° (test carried out in a laboratory climatic chamber at ambient pressure).

A cross-section was made in an area of little cable damage. The diameter of the cable was measured by image analysis (**Figure 69**). As the cable was not perfectly circular, its diameter varied between 9.1 mm and 9.9 mm.

The EDS analysis determined that the conductor strands were of copper.

Figure 67: Bluish colouring of conductor (overheating of conductor)
Source: BEA

Figure 68: Deformation of strands making up conductor
Source: BEA

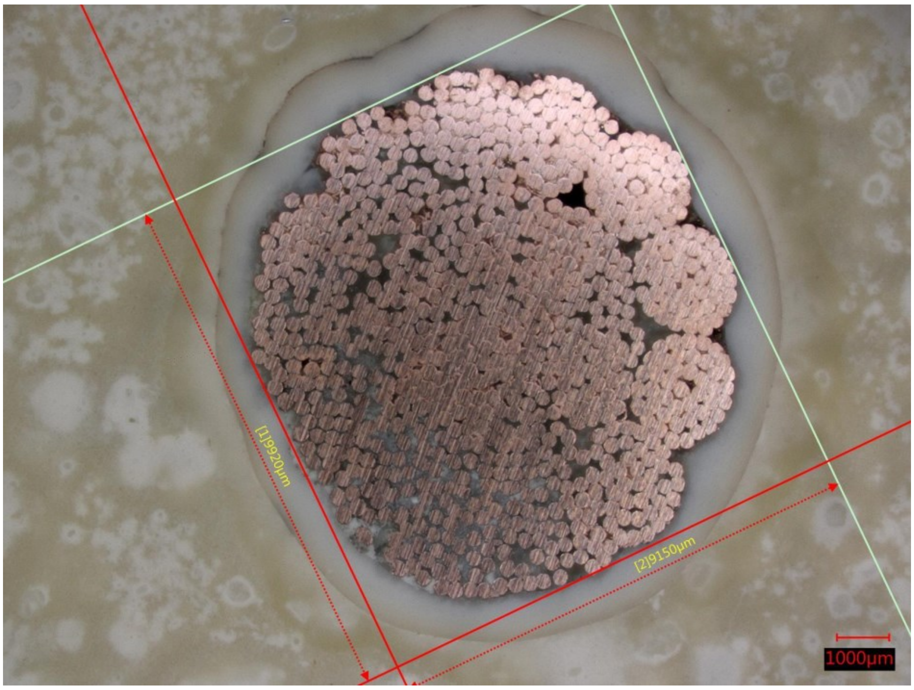


Figure 69: Cross-section of conductor
Source: BEA

2.4.4.5 – Tests carried out by Concorde Aircraft Batteries

Three tests were carried out by Concorde Aircraft Batteries on the same type of battery as the battery on HB-PNP. Le BEA was not present during these tests.

Tests carried out and results obtained:

Test 1:

Content and conditions:
This test consisted of heating the battery terminal with a 550 kW soldering iron.

Figure 70: Test 1
Source: Concorde Aircraft Batteries

The test was stopped when parts 1 and 2 of the terminal had separated (see **Figure 56**).

Findings:
Parts 1 and 2 of the terminal separated after a period of around 6 minutes. This separation was not associated with a flame. The damage was concentrated on the terminal.

The detailed examination of the terminal found that the solder had melted. The epoxy sealant was still present however.

Figure 71: Separation of parts 1 and 2 of terminal
Source: Concorde Aircraft Batteries

Figure 72: Detail of terminal after test
Source: Concorde Aircraft Batteries

Test 2:

Content and conditions:
This test consisted of heating the terminal battery with a 1000 W induction heater.

Figure 73: Test 2
Source: Concorde Aircraft Batteries

The test was stopped when parts 1 and 2 of the terminal had separated

Findings:
Parts 1 and 2 of the terminal separated after a period of around 2 minutes. This separation was not associated with a flame. The damage was concentrated on the terminal.

The detailed examination of the terminal found that the solder had melted. The epoxy sealant was still present however.

Figure 74: Separation of parts 1 and 2 of terminal
Source: Concorde Aircraft Batteries

Figure 75: Detail of terminal after test
Source: Concorde Aircraft Batteries

Test 3:

Content and conditions:

This test consisted of heating the environment of the positive terminal of the battery using a flame generated by a hydrogen/oxygen system.
The test was stopped when parts 1 and 2 of the terminal had separated.

Findings:
Parts 1 and 2 of the terminal separated after a period of around 2 minutes.

The detailed examination of the terminal found that the solder had melted. The epoxy sealant was no longer present.

Figure 76: Separation of parts 1 and 2 of terminal
Source: Concorde Aircraft Batteries

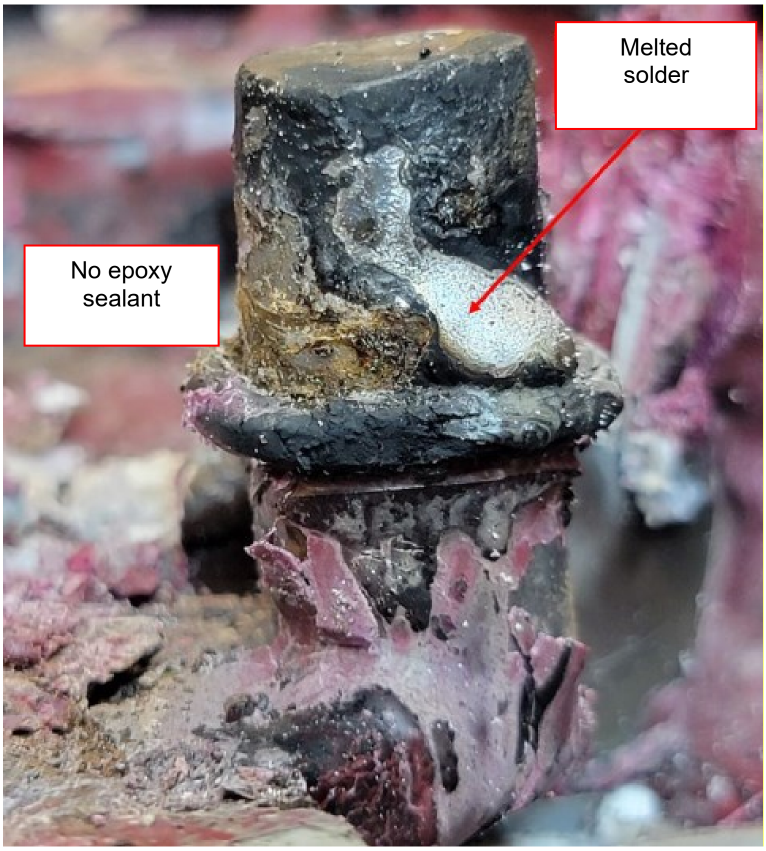


Figure 77: Detail of terminal after test
Source: Concorde Aircraft Batteries

2.4.4.6 – Examination of an identical battery

2.4.4.6.1 – Aeroplane registered HB-PPB

The Piper PA-28RT-201T registered HB-PPB was maintained in the same workshop as the damaged aeroplane. Its electrical system was also similar to that of the damaged aeroplane.

HB-PPB was equipped with Concorde battery RG-35AXC (**Figure 78**), identical to the one on HB-PNP.

Figure 78: Battery removed from HB-PPB
Source: BEA

According to the battery manufacturer's documentation, its specifications are the following:

Volts	12 V
Capacity C_1 *<u>Definition</u> (source: Concorde Aircraft Batteries technical bulletin dated 1 May 1998): Quantity of electricity (in Ah) which the cell or battery is capable of delivering in 1 h, throughout its normal service life, after full charge, under conditions defined with regard to temperature and end discharge voltage.*	33 AH
Maximum weight	14.5 kg

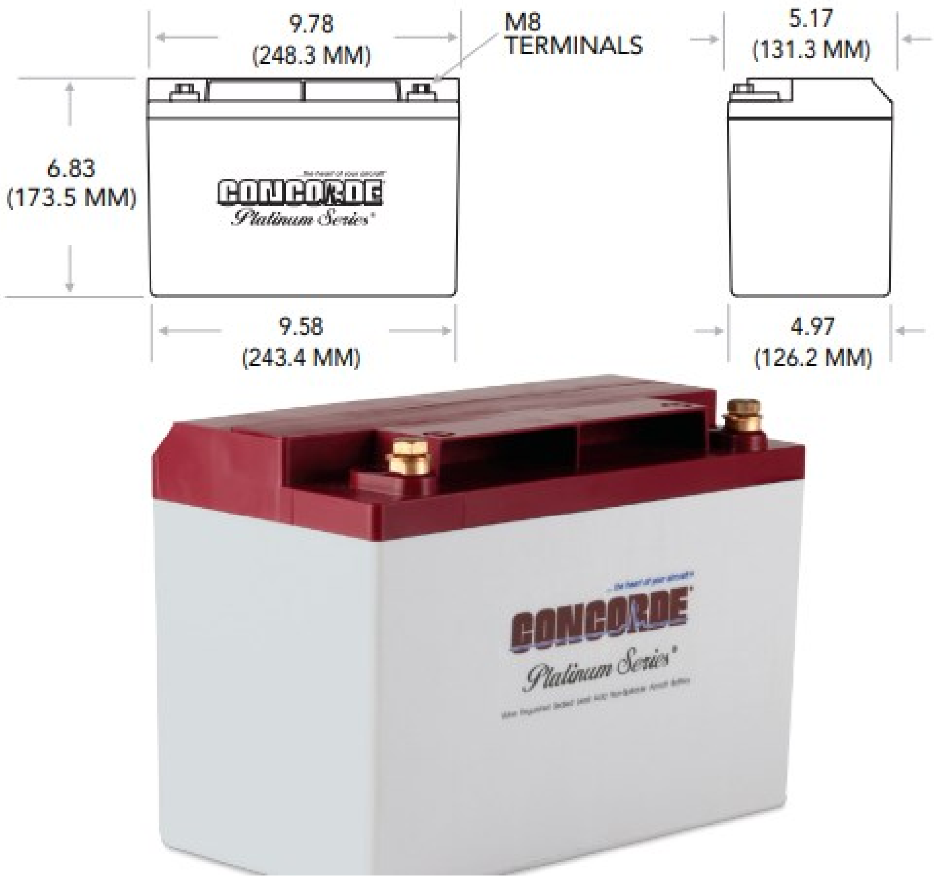


Figure 79: Battery dimensions

Source: https://www.concordebattery.com/knowledge-base.html?media=brochure

2.4.4.6.2 – History of battery

The last battery checks were the following:

Date	**SoH[13] measurement**
08 Jan 2019	95.5%
12 Aug 2019	96.3%
23 Jan 2020	94%
13 Oct 2020	87%
28 April 2021	81.6%

According to the battery manufacturer, these values correspond to nominal aging.

[13] SoH: State of Health of the battery. The SoH indicates the wear of a battery.

The manufacturer, Concorde Aircraft Batteries, also gave the BEA the following information:

- the battery was sent out by CONCORDE on 31 August 2017;
- at this time, it weighed 13,644.1 g.

2.4.4.6.3 – External appearance of battery

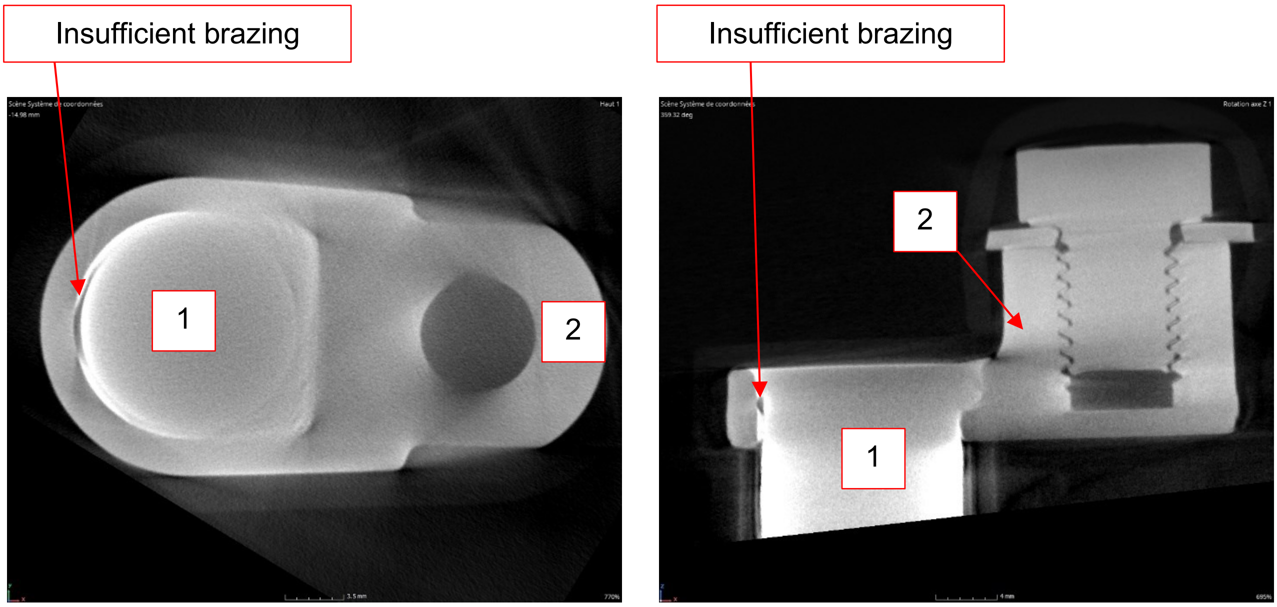
No damage was found.

2.4.4.6.4 – Non-destructive examination of battery terminals

Each battery terminal was subject to a tomographic examination. The resolution of this examination was 56 µm. With this resolution level, the ability to observe cracks cannot be guaranteed.

No particularity was identified on the positive terminal.

Insufficient brazing was observed locally in line with the negative terminal (**Figure 80**).



Top view Side view

Figure 80: Negative terminal
Source: BEA

2.4.4.6.5 – Process, tests and measurements

The tests and measurements were carried out in the National Solar Energy Institute (INES) laboratory of the French Atomic Energy Agency (CEA) at Chambéry.

The process followed during this work is set out in the table below.

Process phase	Operations
0	Physical examination as defined in standard NF EN 60952-1
1	Assessment of the State of Charge (SoC)
2	Battery completely charged
3	Three recharge[14] - discharge[15] cycles Definition of battery capacity Definition of battery SoH
4	Examination of overload behaviour at 23°C and at 40°C
5	Examination of behaviour during a micro-cycle at I = 70 A

During these tests, in addition to the electrical measurements, three thermocouples were installed. Their positions (TC1, TC2 and TC3) are indicated in **Figure 81** below.

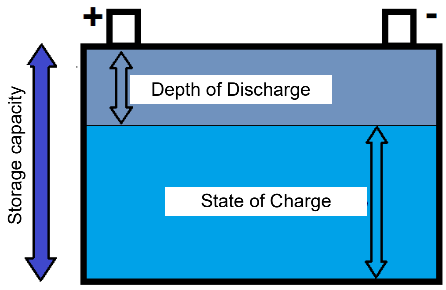
Figure 81: Position of thermocouples on battery
Source: INES CEA

Assessment of SoC on battery's arrival at laboratory

The SoC, expressed as a %, is the residual capacity relative to the nominal capacity of the accumulator at the time of measurement t, in other words the remaining energy in the battery.

The Depth of Discharge (DoD) expressed as a %, is the capacity that has been discharged relative to the nominal capacity of the accumulator. In other words, the energy used in the battery.

The sum of the values SOC and DOD is always 100%.



[14] The battery was systematically charged at a temperature of 21°C, at a constant voltage of 14.1 V and for a minimum duration of 20 h.

[15] The battery was discharged as per the specifications of standard NF EN 60952-1 / chapter 5.1 - Capacity test at a 1 I1 rate (version in force published on 2014), at a temperature of 21°C, a constant current of 33 A, with the end of process being defined as when the cut-off voltage (EPV) is reached, i.e. 10 V (1.67 V per cell as defined in standard NF EN 60952-1).

The SoC was determined by defining the relationship between the Open Circuit Voltage (OCV) and the state of charge with the following equation (conventional method regularly used by the CEA):

$$SoC\ (\%) = (83.333 \text{ x } OCV) - 975$$

On arriving in the laboratory, the battery had the following characteristics:

Weight	13,634.8 kg. => *i.e. 9.3 g less than the weight of the battery when it was initially sent out by Concorde in 2017. This weight change is not significant.*
Impedance at 1 kHz	5.65 mOhm
OCV	12.76 V

The SoC was therefore determined as being SoC = 88 %.

Definition of battery capacity, definition of SoH

SoH is the abbreviation of State of Health.

The SoH is a percentage which evolves from 100% to 0% and is used to precisely characterise the level of deterioration of a battery. Initially, a battery has a certain capacity expressed as ampere-hour (Ah). Over time, batteries undergo internal electrochemical reactions and physical degradation that reduce this initial capacity.

The SoH is therefore used to compare the maximum capacity of the battery at a given time with the maximum capacity of the battery when it was new.

The three battery discharge tests defined the battery capacity as between 10 and 10.5 Ah, i.e. a SoH of between 30 and 32%. A significant difference can therefore be noted between this value and the value defined on 28 April 2021 of 81.6%. This difference seems to indicate early ageing of the battery between 28 April 2021 and the date of examination at the CEA. The removal of the battery and its storage without charge before the date of examination at the CEA was probably a contributory factor.

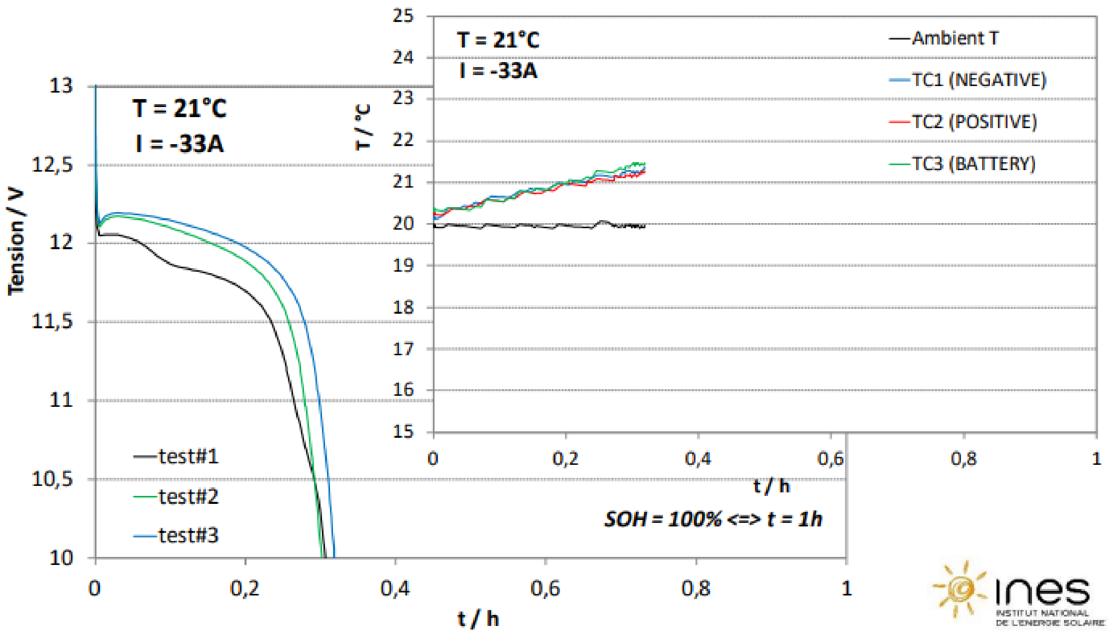


Figure 82: Data acquired during three battery discharge operations
Source: INES CEA

Examination of overload behaviour at 23°C ± 5°C and at 40°C

These tests were based on the specifications of standard NF EN 60952-1 (chapter 5.12 - Service life in overload), with the following adaptations:

- these tests were carried out with a partially charged battery, after the previous discharge operation;
- the voltage was stabilised at 14 V (14.5 V required by standard);
- the test time was reduced to 150 h.

After the overload phase, the battery was discharged in the same way as above.

On completion of the test carried out at 21°C, the following observations were made:

- current stabilised at 86 mA after 80 test hours;
- no elevation in battery temperature;
- capacity when discharged slightly higher than that measured previously.

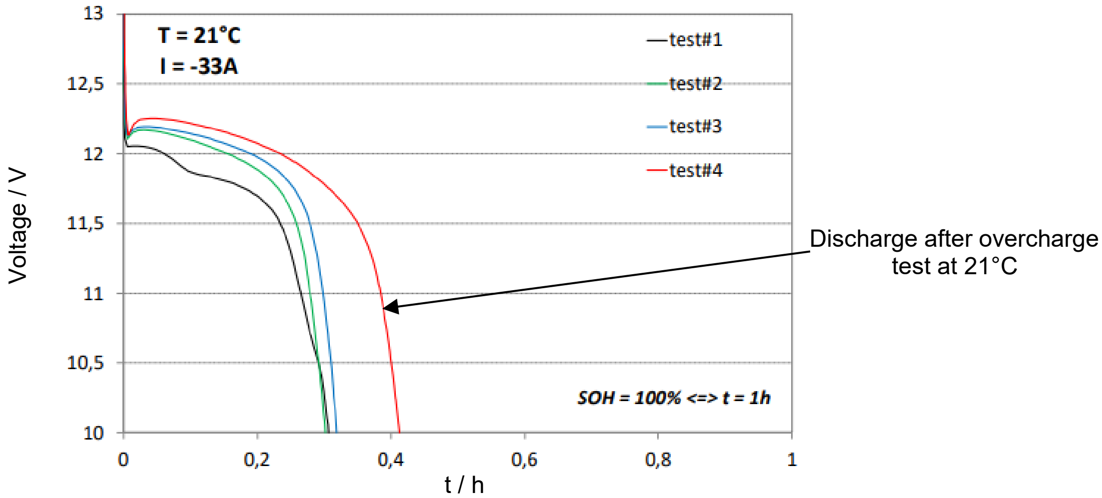


Figure 83: Data acquired during battery discharge after overload test at 21°C
Source: INES CEA

On completion of the test carried out at 40°C, the following observations were made:

- no elevation in battery temperature;
- capacity during discharge far higher than that measured previously.

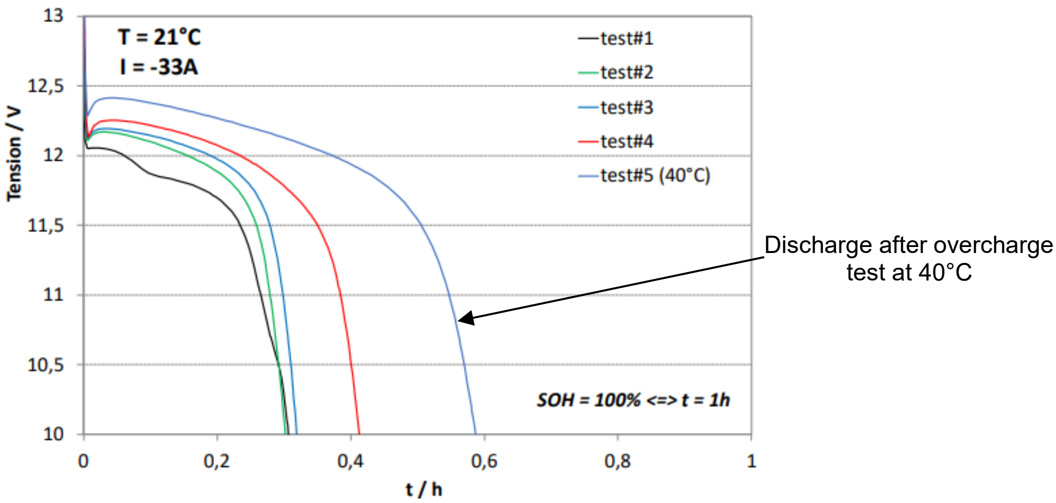


Figure 84: Data acquired during battery discharge after overload test at 40°C
Source: INES CEA

Examination of behaviour during a micro-cycle at I = 70 A

During these tests, it was observed that:

- the battery voltage remained in the specified limits of use;
- the temperature of the battery's "body" increased to around 50°C;
- the temperature at the battery's terminals increased to around 40°C.

According to the CEA experts, these results show that the application of such a current on a battery with a lower electrolyte saturation of the separator could initiate a violent thermal runaway process. If this happens, the temperature will rise further.

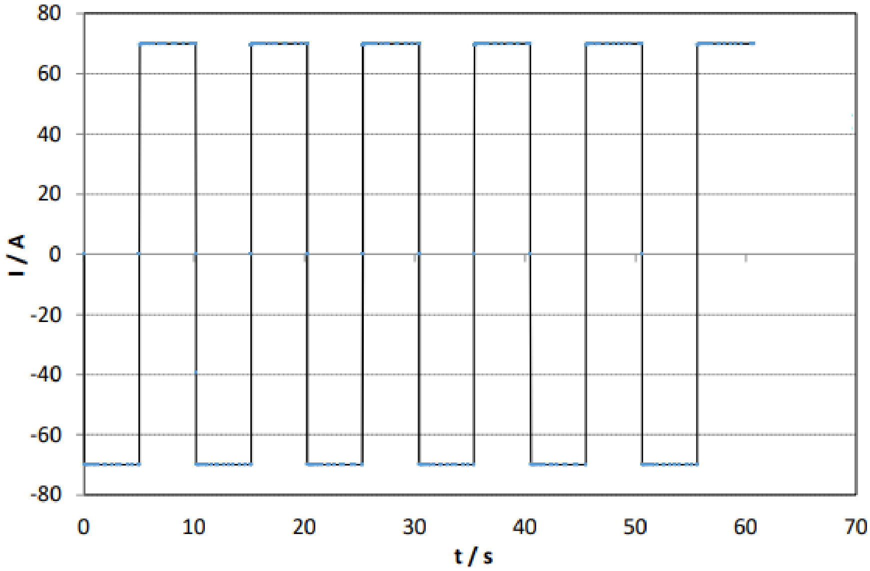


Figure 85: Current profile during test
Source: INES CEA

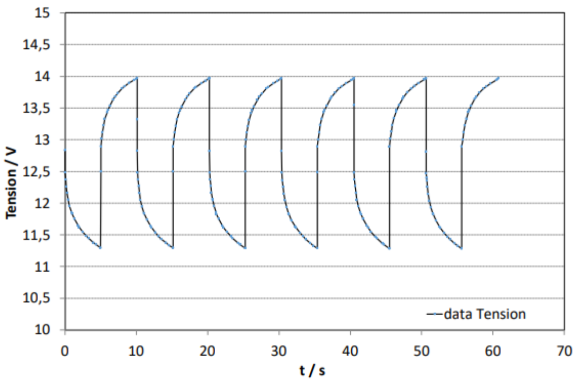


Figure 86: Voltage profile at beginning of test
Source: INES CEA

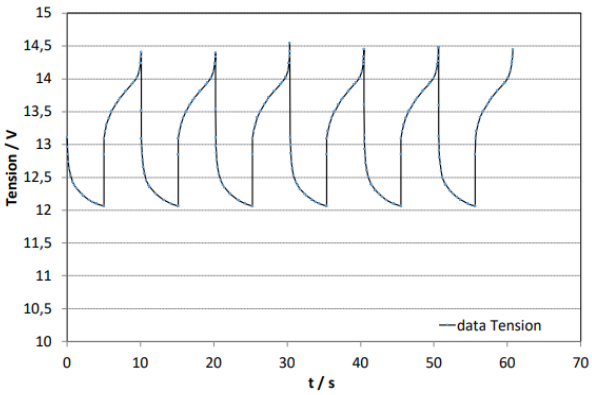
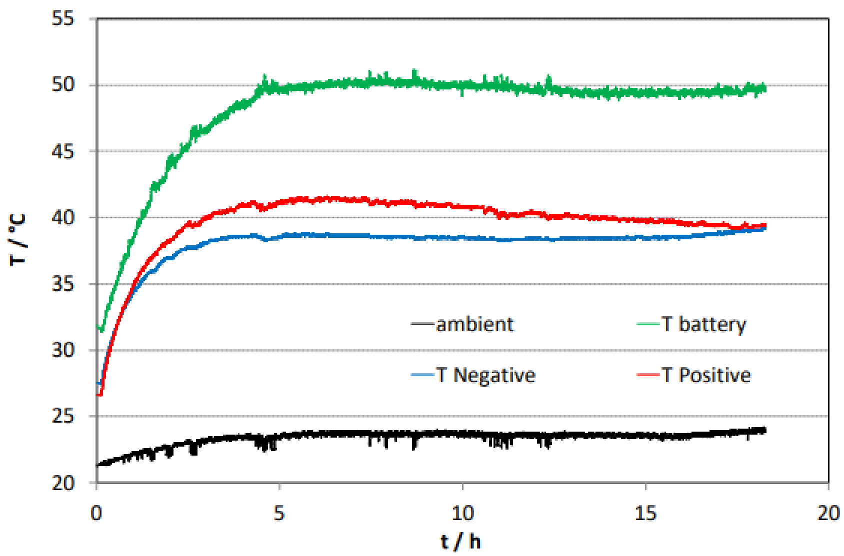


Figure 87: Voltage profile at end of test
Source: INES CEA



Key:

ambient temperature

battery body temperature

temperature at positive terminal

temperature at negative terminal

Figure 88: Temperature measurements during test
Source: INES CEA

2.4.4.6.6 – Destructive examination

<u>Note:</u> The battery's weight remained unchanged between the start and end of the tests described above. The internal resistance of the battery also remained constant. These two findings indicated that the electrical tests were not "destructive". The observations during the destructive examination effectively showed that the battery had aged before it was sent to the CEA.

The CEA specialists defined battery RG-35AXC as a very high-quality assembly.

The battery was composed of six cells (**Figure 89**).

Figure 89: Cells making up battery
Source: INES CEA

The table below indicates for each cell, the sulphuric acid density (at 25°C and for 12.99 V at the battery terminals before the battery was opened), the voltage and the internal resistance:

Cell number	Sulphuric acid density	Voltage(V)	Internal resistance (Ohm)
1 (associated with negative terminal of battery)	1.323	2.153	0.94
2	1.337	2.166	0.85
3	1.340	2.170	0.83
4	1.337	2.166	0.86
5	1.332	2.160	0.87
6 (associated with positive terminal of battery)	1.326	2.155	0.96

Each cell had eight positive plates measuring 110 x 115 mm and 1.4 mm thick and nine negative plates measuring 110 x 115 mm and 1.1 mm thick (**Figure 90**). The positive plates were enveloped in (thickness of envelope: 1.08 mm):

- outer coat of polyethylene;
- inner coat of AGM, well saturated in electrolyte.

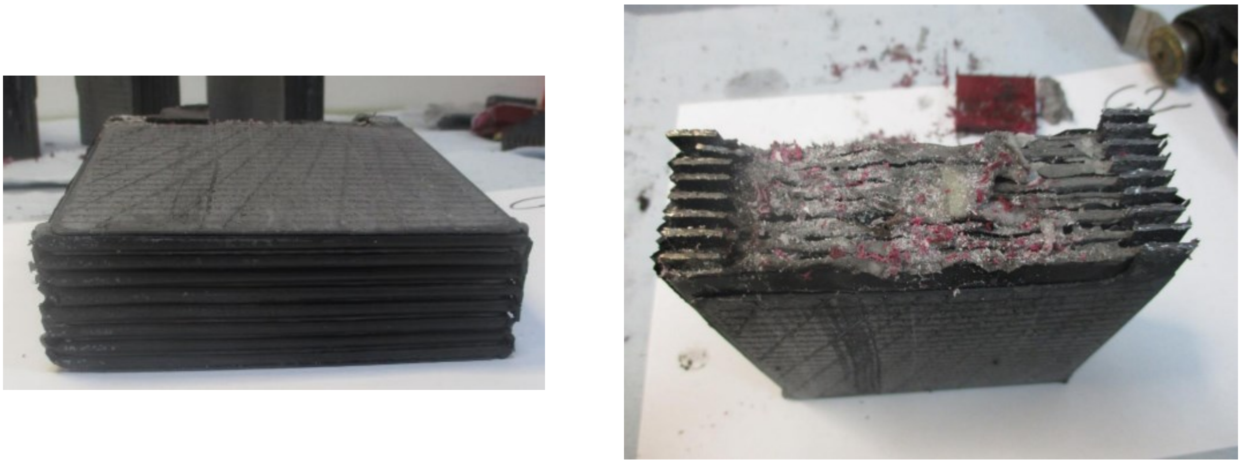


Figure 90: Cell 2
Source: INES CEA

The detailed examination of each cell did not identify damage associated with the manufacturing and assembling process.

The negative plates were in very good condition with a homogeneous appearance (**Figure 91**).

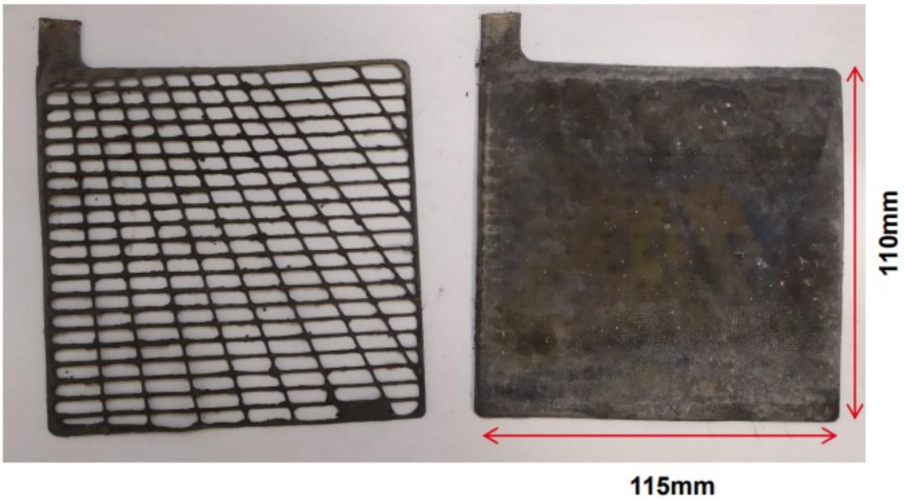


Figure 91: Example of negative plate
Source: INES CEA

The positive plates had moderate to severe corrosion (**Figure 92**). These plates had become very fragile due to this damage.

According to the CEA specialists, this degraded state of the positive plates may explain the battery's low capacity measured in the laboratory.

Figure 92: Example of positive plate
Source: INES CEA

The valves were in very good condition.

2.4.4.6.7 – Conclusion of CEA experts

In conclusion, these tests showed that:

- When the battery arrived in the laboratory, its SoH was around 30%, a lot lower than the last check carried out on 28 April 2021. According to the CEA specialists, the substantial decrease in the battery's capacity could be the result of the corrosion of the positive plates. This corrosion is the most common ageing mechanism at the end of the service life of lead batteries, with a usage pattern similar to that on aircraft.
- The results of the micro-cycle tests at I = 70 A showed that the application of such a current on a battery with a lower electrolyte saturation of the separator could initiate a violent thermal runaway process.

- The battery was capable of safely operating at a voltage of 14 V, at a temperature of 40°C. This finding was consistent with the observations made during the destructive examination, as well as with the technology of this battery which protects it from excessive oxygen recombination.

2.4.4.6.8 – Comments and tests carried out by Concorde Aircraft Batteries

Concorde Aircraft Batteries broadly agreed with the work carried out.

The two main comments are the following:

- The small capacity measured in the laboratory was probably the result of a sulfation phenomenon due to the battery being stored without being recharged from 28 April 2021 to the measurement date in the CEA laboratory.

  According to the CEA specialists, the sulfation phenomenon did not occur. They substantiated this by the findings made during the destructive examination.

- During the micro-cycle test at I = 70 A, the temperature of the battery's body reached 50°C and then remained stable at this temperature. According to the manufacturer, it is not possible to affirm the possibility of incipient thermal runaway based on this data.

  According to the CEA specialists, the micro-cycle test minimised the increase in temperature with respect to a situation in which the amperage would be continuous.

Concorde Aircraft Batteries carried out tests based on three scenarios described below. The tests were carried out with a six-year old RG-35AXC battery manufactured in April 2017.

- **<u>Scenario 1:</u>**

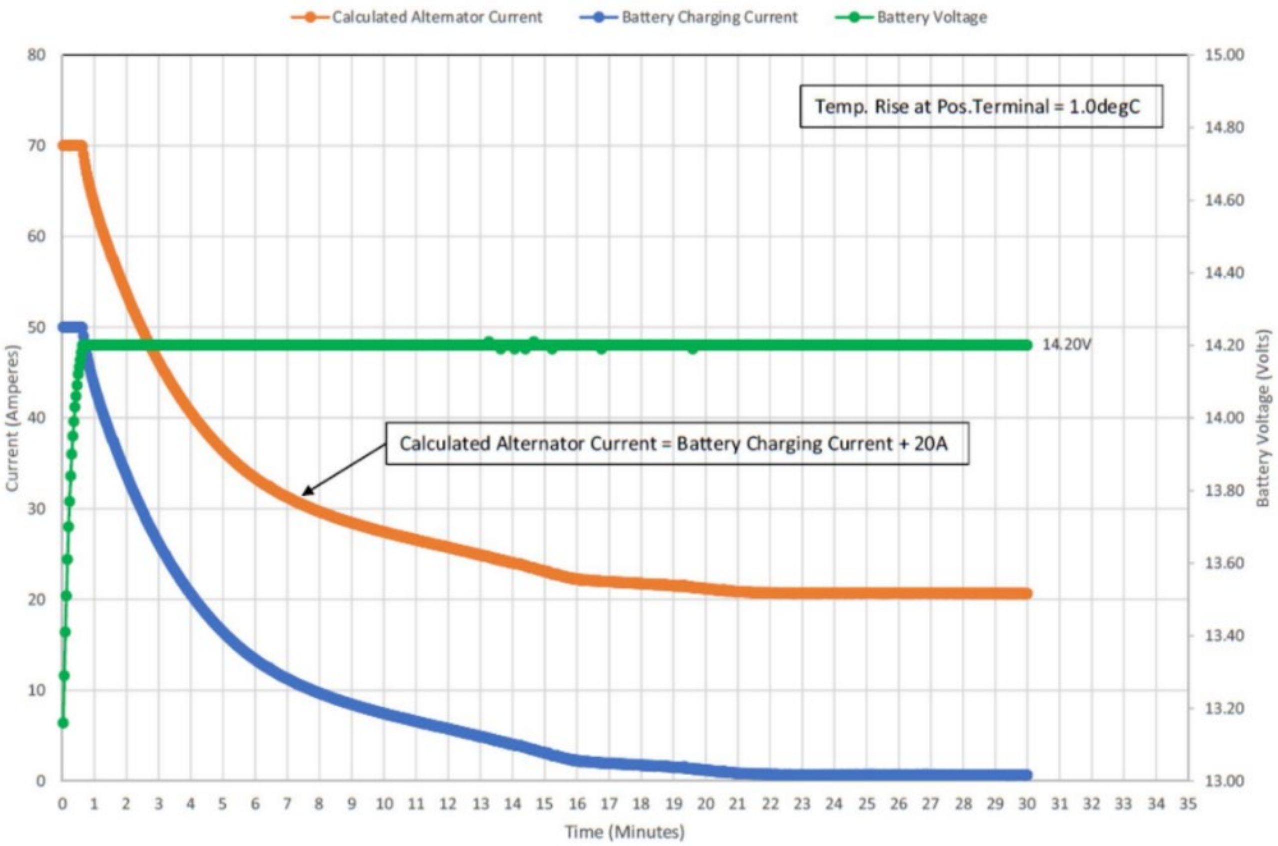
In this scenario, the battery starts up the aeroplane's engine and then the alternator recharges the battery normally after the engine has started up and during the flight. It was estimated that the before flight checks and engine start-up reduced the battery capacity by around 12%.

A current of 33 A for 5 minutes was used to simulate the loads before the engine start-up.

Three engine start-ups were simulated (peak amperage = 330 A, duration = 15 seconds).

A load voltage of 14.2 volts was assumed, which corresponds to the typical setting of voltage regulators.

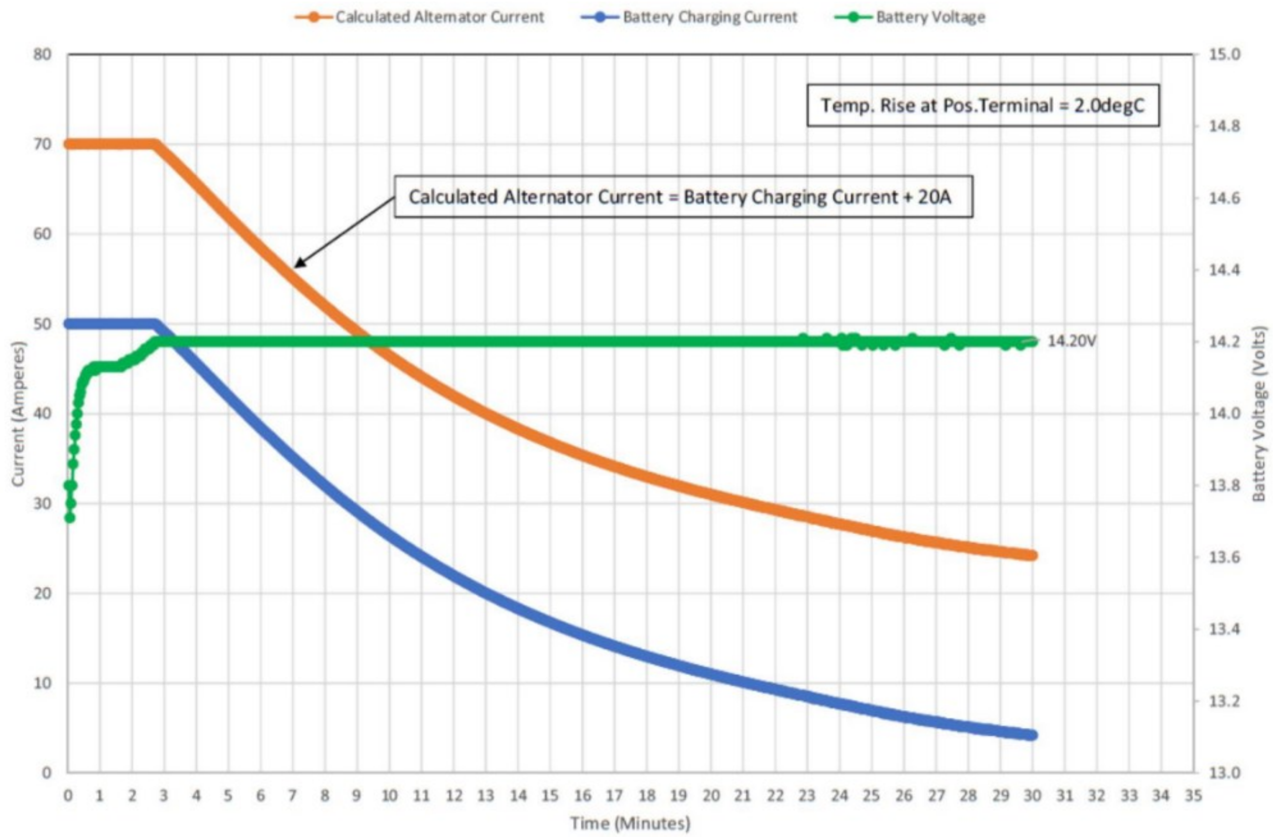
The alternator delivered its maximum power with a maximum output amperage of 70 A for less than a minute. Once the battery voltage reached 14.2 volts, the voltage regulator maintained this set point and the current decreased as the battery's state of charge increased. After 15 minutes, the alternator current stabilised at around 23 A. It should be noted that the temperature measured at the positive terminal only rose by 1°C during this test.



- **<u>Scenario 2:</u>**

In this scenario, the battery starts up the aeroplane's engine and then the alternator recharges the battery normally after the engine has started up and during the flight. It was estimated that the before flight checks and engine start-up reduced the battery capacity by around 34%.

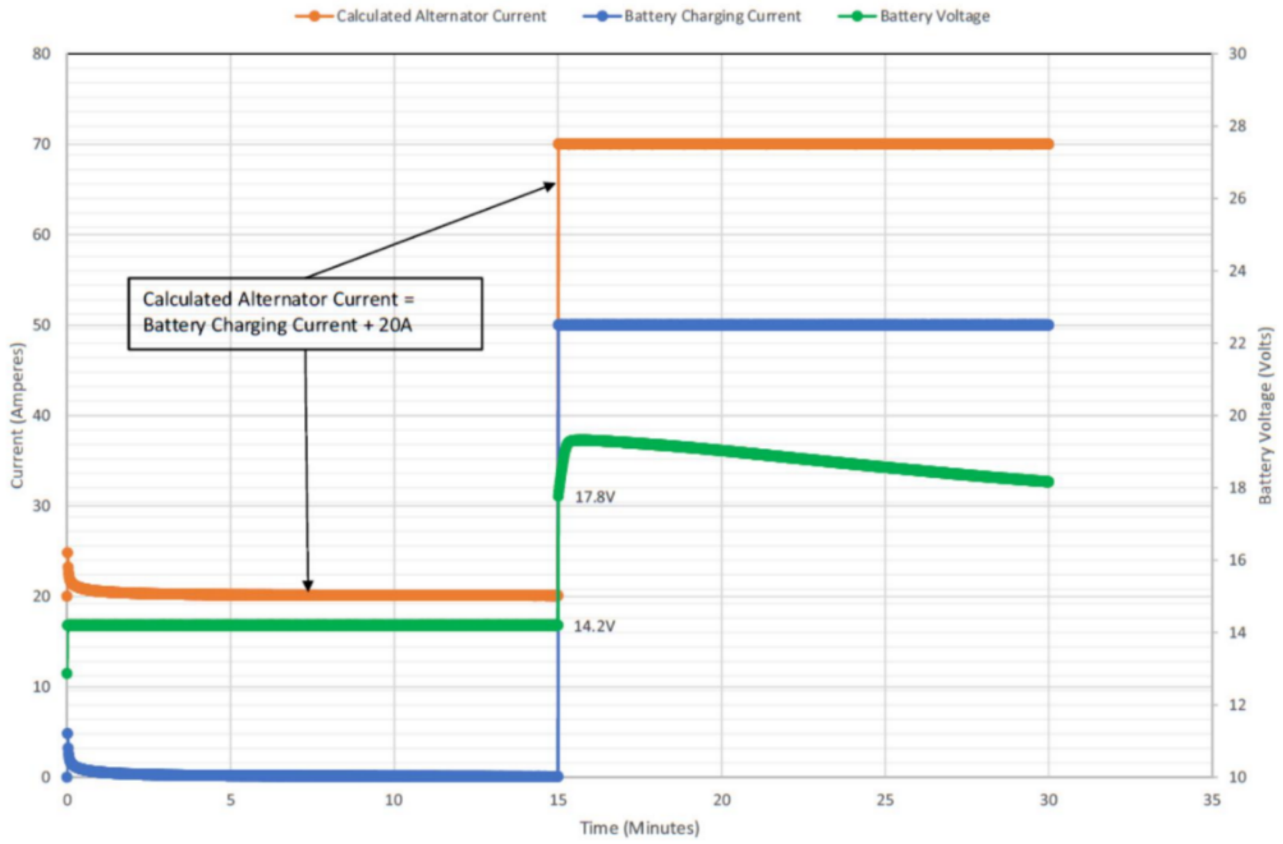
The alternator delivered its maximum power with a maximum output current of 70 A for less than 3 minutes. Once the battery voltage reached 14.2 volts, the voltage regulator maintained this set point and the current decreased as the battery's state of charge increased. After 15 minutes, the alternator current stabilised at around 37 A. It should be noted that the temperature measured at the positive terminal only rose by 2°C during this test.



- **<u>Scenario 3:</u>**

Scenario 3 assumes that the sudden increase in alternator output current (up to 70A) was due to a sudden failure of the voltage regulator, resulting in the alternator supplying the battery with a high output voltage.

The battery was charged to 14.2 volts for 15 minutes, then a constant current of 50 A was applied for 15 minutes. The battery voltage immediately exceeded 17 V (the first data point in 1 second was 17.8 V). The temperature of the positive terminal reached 32°C during this test, i.e. an increase of only 13°C.



Concorde Aircraft Batteries specifies that there is no scenario or known battery failure mode which could explain the sudden increase in the alternator's power output after a flight time of 15 minutes.

2.4.5. Diode device between alternator and master solenoid

2.4.5.1 – Composition of diode device

This diode device is composed of a metal cooler, secured to the floor of the aircraft at four points. A detailed diagram of how it is assembled to the aeroplane floor is given in **Figure 94**.

The metal cooler is made of aluminium alloy. Its exterior is protected by an electrolyte anodised layer, with a melting temperature of around 700°C (defined by the BEA using oven tests[16]).

Four identical diodes, manufactured by Motorola, are positioned in the centre of the metal cooler. (**Figure 93**).

[16] These tests were started at 250°C and continued until fusion was observed. Fusion occurred between 690 and 720°C. At each temperature interval, the temperature was stabilized for 30 minutes. These tests were carried out at room pressure. The oven used was a Carbolite GERO, type ELF 11/23 (2022 model).

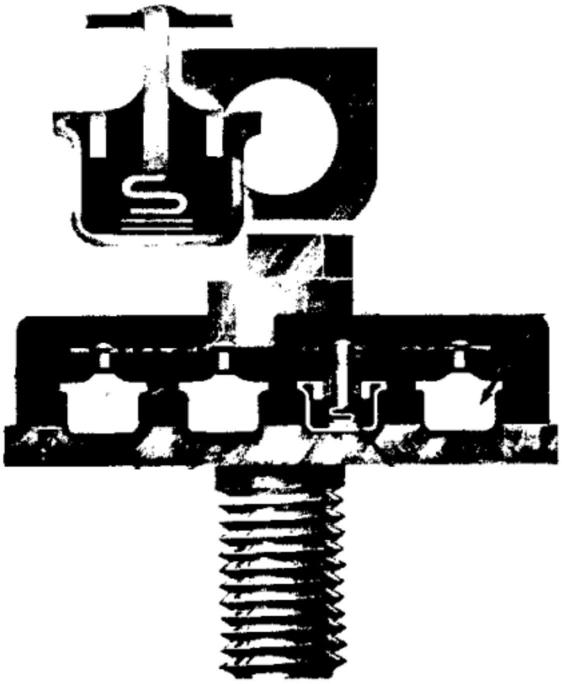


Figure 93: Diagram taken from equipment datasheet
Source:
https://www.datasheetarchive.com/pdf/download.php?id=f047b9144a0ad735bf38bf912096678e62d51e&type=O&term=MR1215SLR

This device is attached to the aeroplane's floor. It is electrically insulated from the floor by a phenolic shield and electrically insulated from the fastening parts by insulating washers.

Based on the information provided by Concorde Aircraft Batteries, the phenolic shield has the following characteristics:

- flashpoint: 349°C;
- self-ignition temperature: 508°C.

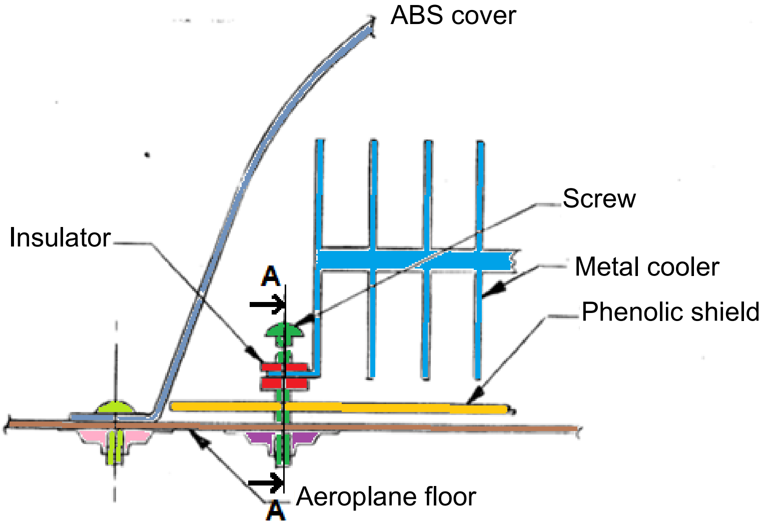
The softening temperature and melting temperature of an insulating washer were defined by the BEA by means of oven tests[17]. The temperatures are the following:

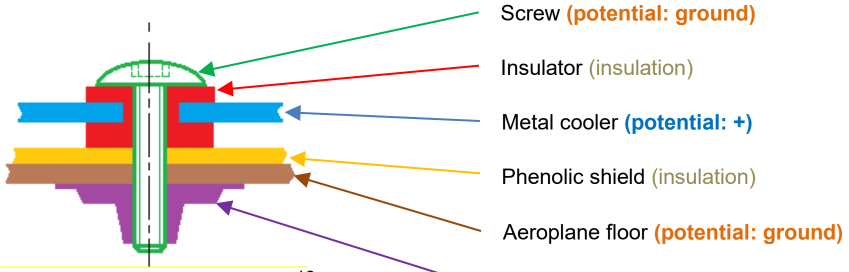
- softening temperature: during the tests carried out, softening started at around 450°C;
- melting temperature/destruction by thermal effect: destruction was observed at a temperature of between 480 and 500°C.

Softening of insulating washer

Destruction of insulating washer

[17] These tests started at 80° with the temperature being increased by 30° intervals until fusion was observed. At each temperature interval, the temperature was stabilised for 30 min. These tests were carried out at room pressure. The oven used for temperatures between 80 and 290°C was a Memmert. The oven used for temperatures over 290°C, was a Carbolite GERO, type ELF 11/23 (2022 model).





A-A schematic diagram[18]

Figure 94: Diagram of assembly of cooler on aeroplane floor

Source: https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&ved=2ahUKEwjWsbr3qf-AAxV2VaQEHT1dDtsQFnoECAcQAQ&url=https%3A%2F%2Fwww.dolmint.com%2Fsites%2Fdefault%2Ffiles%2Fservice%2FSB_0623.pdf&usg=AOvVaw0nCM9aih5LzjOE1eou5DNK&opi=89978449

[18] The figure proposed is a schematic diagram to facilitate understanding and is not a to-scale representation.

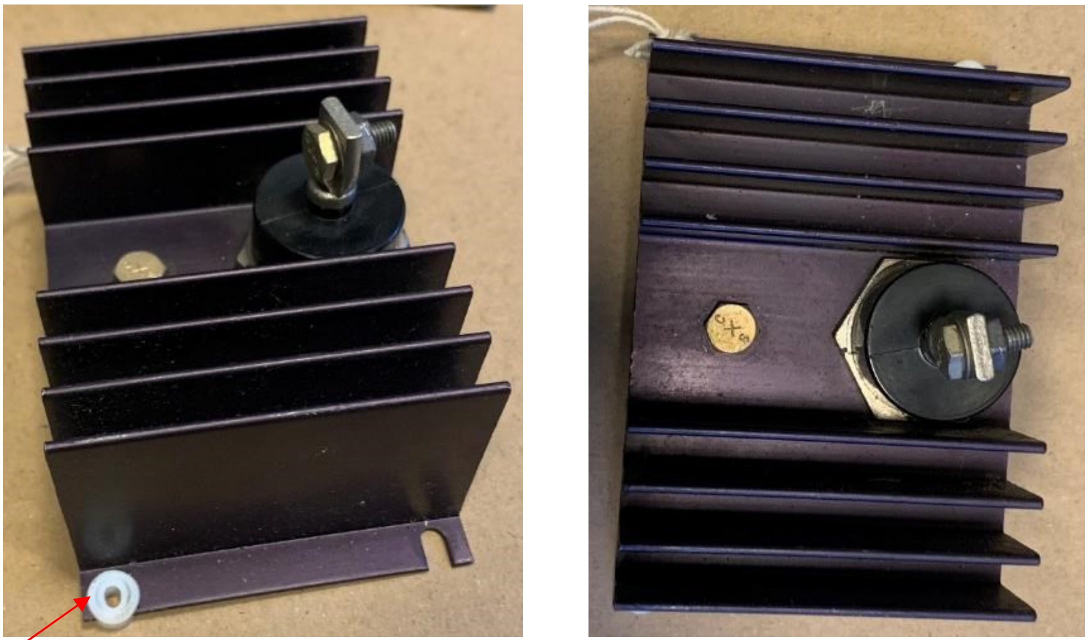


Figure 95: Diode device supplied by Concorde Aircraft Batteries to the BEA[19]
Source: BEA

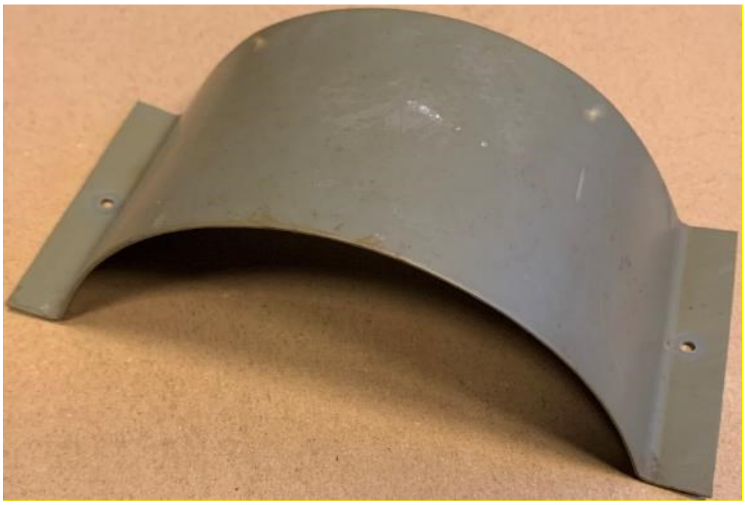


Figure 96: ABS cover supplied by Concorde Aircraft Batteries to the BEA
Source: BEA

The aeroplane manufacturer, Piper, published Service Bulletin No 623 on 1 November 1978[20]. HB-PNP's serial number is not one of the numbers concerned by this Service Bulletin.

This document specifies that if one or several insulating washers are incorrectly installed, the associated attaching screws may come into mechanical contact (and therefore electrical contact) with the diode device, causing a temporary short-circuit and maximum output demand from the alternator.

[19] When this item was received by the BEA, it was observed that an insulating washer (normally four) was missing.
[20] https://www.google.com/url?sa=t&rct=j&q=&esrc=s&source=web&cd=&ved=2ahUKEwjWsbr3qf-AAxV2VaQEHT1dDtsQFnoECAcQAQ&url=https%3A%2F%2Fwww.dolmint.com%2Fsites%2Fdefault%2Ffiles%2Fservice%2FSB_0623.pdf&usg=AOvVaw0nCM9aih5LzjOE1eou5DNK&opi=89978449

Piper also informed the BEA of the following points:

- The Piper maintenance manual applicable for model PA-28RT-201T has no maintenance requirement with respect to this diode device.
- Service Bulletin No 623, published on 1 November 1978, applies to various Arrow models manufactured between 1975 and 1978. PA-28RT-201T was manufactured between 1979 and 1988, which is outside of the applicable dates given in SB 623.

2.4.5.2 – Detailed examinations of diode device

The diode device had the following damage:

- The ABS cover (cf **page 22**) was totally destroyed. Only the flanges of this cover were still partially present with their attaching screws in position and tightened. The residue from the flange along the fuselage was more substantial.
- In one of its corners, the metal cooler had thermal effect damage which had resulted in the melting of the alloy of which it is composed (**Figure 101**). The associated attaching screw and insulating washer were no longer present. However, the three other attaching screws were in position (**Figure 98**)[21].
- The metal cooler also had a brown/purple discolouration. Initially, the metal cooler was black in colour. Concorde Aircraft Batteries informed the BEA that a test had been carried out on a similar metal cooler, by exposing it to a propane torch. At the end of this test, Concorde Aircraft Batteries noted that the external colour of the metal cooler had changed from black to “purple”.
- The phenolic shield was in position under the metal cooler (**Figure 99**). The shield was blackened around the attaching point in line with where the metal cooler had melted (attaching point arbitrarily numbered Pt1) (**Figure 99** and **Figure 100**). The shield was also partially burnt in line with the front left corner (around the attaching point arbitrarily numbered Pt3), situated the closest to the positive terminal of the battery.
- The aluminium alloy plate making up the aeroplane floor was blackened around the attaching point of the melted part of the cooler (attaching point arbitrarily numbered Pt1) (**Figure 101**). The anchor nut holding the attaching screw also had a darkened appearance (**Figure 102**). This darkening was not observed around the three other attaching points (Pt2, Pt3 and Pt4).
- The diode assembly had separated from the metal cooler after the melting of the brazed areas (**Figure 103**);
- The conductor coming from the alternator was correctly connected to the diode device (**Figure 104** and **Figure 105**). However, the core of the conductor was oxidised and showed "swelling" level with the lug fixed to the metal cooler.
- The 5A fuse, connected to the diode device, was in position and whole (**Figure 106**). The associated fuse holder was intact and undamaged.

In conclusion, the findings on the diode device indicated an abnormally high increase in temperature, well in excess of the amount of heat that can be evacuated by the metal cooler.

[21] To simplify understanding in the rest of the document, the metal cooler attaching points are arbitrarily numbered Pt1, Pt2, Pt3 and Pt4 as shown in **Figure 98**.

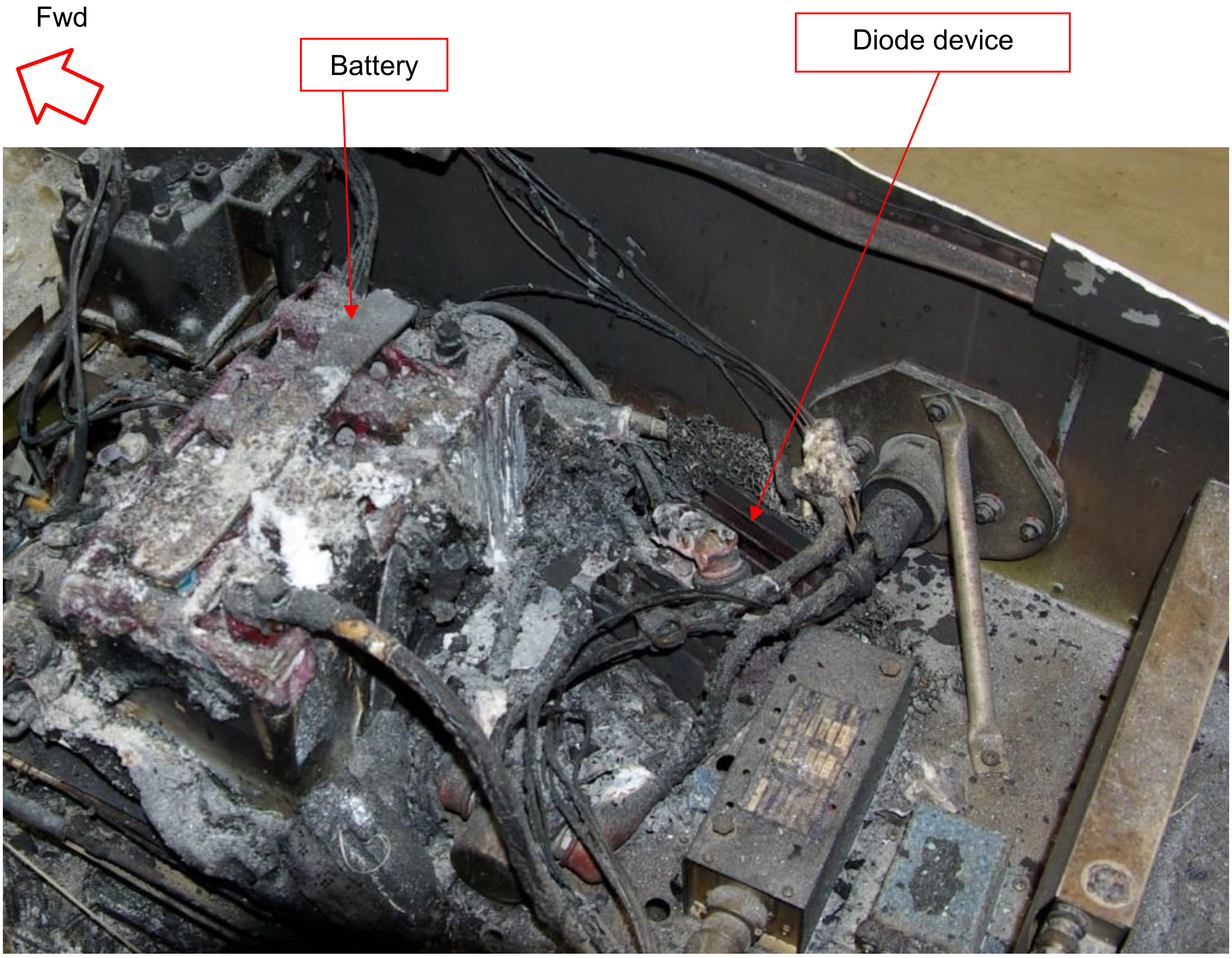


Figure 97: Diode device in position on aeroplane floor
Source: BEA

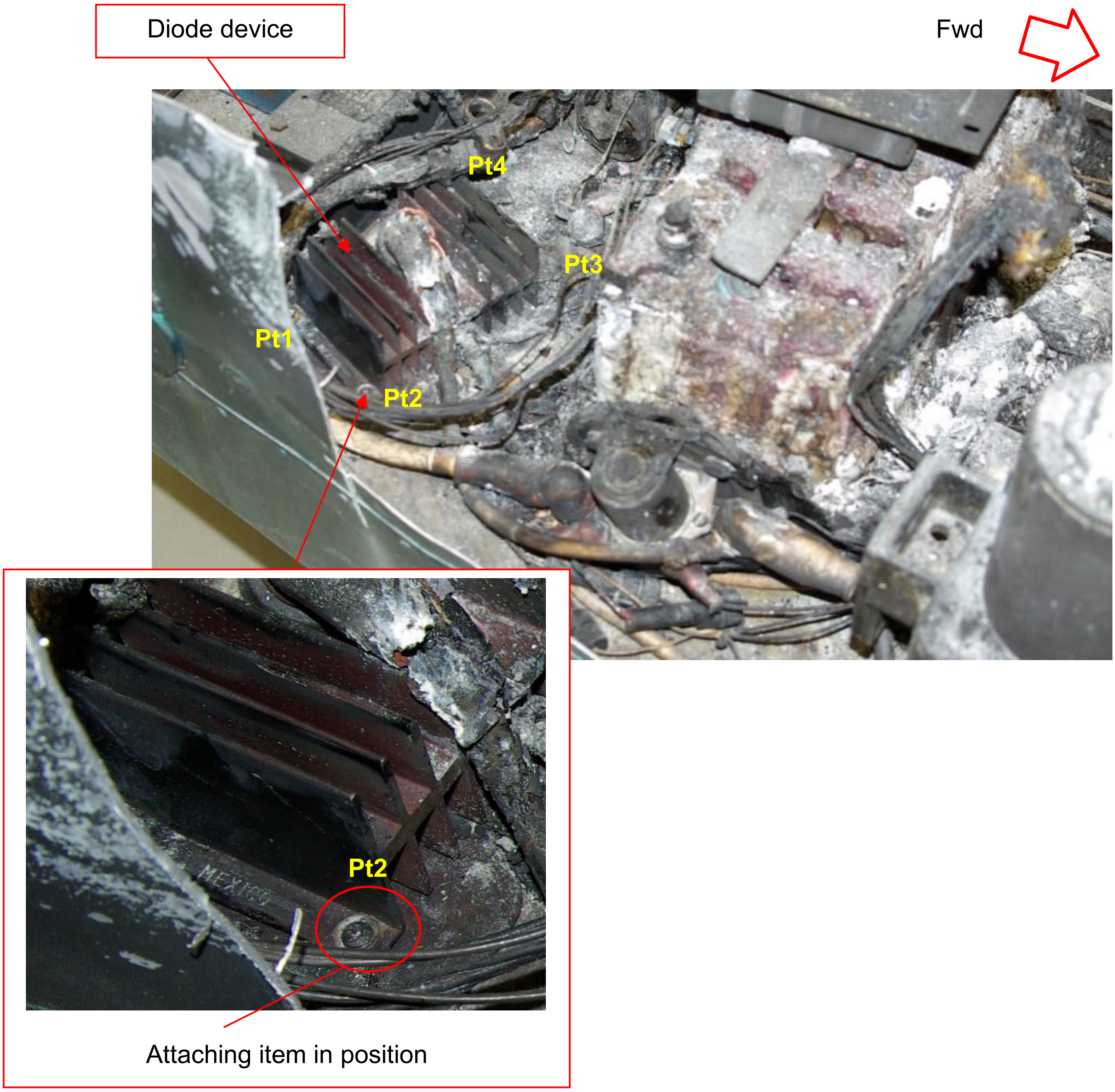


Figure 98: Diode device in position on aeroplane floor
Source: BEA

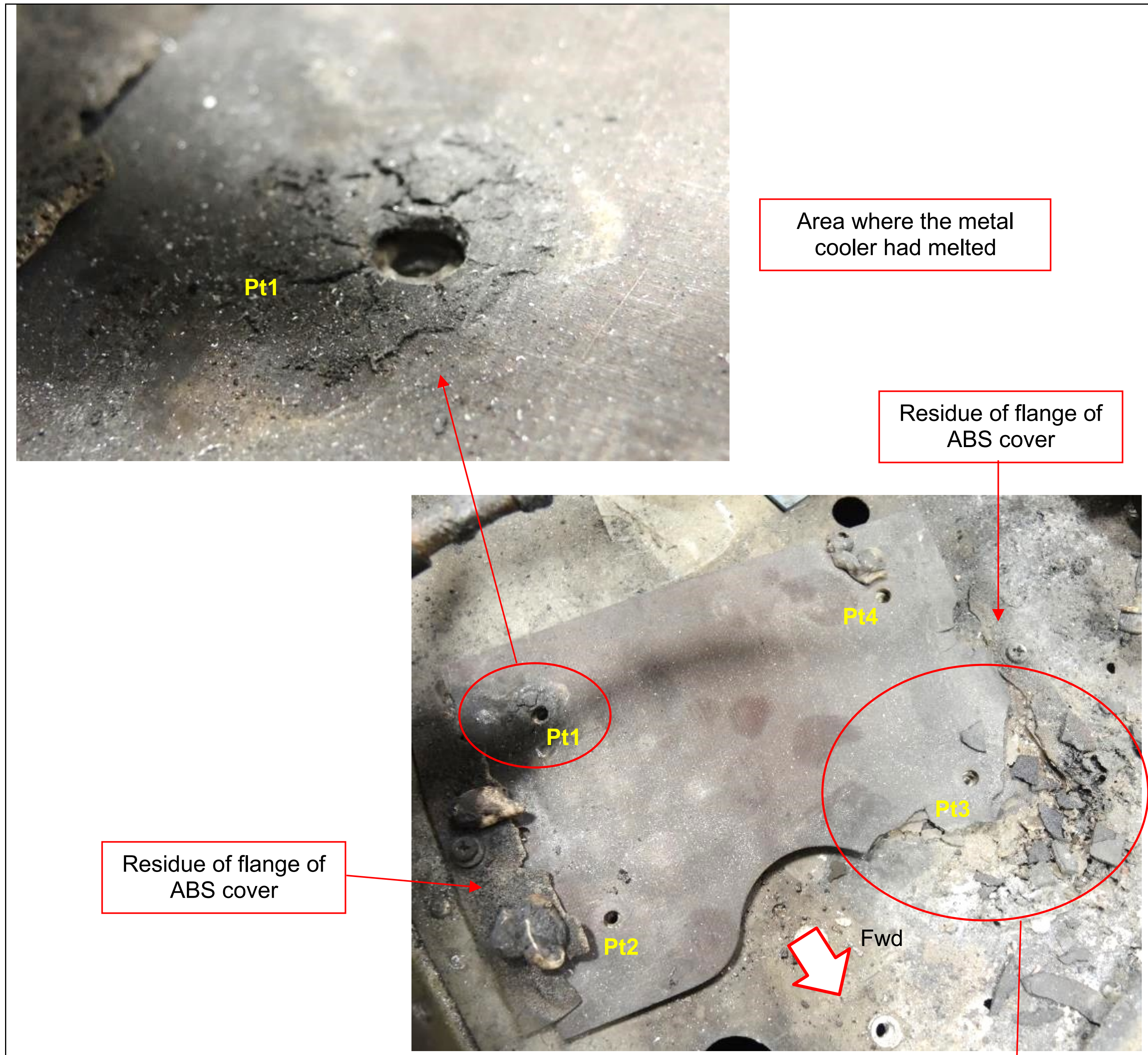


Figure 99: Phenolic shield in position on aeroplane floor
Source: BEA

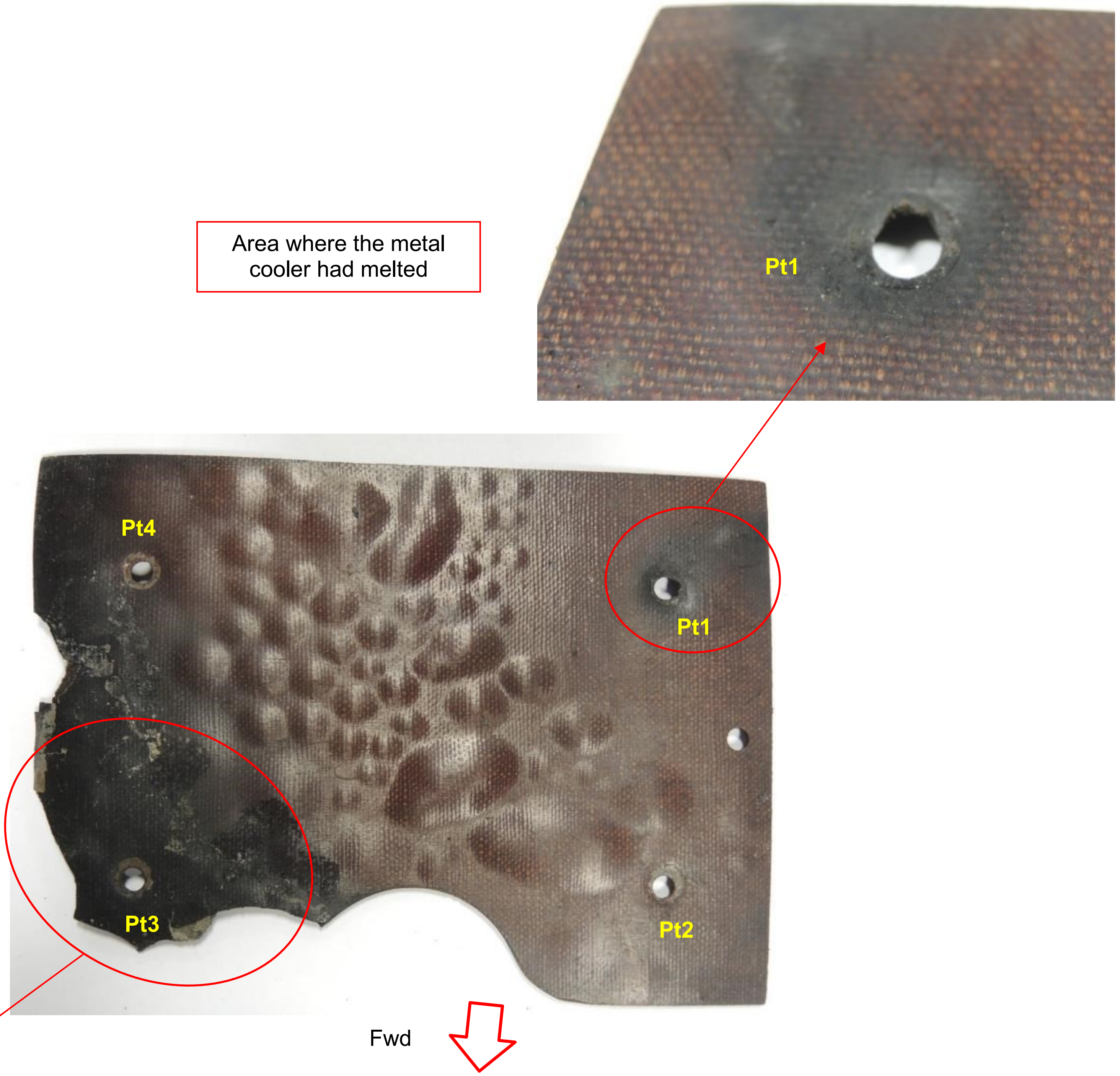


Figure 100: Phenolic shield – face of shield in contact with aeroplane floor
Source: BEA

Area closest to + terminal of battery – most damaged area of shield

<u>Reminder:</u> the points where the diode device is attached to the aeroplane floor are arbitrarily numbered Pt1, Pt2, Pt3 and Pt4 to facilitate comprehension.

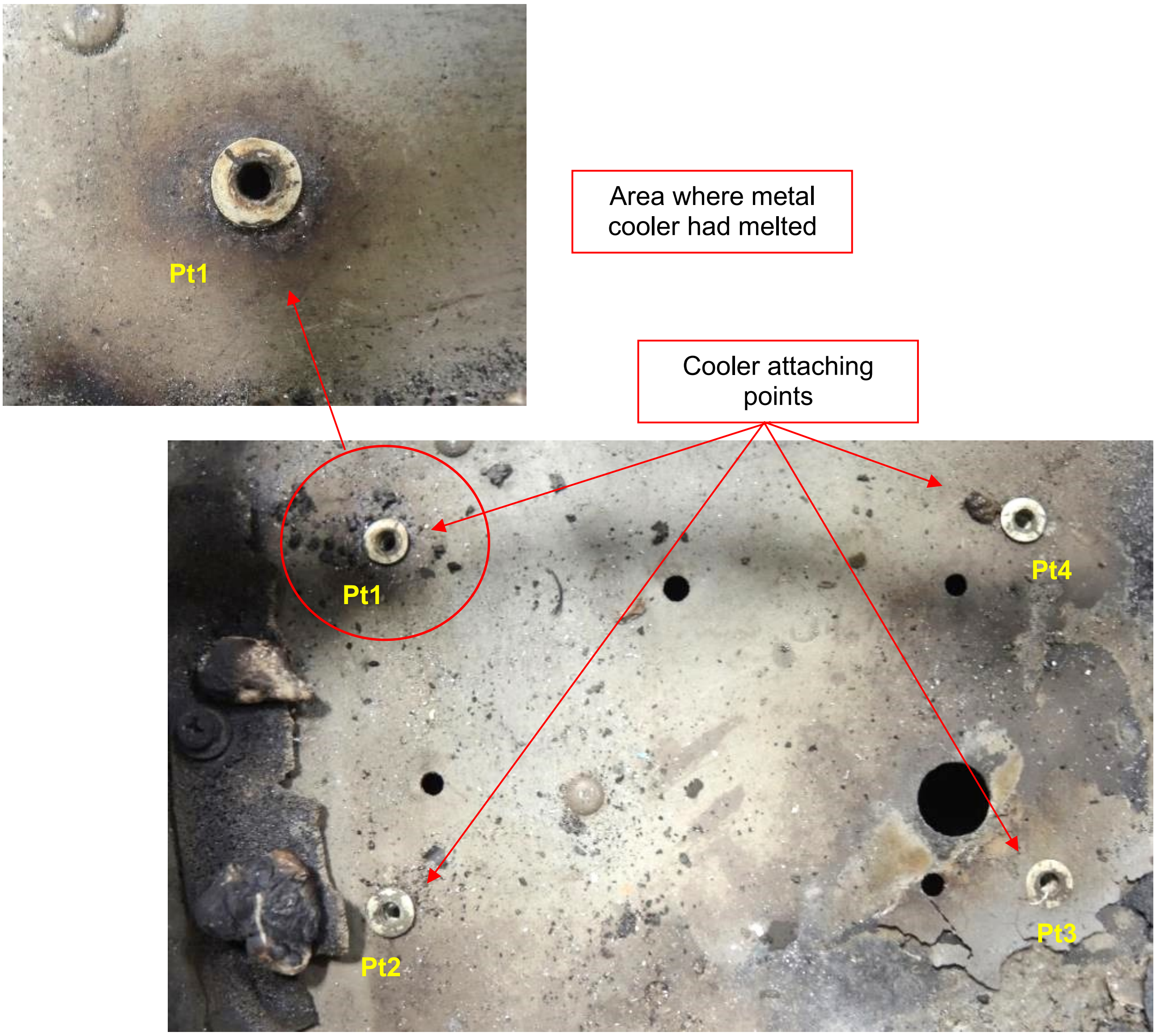


Figure 101: Aeroplane floor under diode device
Source: BEA

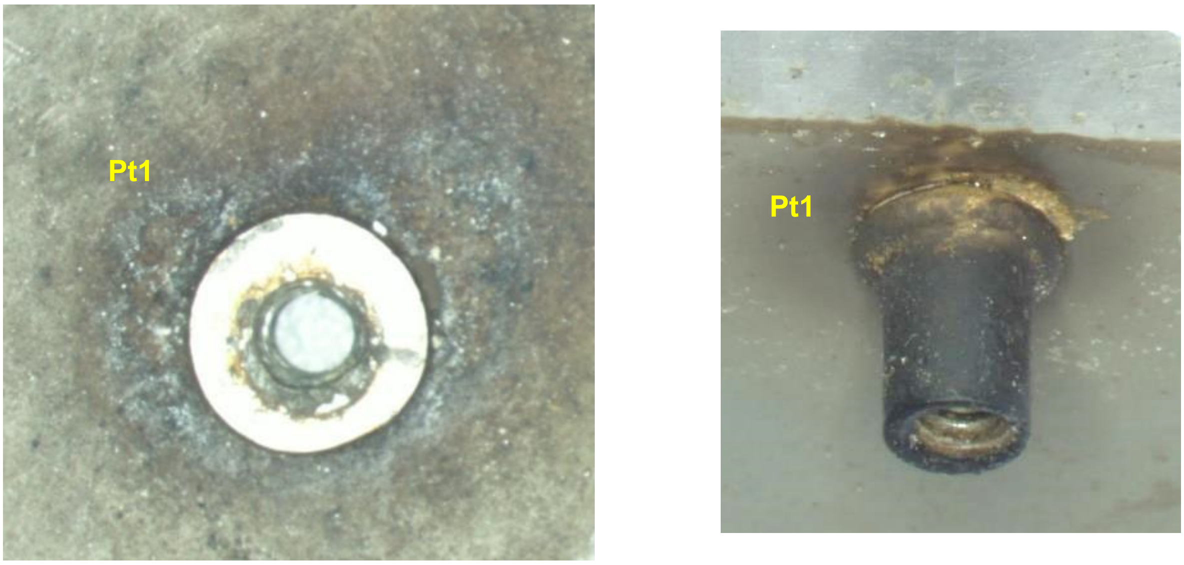


Figure 102: Attaching point on aeroplane floor in line with melted area of metal cooler
Source: BEA

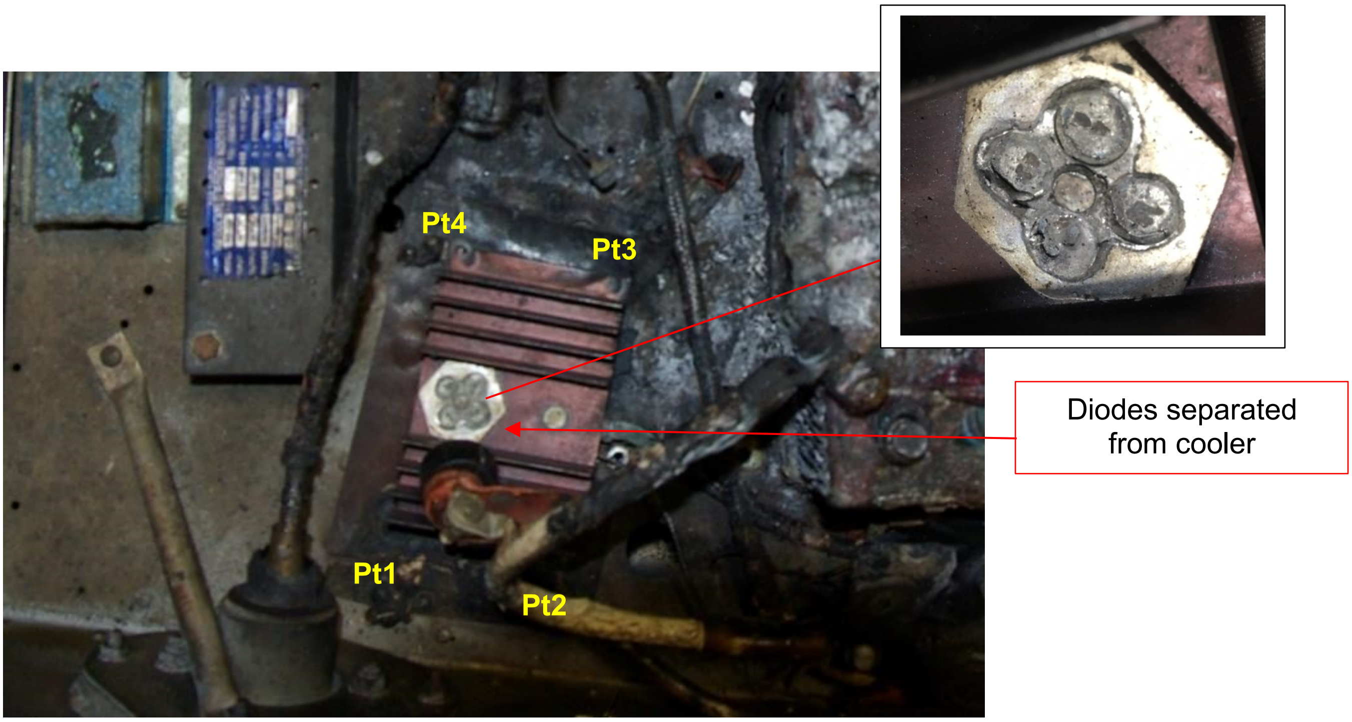


Figure 103: Diodes and its cooler
Source: BEA

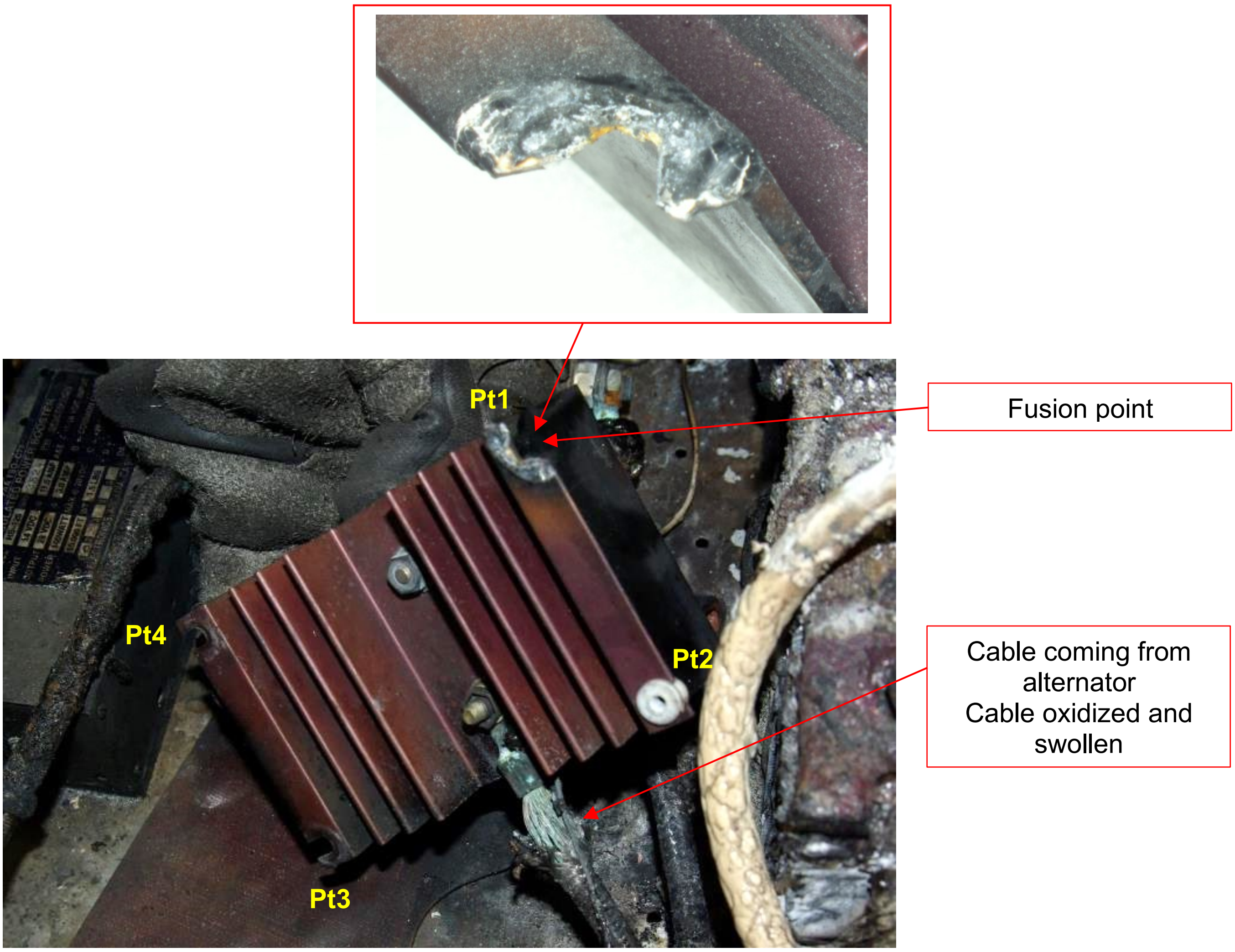


Figure 104: Cable coming from alternator attached to cooler
Source: BEA

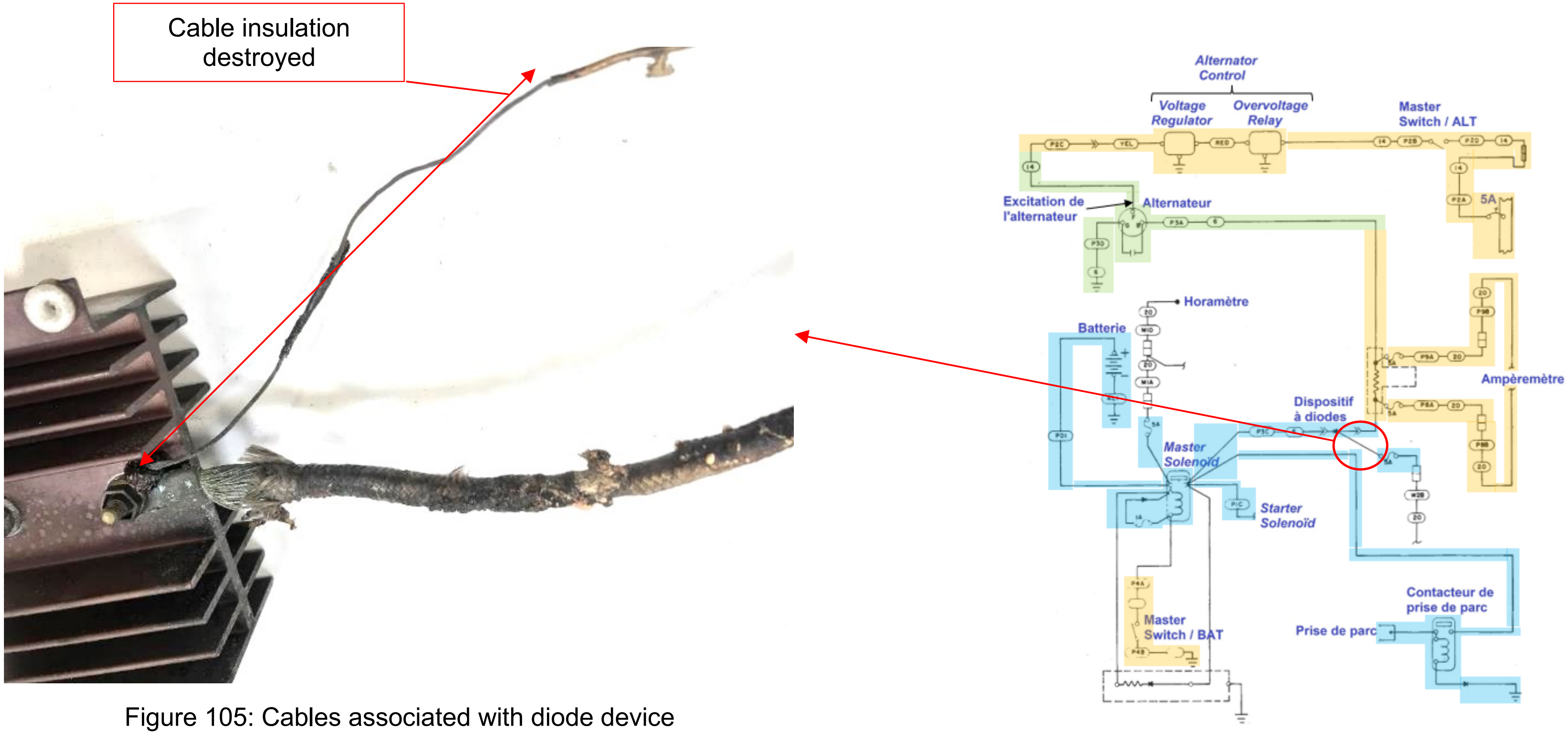


Figure 105: Cables associated with diode device
Source: BEA

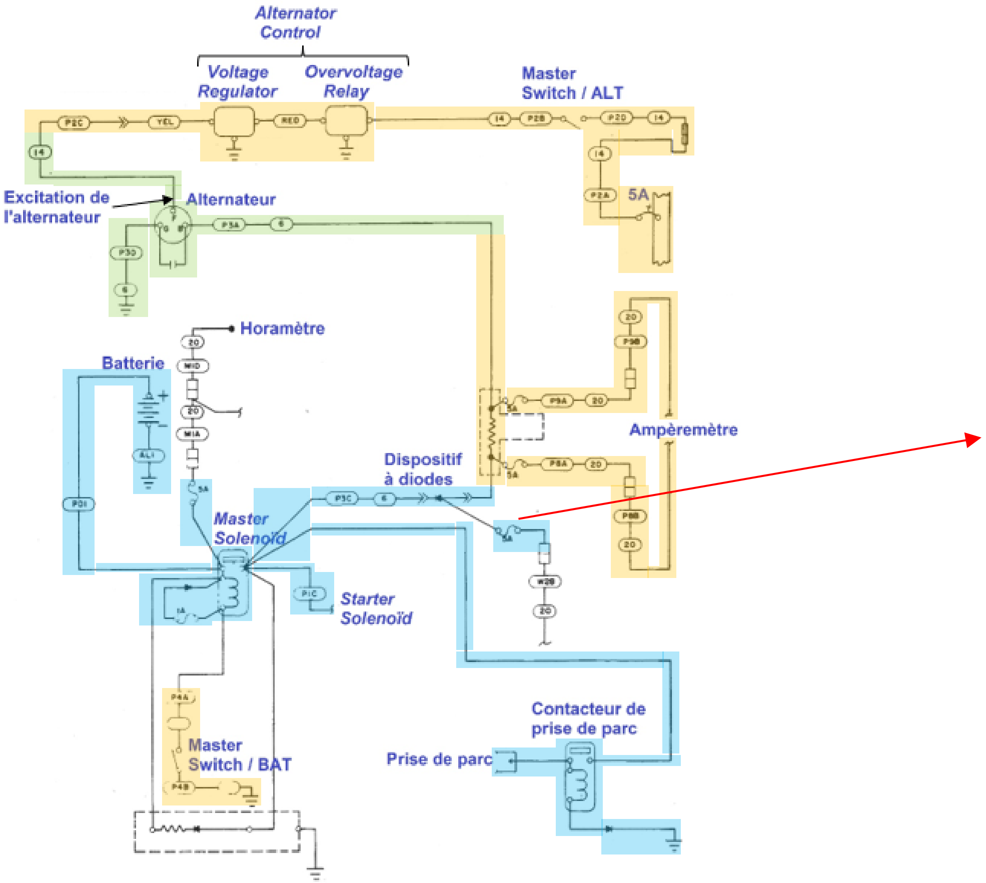


Figure 106: 5A fuse associated with diode device

Source: BEA

The corner of the metal cooler with thermal effect damage (in line with Pt1) was examined in the BEA laboratory.

Figure 107: metal cooler and detail view of damaged area

Source: BEA

Sliced samples of the damaged area were taken, coated and then polished in successive stages in order to carry out analyses of the chemical composition in the melted area using energy dispersive spectrometry (EDS).

Beforehand, an EDS analysis was carried out on a similar screw and washer, normally in place on the fixing lug of the metal cooler.

The screw was found to be made of galvanised unalloyed steel (Fe in the core and Zn on the surface). The washer showed a fluorine (F) peak (**Figure 108**).

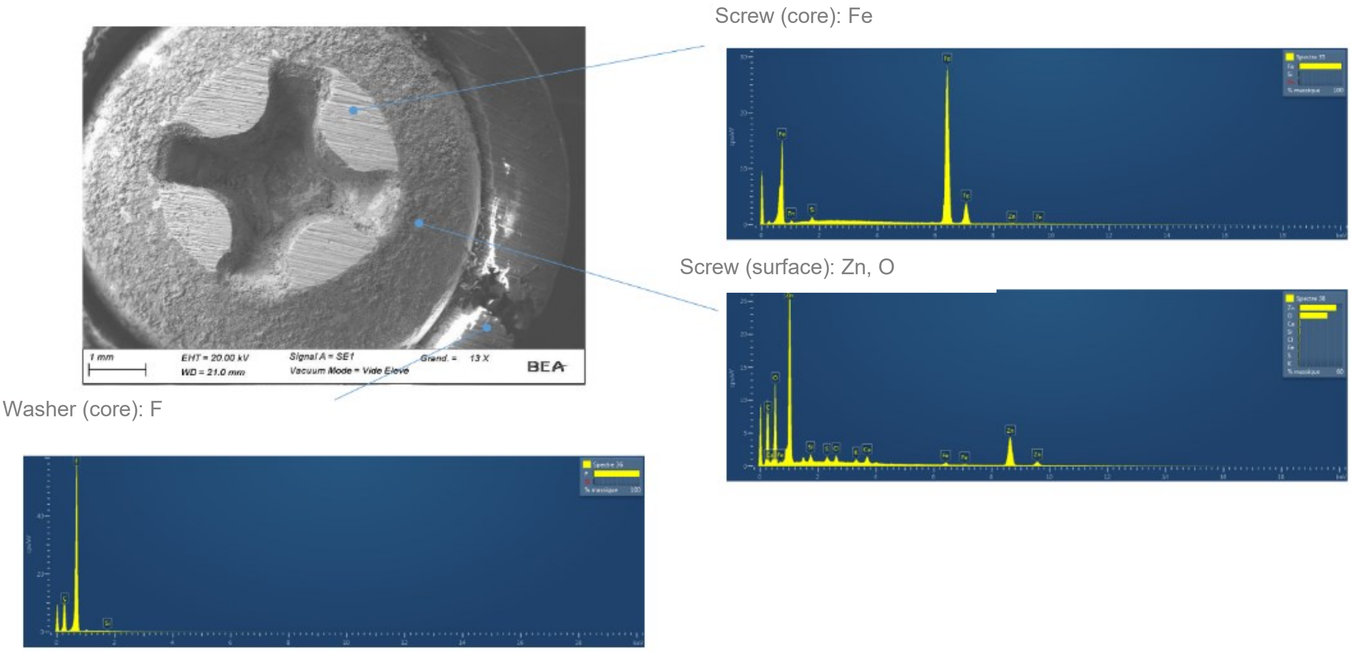


Figure 108: EDS analysis of materials making up screw and washer

Source: BEA

The EDS analysis of the successive sections found the presence of iron (Fe) and fluorine (F) in significant quantities in the melted area, as illustrated in **Figure 109**. The aluminium (Al) is the material making up the cooler.

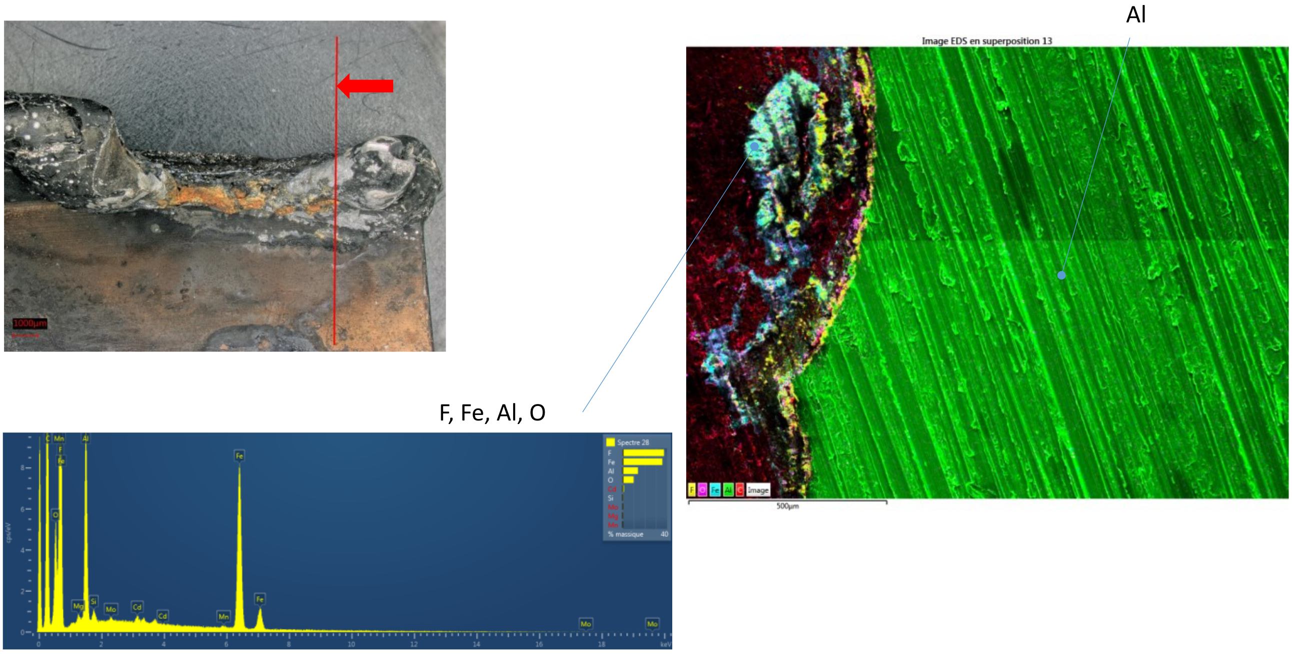


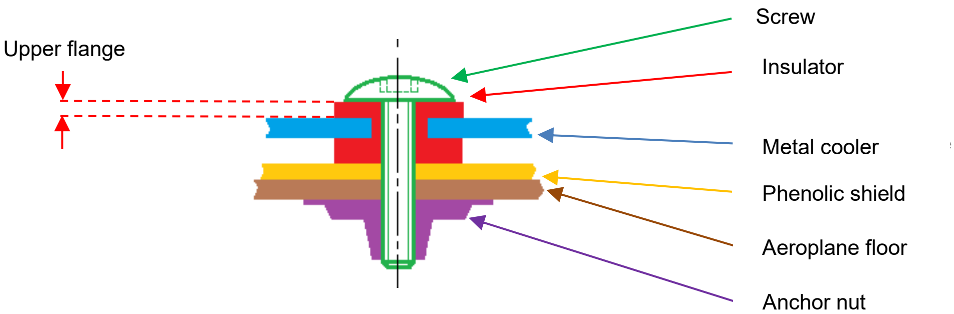
Figure 109: EDS analysis of section of melted area – presence of Fe and F on surface
Source: BEA

The steel screw and washer composed of fluorine were therefore present when the overheating occurred, leading to them melting.

2.4.5.3 – Observation of insulating washers of a diode device installed on an identical aeroplane

The Piper PA-28RT-201T registered HB-PPB is maintained by the same workshop as the aeroplane involved in the accident. Its electrical system is also similar to the accident aeroplane.

The diode device of this aeroplane was removed and the insulating washers examined.
The upper flange of two of the four washers was cracked (**Figure 111** and **Figure 112**).



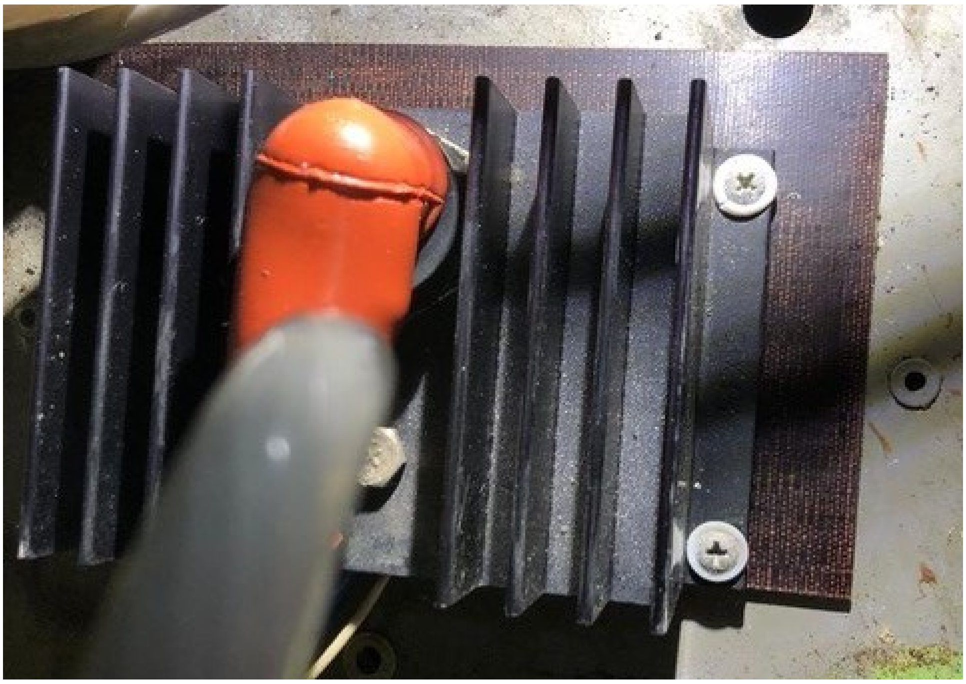


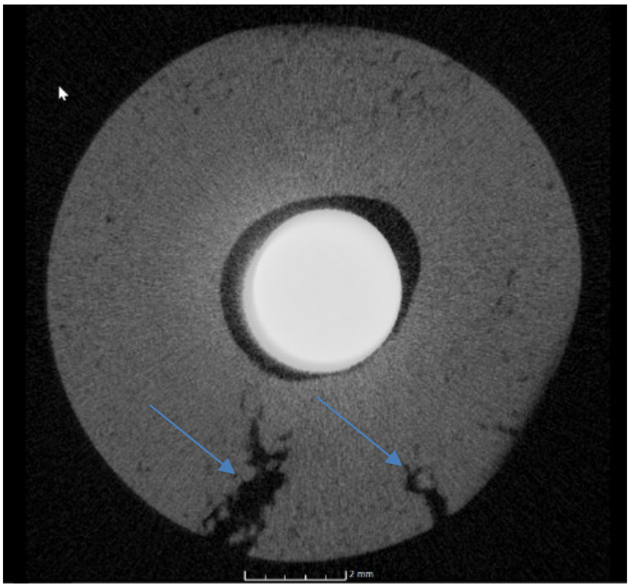
Figure 110: Diode device installed on aeroplane floor (photo taken before removal of ABS cover)
Source: maintenance workshop for HB-PPB

Figure 111: Cracked upper flange of insulating washer
Source: maintenance workshop for HB-PPB

Figure 112: Cracked upper flange of insulating washer
Source: maintenance workshop for HB-PPB

These visual findings were supplemented by tomographic observations.

The figure below shows one of the washers described above, permanent deformations and cracks can be seen (**Figure 113**).



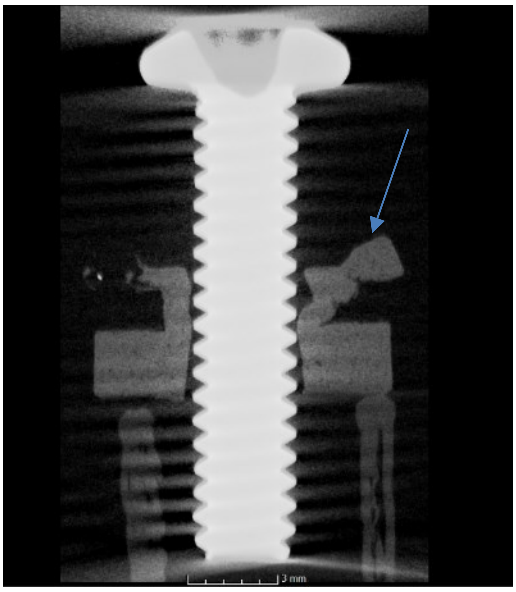


Figure 113: Example of screw and washer observed – the washer shows cracks (LH side) and plastic deformations (RH side)
Source: BEA

The two figures below show a tomographic view of an assembly that can be considered "nominal" (only the phenolic shield is not in place) (**Figure 114**) and an assembly with a damaged insulating washer (**Figure 115**).

With the "nominal" assembly (**Figure 114**), it can be seen that the metal cooler is correctly insulated from the attaching screw.

In the other assembly (**Figure 115**), it can be seen that the insulating thickness between the screw head and the metal cooler is clearly reduced, but there is no mechanical contact between the two metal parts.

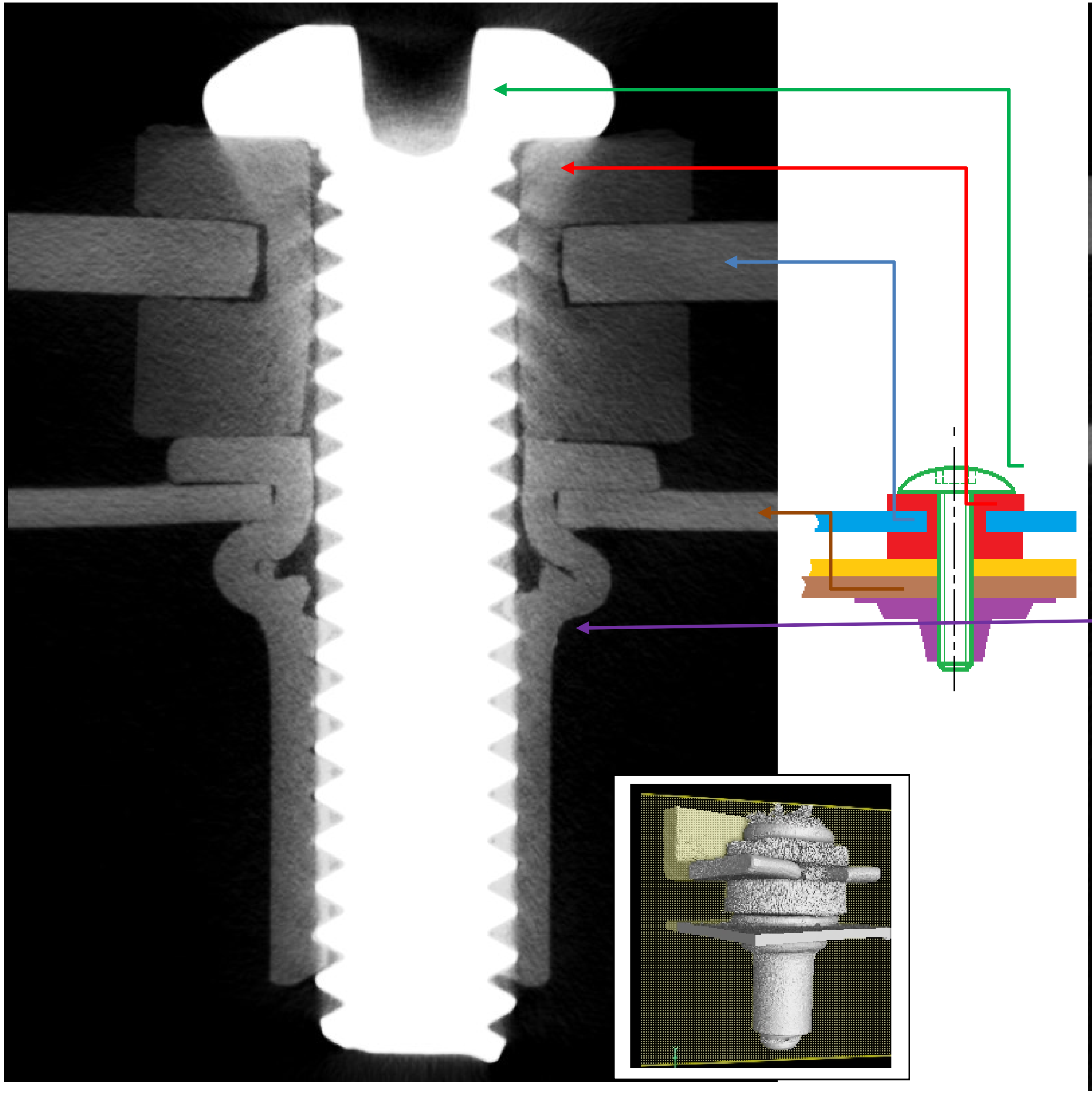


Figure 114: Nominal assembly
Source: BEA

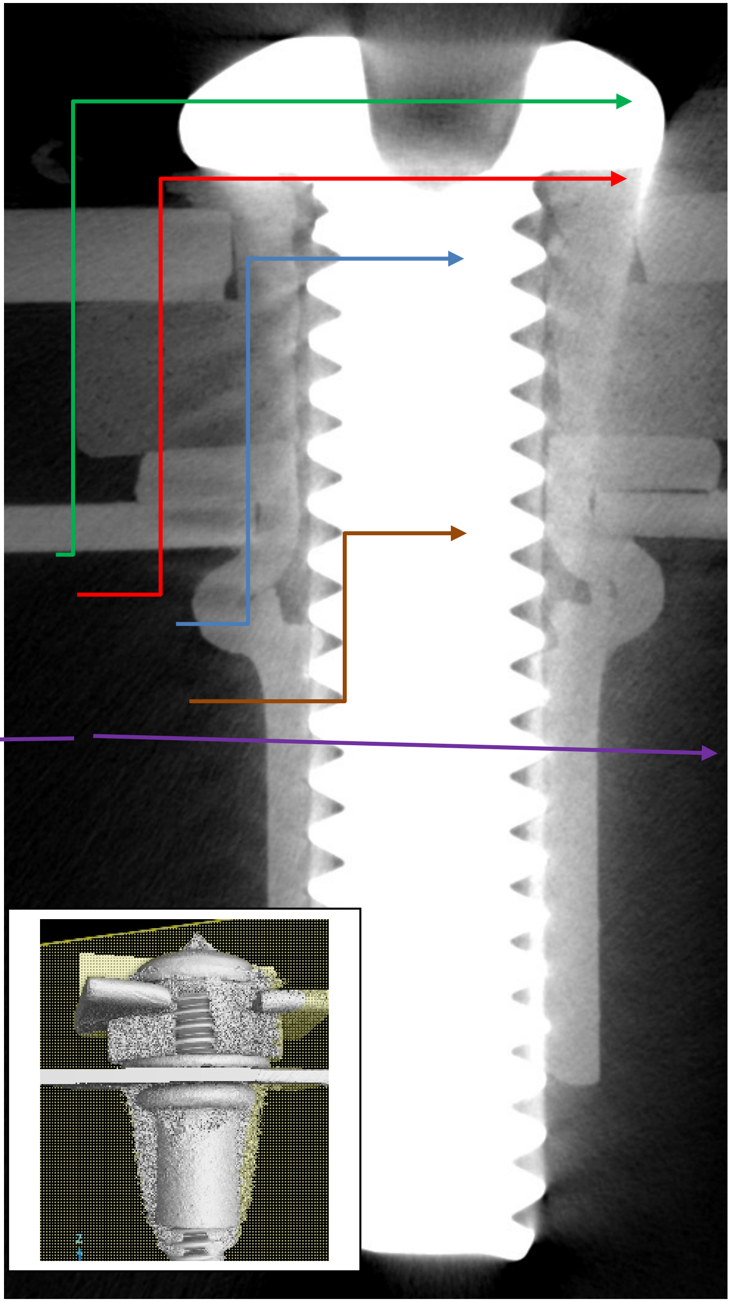


Figure 115: Assembly with a damaged insulating washer
Source: BEA

2.4.5.3 – Detailed description of diodes and associated conductors

A high-magnification visual examination of the separation area was carried out in the BEA laboratory. The surfaces normally in contact had areas where the solder was absent. These areas showed exposed base material. The separation of the diode was very probably the result of a rise in temperature of the solder leading to the separation of the two soldered surfaces and locally exposing the base material.

The conductor leading to the master solenoid had signs of overheating, in particular at the insulation (**Figure 116**).

Figure 116: diode and its conductor
Source: BEA

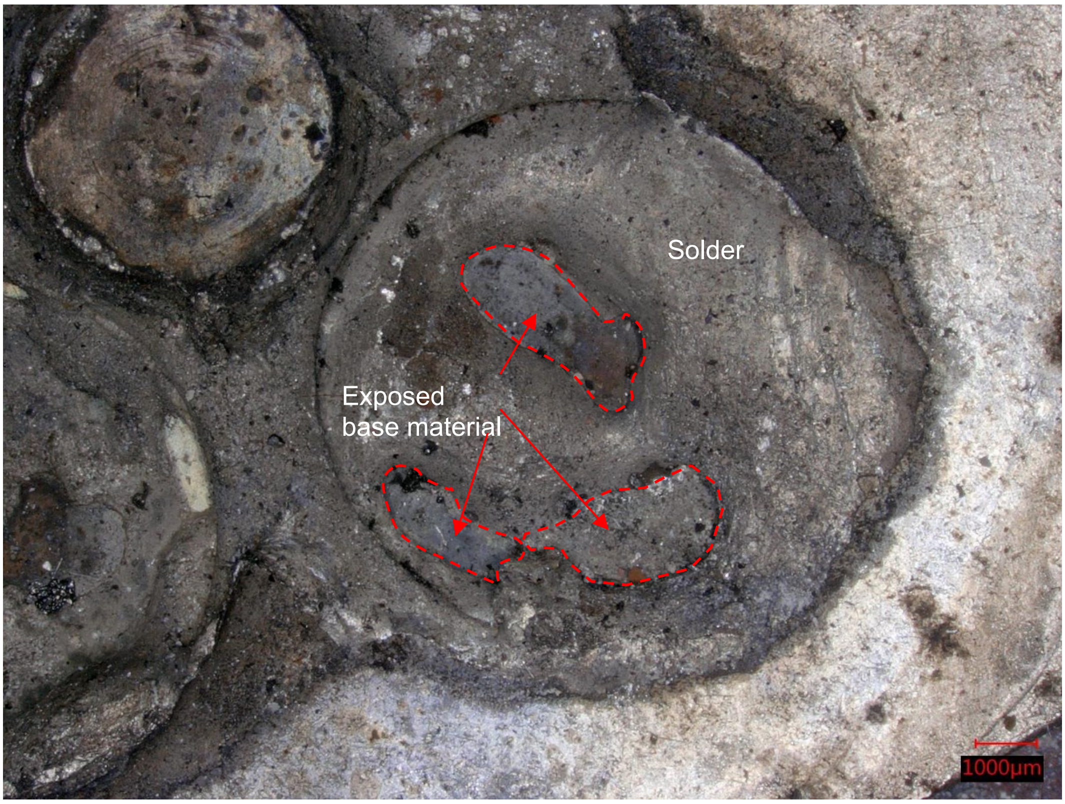


Figure 117: high-magnification photo of diode solder
Presence of areas without solder
Source: BEA

An X-ray examination was carried out (**Figure 118**). It showed:

- the presence of each cell;
- the visually undamaged content of each cell.

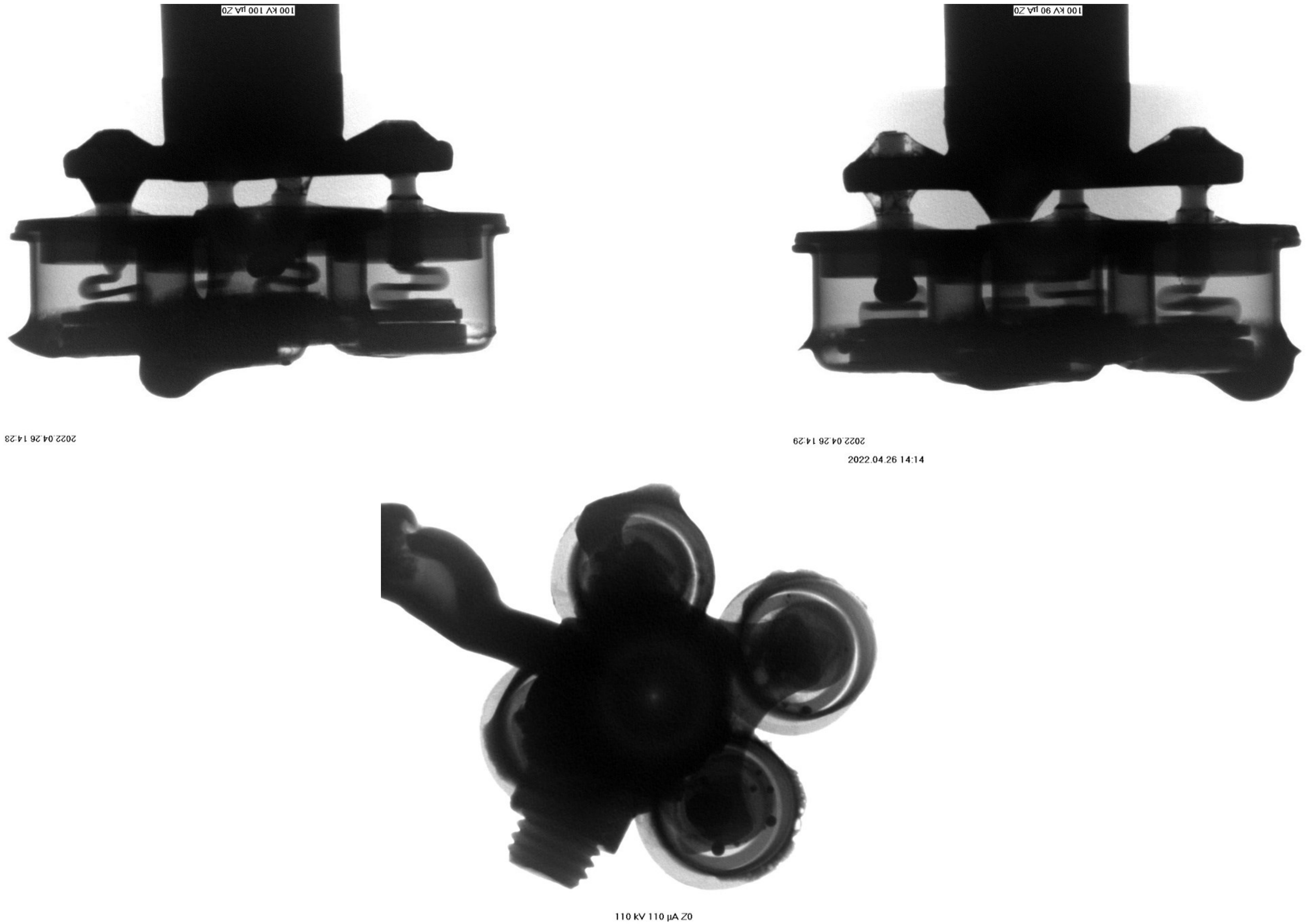


Figure 118: X-ray observations of diode device
Source: BEA

2.4.5.5 – Electrical test on diode device of accident aeroplane

Two types of test were carried out:

- **Test 1:**

  The purpose of this test was to measure the voltage drop across the diode when it is forward biased and then when it is reverse biased. This test was carried out using a digital multimeter, in "diode test" mode. The diode test mode of a multimeter produces a small voltage across the measurement leads. The multimeter then displays the voltage drop when the measurement leads are connected to a forward biased diode.

- **Test 2:**

  The purpose of this test was to measure the forward biased and then reverse biased resistance. This test was also carried out using a digital multimeter, in "resistance" mode.

In order to test each cell individually, the residues of melted and resolidified solder between the cells were removed.

The cells were arbitrarily identified 1 to 4. The results were as follows:

	Voltage drop Forward bias	Voltage drop Reverse bias	Resistance Forward bias	Resistance Reverse bias
Cell 1	0 V	0 V	0 Ω	0 Ω
Cell 2	0 V	0 V	0 Ω	0 Ω
Cell 3	0.47 V	1.7 V	4 kΩ	5.5 kΩ
Cell 4	0 V	0 V	0 Ω	0 Ω

These values indicate a short-circuit of cells 1, 2 and 4. Cell 3 had different values but these do not correspond to the expected values for a functional diode. This latter also seemed to be damaged.

Note: Generally, a diode will short circuit when the current through it exceeds the maximum value that can be borne by this component.

In conclusion, the four identical cells making up the device were all damaged and not functional.

2.4.5.6 – Tests on undamaged diode devices by Concorde Aircraft Batteries

Four tests were carried out by Concorde Aircraft Batteries on diode devices of the same type as that on HB-PNP. The BEA was not present during these tests.

Tests carried out and results obtained:

Test 1:

Content and conditions:
This test consisted of checking that the diode device was capable of bearing an amperage of around 70 A for 30 min. During this test, the current was 77 A (maximum value of alternator output with an overcurrent of 10 %).
During this test, the temperature was measured using an infrared thermometer.

Findings:
The diode device body had reached a temperature of 140°C at the end of the test.
The metal cooler had reached a temperature of 106°C where it was close to the diode device and 101°C in line with the attaching points to the aeroplane structure.
After the test, the diode device was still functional.

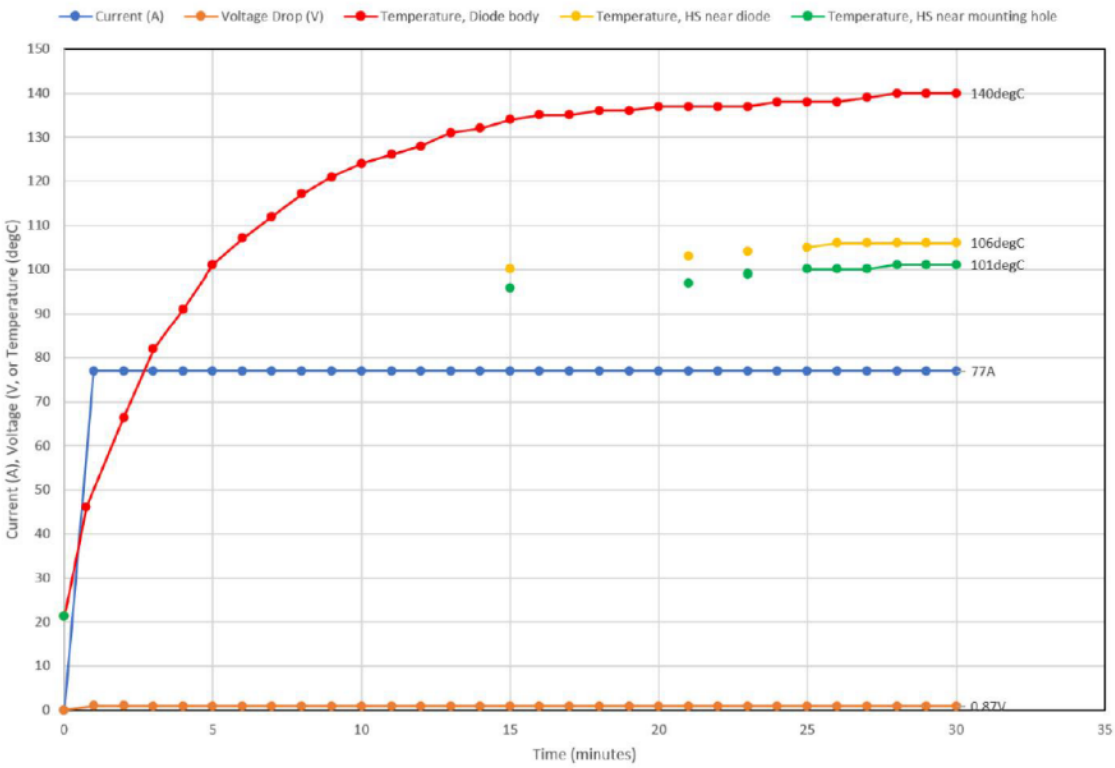


Figure 119: Evolution of parameters during test 1
Source: Concorde Aircraft Batteries

Test 2:

Content and conditions:
This test consisted of checking that the diode device was capable of bearing an amperage of around 70 A for 60 min. During this test, the current was 77 A (maximum value of alternator output with an overcurrent of 10 %).
During this test, the temperature was measured using an infrared thermometer.

Findings:
After the test, the diode device was still functional.

Test 3:

Content and conditions:
This test consisted of simulating a short-circuit between the metal cooler and the structure of the aeroplane. The insulating washer between the metal cooler and an attaching screw was removed and replaced by a slightly bent metal washer (**Figure 120**). The aim was for the bent washer to penetrate the oxide coating which protects the outer surface of the metal cooler. The test was repeated three times.
For the duration of two tests, the current was 77 A (nominal value of alternator with an overcurrent of 10%). For the duration of one test, the current was 70 A.
For a fixed current, the voltage drop in line with the ground fault was measured. Concorde Aircraft Batteries deduced the dissipated power (W = U.I) from this.

During each test, the technician regularly turned the attaching screw assembled with the bent metal washer.

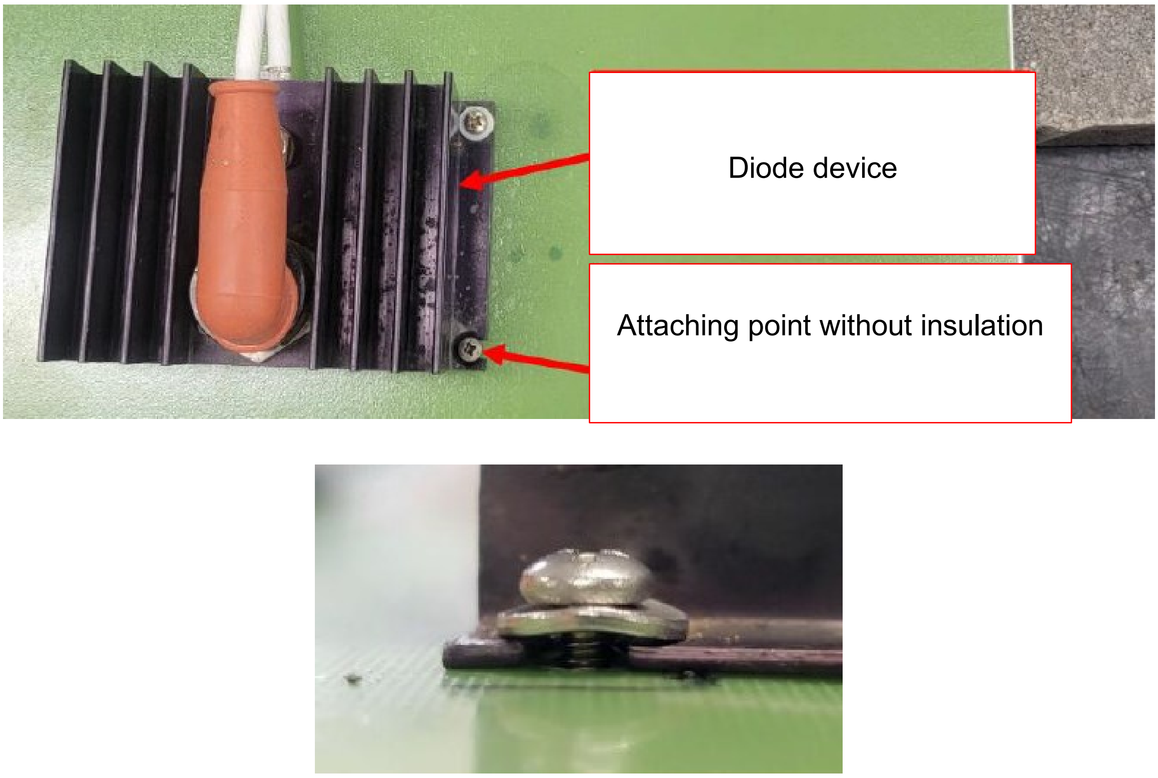


Figure 120: Installation for test 3
Source: Concorde Aircraft Batteries

Findings:
The findings are summarised and illustrated in the table below.

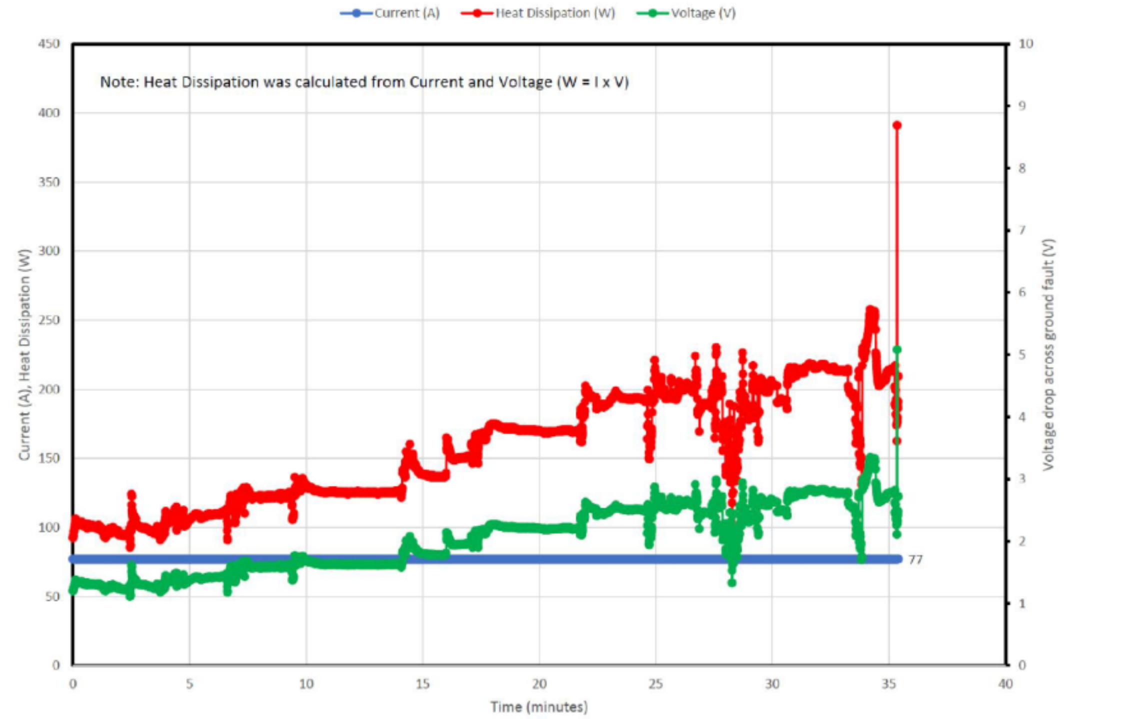
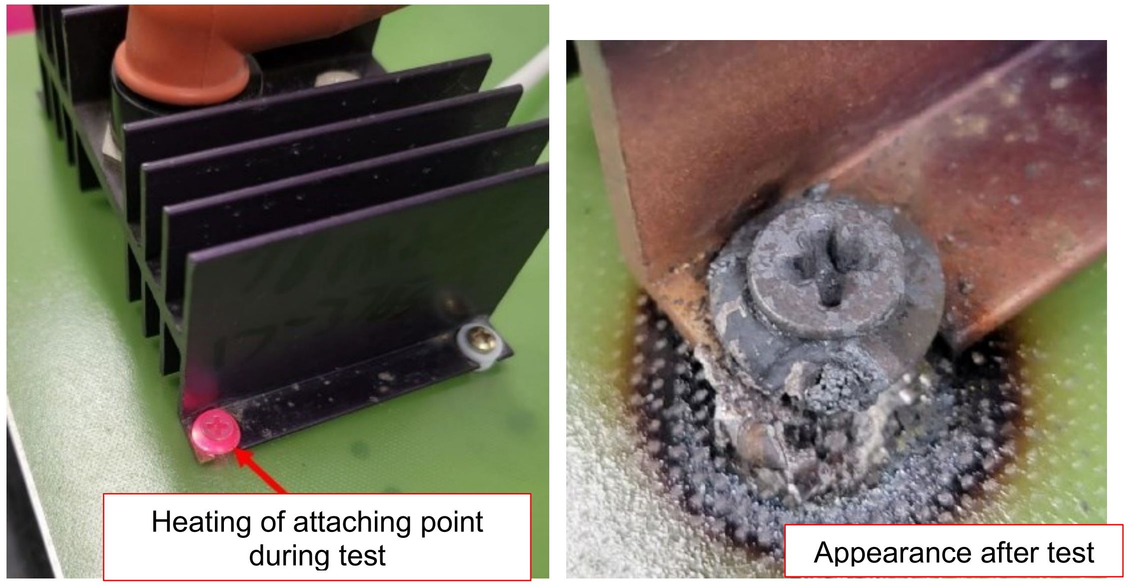
<table>
<tr>
<td>Test 3-1</td>
<td>The voltage drop in line with the ground fault increased during the test, partly due to the screw being turned by the technician.
Thermal dissipation was at its highest at the end of the test. The aluminium alloy of the metal cooler had melted in line with the attaching point. This melting occurred after a test time of approx. 35 min.</td>
<td>
</td>
<td>

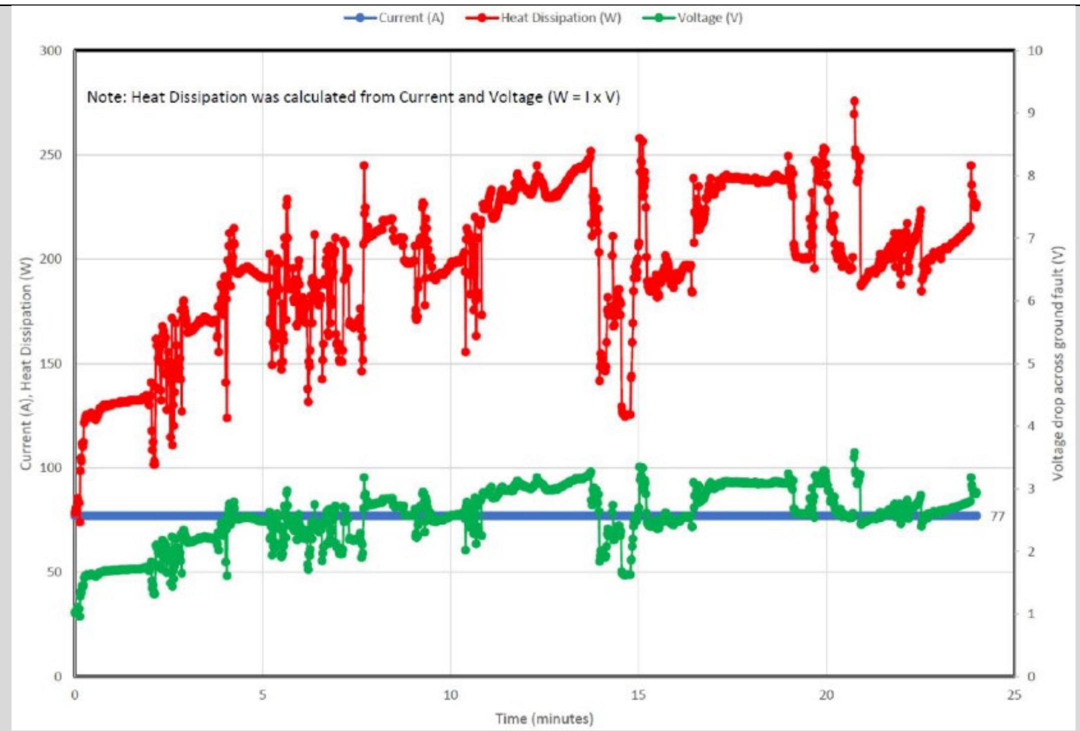
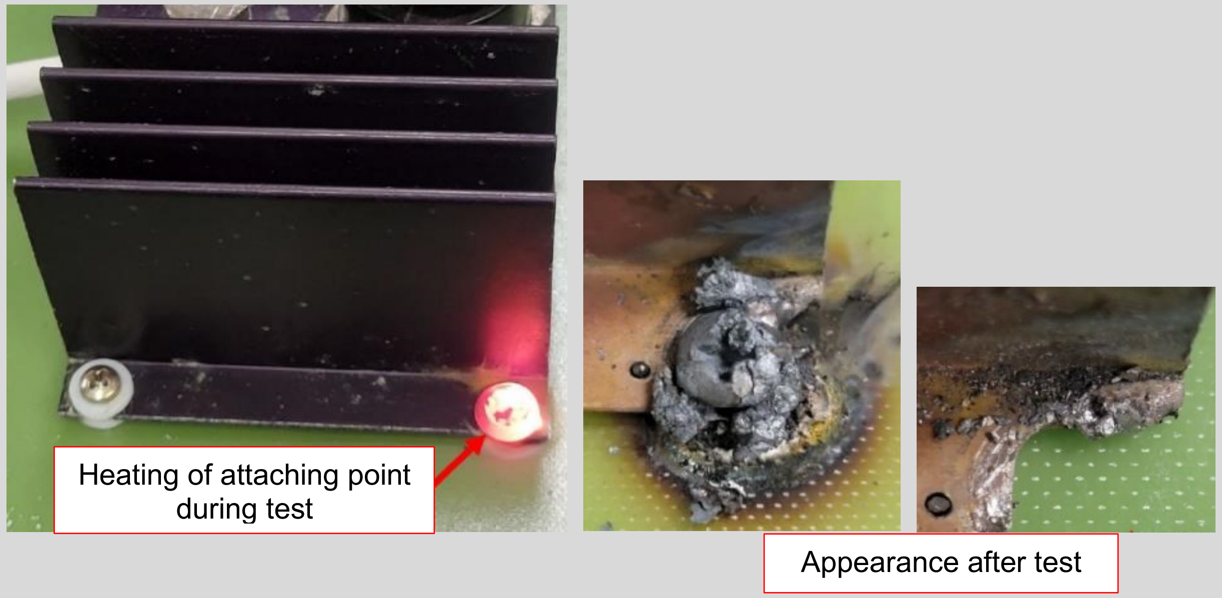
Figure 121: Installation during test
Source: Concorde Aircraft Batteries</td>
</tr>
<tr>
<td>Test 3-2</td>
<td>The voltage drop in line with the ground fault increased during the test, partly due to the screw being turned by the technician.
Thermal dissipation was at its highest at the end of the test. The aluminium alloy of the metal cooler had melted in line with the attaching point. This melting occurred after a test time of approx. 25 min.</td>
<td>
</td>
<td>

Figure 122: Installation during test
Source: Concorde Aircraft Batteries</td>
</tr>
<tr>
<td>Test 3-3</td>
<td colspan="3">The findings were identical to test 3-1. However, the lower part of the metal cooler, at the attaching point melted after a test time of around 9 min.
Figure 123: Installation during test (*Source: Concorde Aircraft Batteries*)</td>
</tr>
</table>

Test 4:

Content and conditions:

This test consisted of reproducing test 3 with the ABS cover in position. The cover was positioned when the colour of the attaching point indicated intense heat (**Figure 124**).

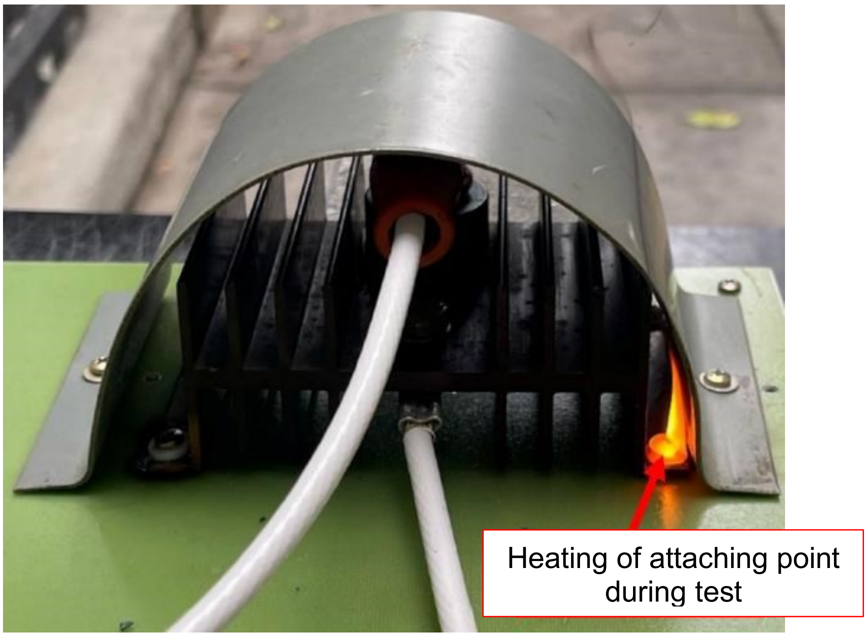


Figure 124: ABS cover in position with attaching point subject to intense heat
Source: Concorde Aircraft Batteries

Findings:

The ABS cover burst into flames around 20 s after it had been put in position. The cover was destroyed after 13 min (**Figure 125** and **Figure 126**).

Figure 125: Burning ABS cover
Source: Concorde Aircraft Batteries

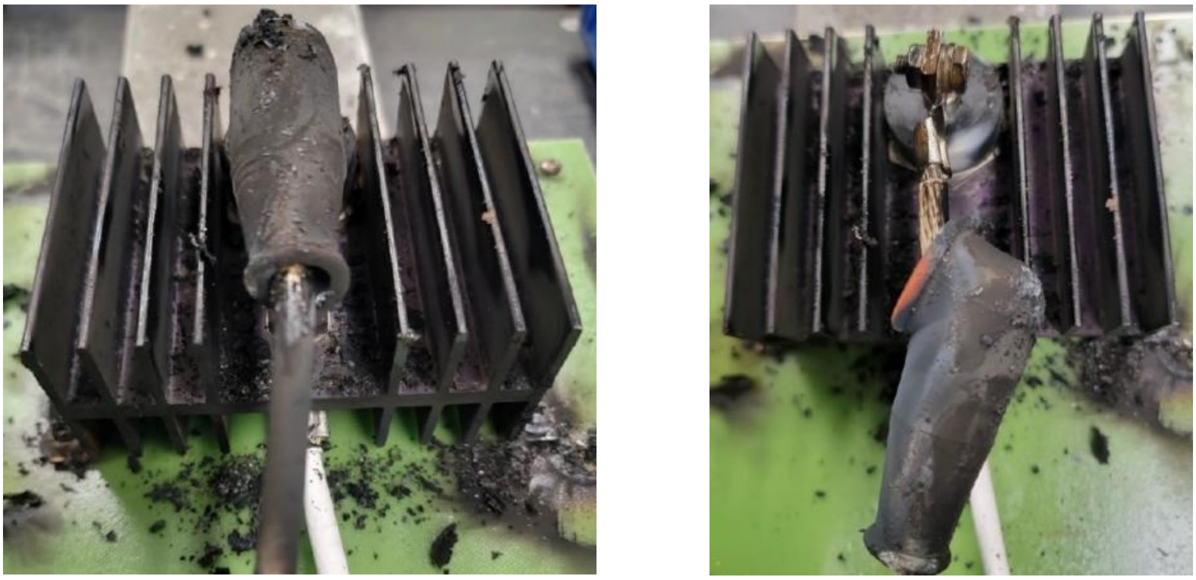


Figure 126: Condition of diode device after test
Source: Concorde Aircraft Batteries

Test 5:

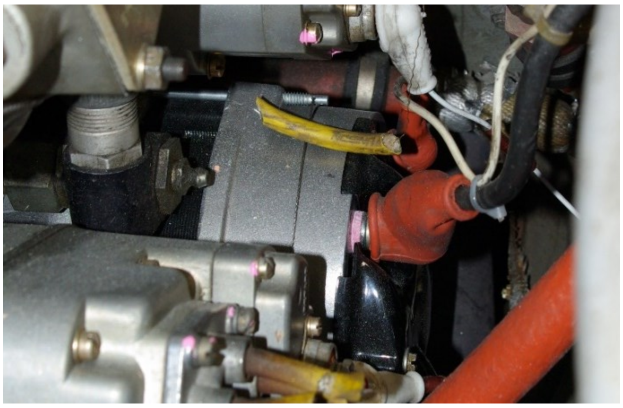
Content and conditions:
This test consisted of exposing the diode device of test 4 to a hydrogen/oxygen flame.

Findings:
The diode device separated from the metal cooler after around 8 min of exposure to the flame (**Figure 127**). This separation was identical to that observed on HB-PNP.
After this separation, the diodes were in a short-circuit as was the case for the diodes of HB-PNP.

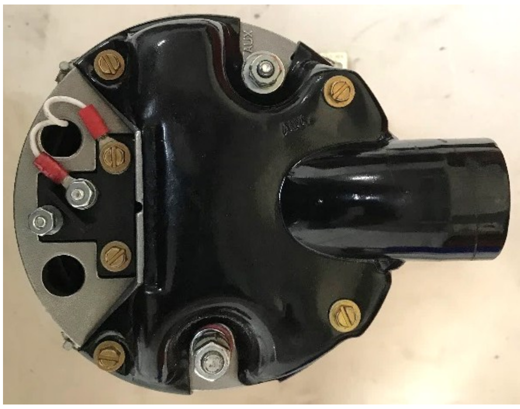
Figure 127: Separation of diode bridge from metal cooler
Source: Concorde Aircraft Batteries

2.4.6. Power system components located at front of aircraft

The components in the engine compartment (see **Figure 39**) were in position. The conductors were attached and undamaged (**Figure 128**).



Conductors in position on the alternator



Alternator removed

Figure 128: Alternator
Source: BEA

The components on or behind the instrument panel (see **Figure 39**) were in position, some were covered with soot. The conductors were not damaged, only the outer surface of their insulation was sometimes blackened due to contact with the smoke (**Figure 129**, **Figure 130** and **Figure 131**).

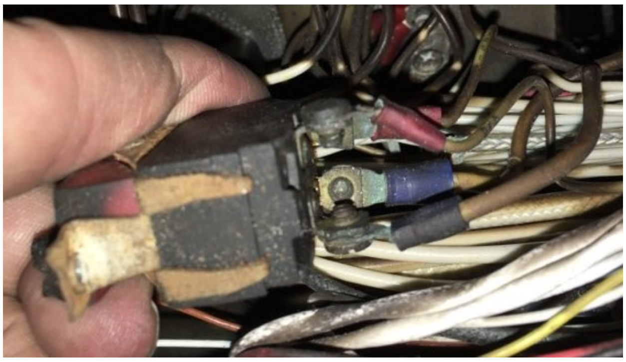
Conductors connected to the ammeter, located behind the instrument panel

Rear side of ammeter

Figure 129: Ammeter
Source: BEA

Instrument panel control – master switch

Conductors connected to master switch, located behind instrument panel



Conductors connected to master switch, located behind instrument panel

Figure 130: Master switch (BAT/ALT)
Source: BEA

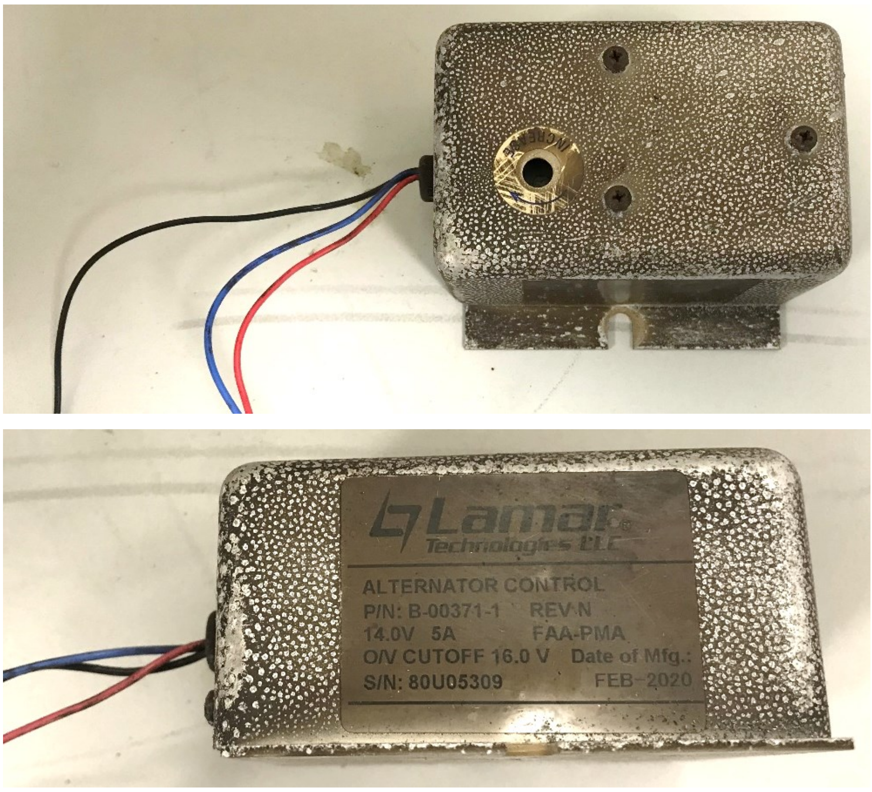
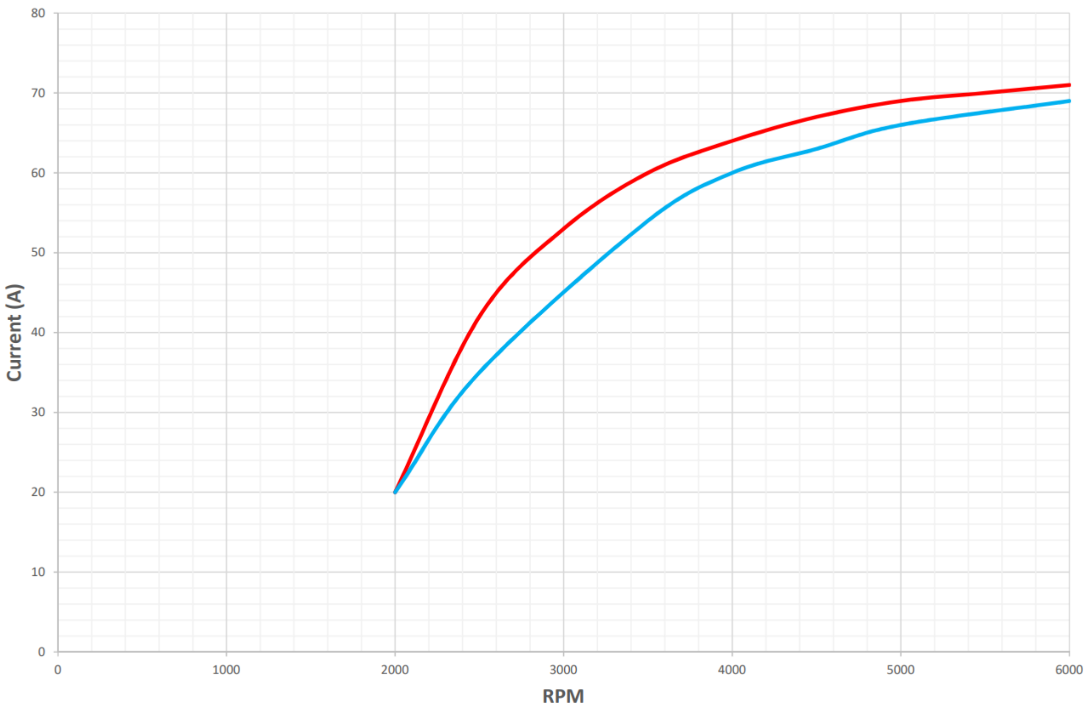


Figure 131: Alternator control
Source: BEA

2.4.7. Alternator

The alternator was tested at the ENAC at Castelnaudary, in the presence of the BEA. This test consisted of applying a nominal voltage to the terminal of the inductor winding in order to produce the highest possible output current at a given speed. The test bench automatically varied the speed from a few hundred to a few thousand revolutions per minute. As the alternator output voltage attempted to rise, the test bench added more load (increased current draw) to maintain the target voltage setpoint, which in the case of a 12 V alternator is 14 V DC. The data was collected automatically during the test and was used to produce an output curve of current versus rotor speed.

The curve obtained during the test is shown in **Figure 132** below.



Key

(red)	Measurements made as rpm increased
(blue)	Measurements made as rpm decreased

Figure 132: Measurements made during alternator test
Source: BEA

According to the alternator manufacturer, the data measured was normal. The manufacturer also specified that this test was sufficient for defining the functionality of the alternator.

2.4.8. Alternator control

2.4.8.1 – Examination

Before being tested, an X-ray examination of the alternator control was carried out by making detailed observations of small area by small area. A reconstructed view of the equipment made up of the various detailed images is shown in **Figure 133**. This examination did not reveal any internal damage.

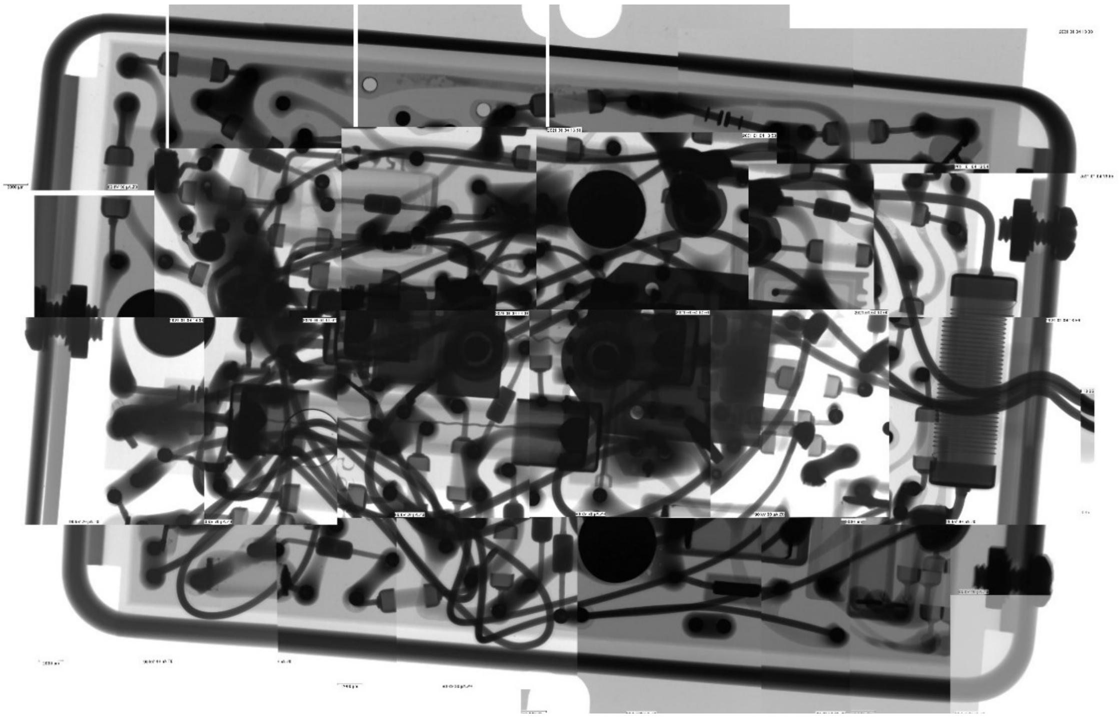


Figure 133: Reconstructed view of alternator control
Source: BEA

2.4.8.2 – Test

The alternator control was tested according to the specifications of the manufacturer, Lamar Technologies. The test was performed as follows:

- Step 1: An adjustable DC power supply was connected between the red (power) and black (ground) wires.
- Step 2: A digital multimeter was positioned between the blue (alternator field) and black (ground) wires.
- Step 3: Starting at 0 V, the supply voltage was gradually increased, the alternator field voltage was expected to follow the input voltage until the regulation point was reached which is defined by the potentiometer accessible from the top of the voltage regulator.

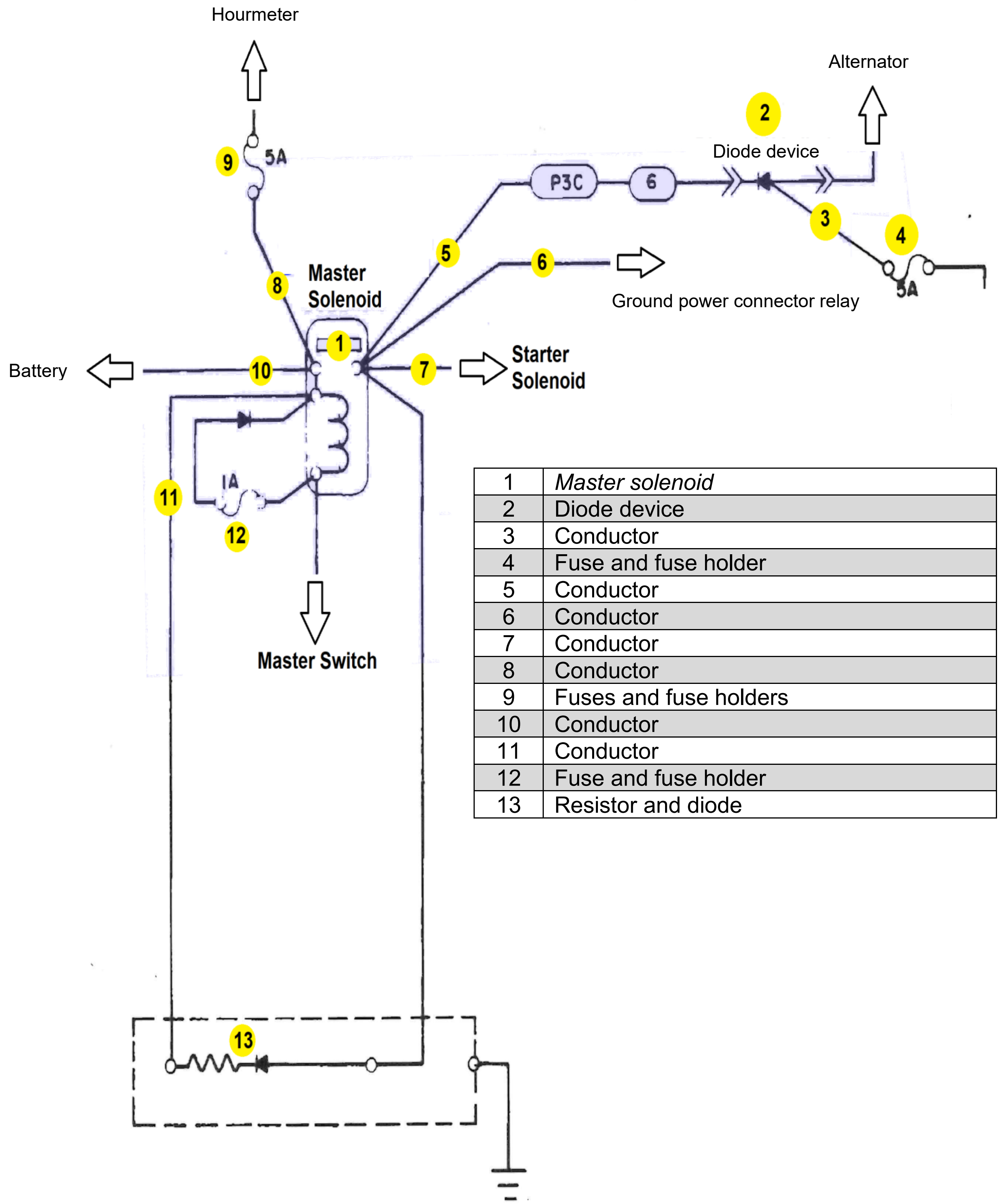
This test was carried out by the ENAC at Castelnaudary, in the presence of the BEA. The following results were obtained:

Supply voltage	Alternator field voltage
12	11.9
12.5	12.4
13	12.9
13.5	13.4
13.6	13.5
13.7	13.3
13.8	13.06
13.9	12.9
14	12.55
14.1	0

According to Lamar Technologies, the behaviour of the alternator control was normal. This component was therefore functional.

2.4.9. Master solenoid and part of associated electrical system

Figure 134 shows the reconstruction of the electrical system associated with the master solenoid.



1	*Master solenoid*
2	Diode device
3	Conductor
4	Fuse and fuse holder
5	Conductor
6	Conductor
7	Conductor
8	Conductor
9	Fuses and fuse holders
10	Conductor
11	Conductor
12	Fuse and fuse holder
13	Resistor and diode

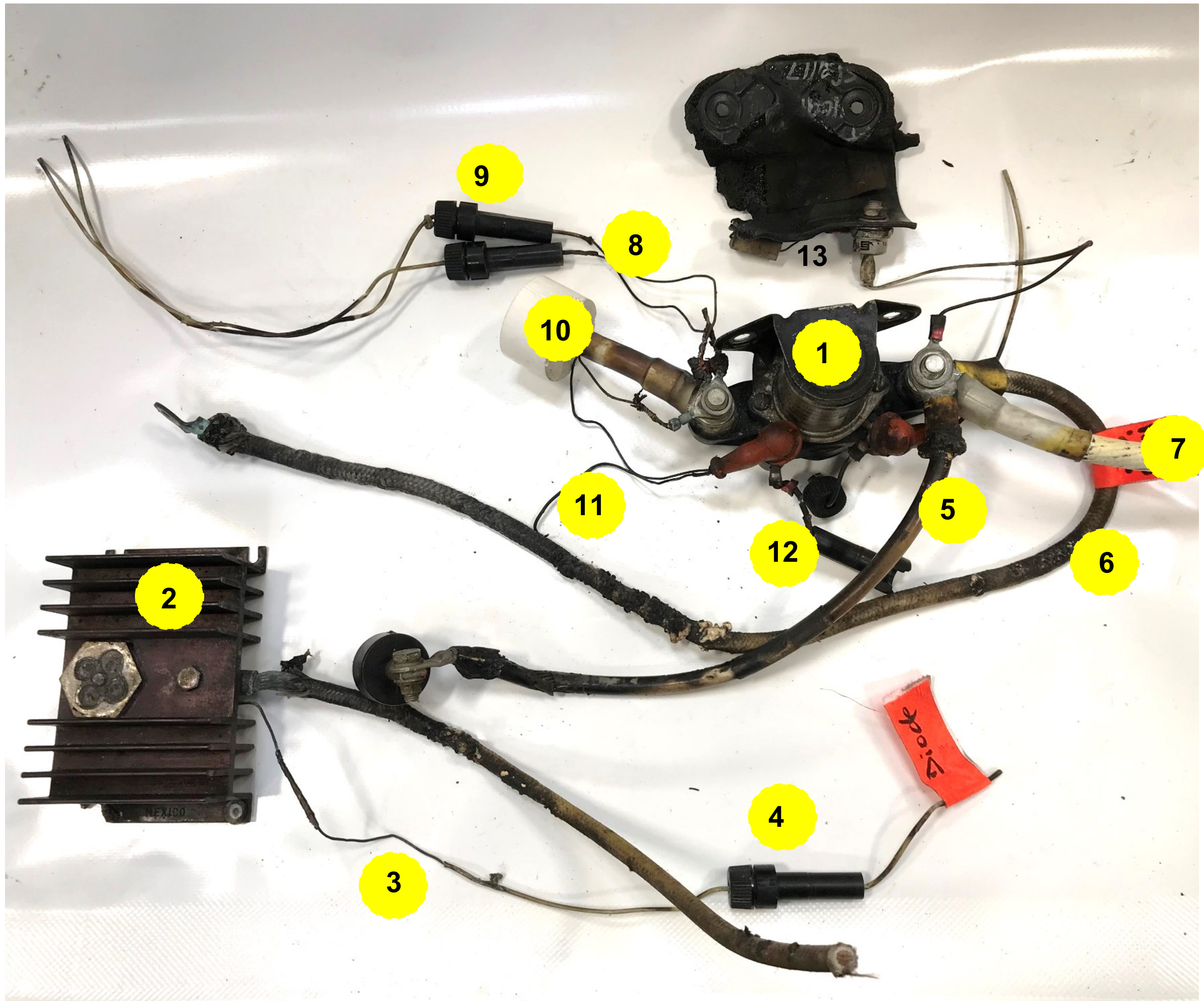


Figure 134: Reconstruction of electrical system associated with master solenoid
Source: BEA

The master solenoid was found in the expected position in the aircraft, along the right side of the battery. Residue from the battery compartment was found along the side of the master solenoid. This residue was deformed due to the heat reached during the accident.

The outer surface of the master solenoid was black in colour due to exposure to high temperatures and smoke (**Figure 135**).

The master solenoid was tested at the BEA. The test consisted of repeatedly electrically controlling the master solenoid and systematically checking for its activation and deactivation. The test was carried out at room temperature. No malfunction was identified. This component was functional.

The master solenoid was then disassembled in the BEA laboratory[22] (**Figure 136**). The internal components of the master solenoid were in position, without significant damage. Damage to the contacts through which the electrical connection is made was noted. This damage corresponded to local melting of the material, due to the occasional presence of electric arcs. This type of damage is common on this type of equipment. This damage did not affect the functionality of the equipment.

Figure 135: Master solenoid
Source: BEA

[22] On this type of equipment, the functional test is insufficient. An assessment of the condition of the internal contacts is important.

Figure 136: Master solenoid disassembled in the BEA laboratory
Source: BEA

The insulation of the conductor (item 3) (**Figure 134** and **Figure 137**) was destroyed over several centimetres level with the diode device.

The fuse holder (item 4) (**Figure 134**) was whole, with the fuse in place and whole (**Figure 106**).

The conductor (item 5) (**Figure 134**) between the diode device and the master solenoid, was in position, secured to the terminal of the latter (**Figure 137**). The external protection, located between the lug on the master solenoid and the conductor, had partially melted (**Figure 138**).

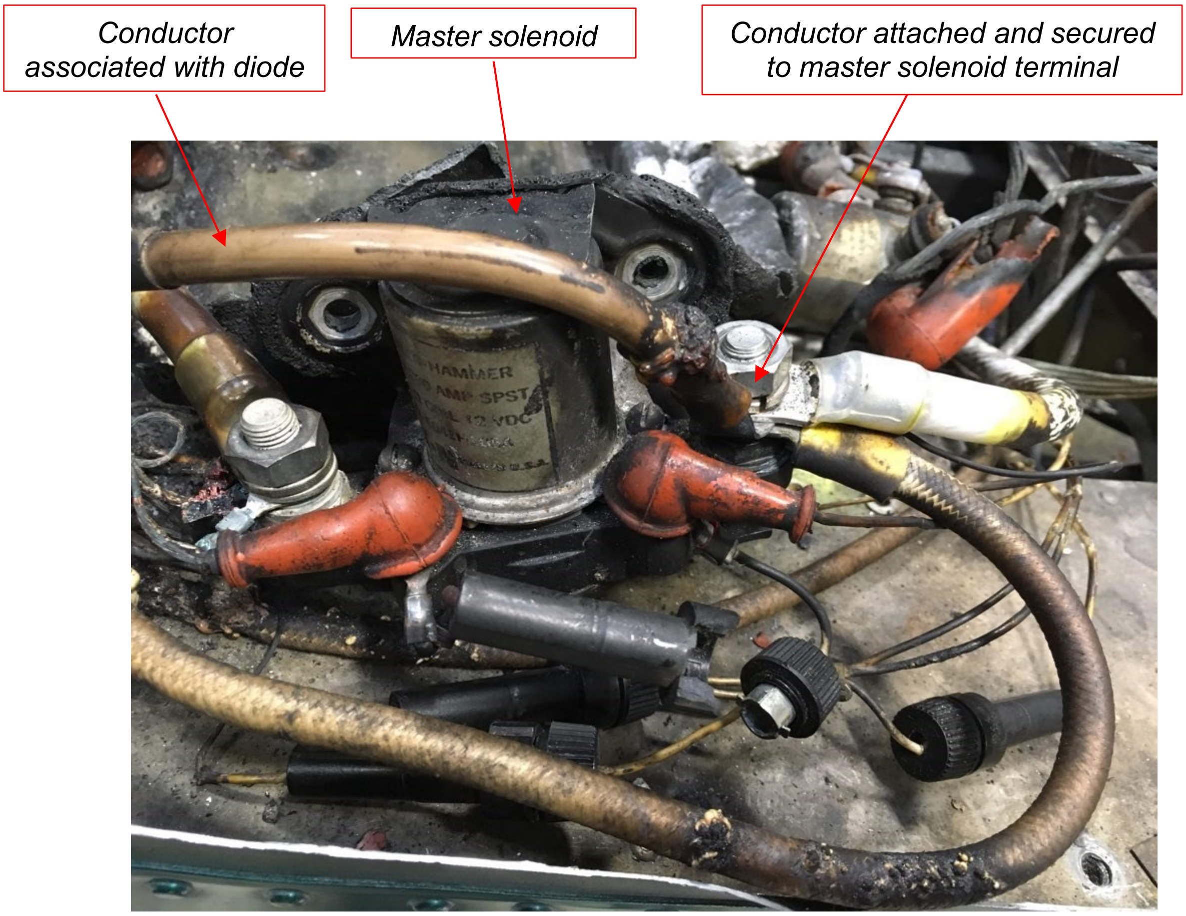


Figure 137: Assembly on aircraft
Source: BEA

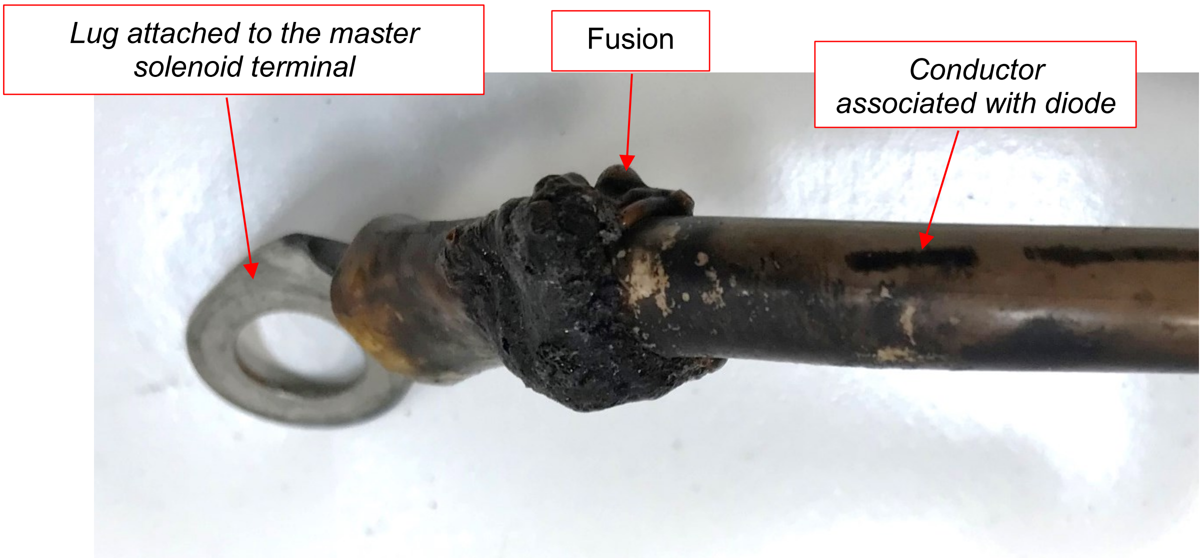


Figure 138: Detail of conductor (item 5)
Source: BEA

The conductor (item 6) (**Figure 134**) between the ground power connector relay and the master solenoid was in position, secured to the latter (**Figure 139**). This conductor showed no substantial damage.

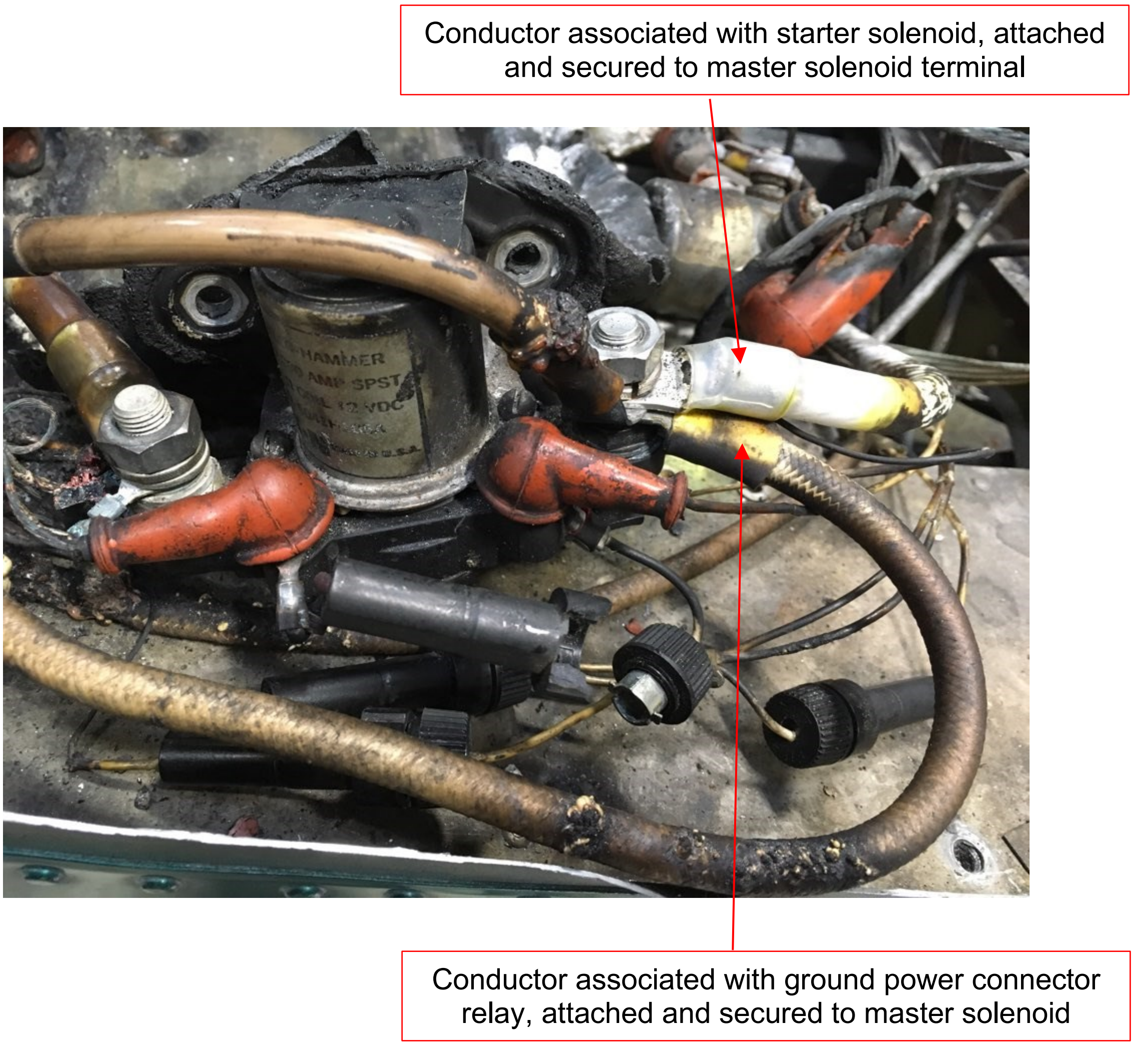


Figure 139: Assembly on aircraft
Source: BEA

The conductor (item 7) (**Figure 134**) between the starter solenoid and the master solenoid was in position, secured to the terminal of the latter (**Figure 139**). This conductor showed no substantial damage.

The conductors (item 8) (**Figure 134**) were attached and secured to the master solenoid terminal. The conductor insulation was destroyed between the master solenoid and the fuse holders (**Figure 140**).

The fuse holders (item 9) (**Figure 134**) were whole, with the fuses in place and whole (**Figure 141**).
The conductors between the fuse holders and the hourmeters were continuous and not damaged.

Figure 140: Conductor (item 8) and fuse holders (item 9)
Source: BEA

Figure 141: Fuses (item 9)
Source: BEA

The conductor (item 10) (**Figure 134**) between the positive terminal of the battery and the master solenoid was in position and secured to the terminal of the latter (**Figure 142**). Level with this attachment, the conductor was of a light brown colour but there was no sign of melting.

Figure 142: Assembly on aircraft
Source: BEA

The conductor (item 11) was in position and secured to the master solenoid terminal. However, this conductor was severed level with the resistor (item 13) (**Figure 143**). This break was situated level with the area where it was observed that the battery box had melted. The break of the conductor was most likely a consequence of the accident[23], melting of the conductor or rupture after embrittlement due to its exposure to an abnormally high temperature.

The device (item 13) (resistor and diode) was whole without external damage (**Figure 144**).

[23] According to the manufacturer Piper, the break of this conductor does not modify the functionality of the master solenoid.

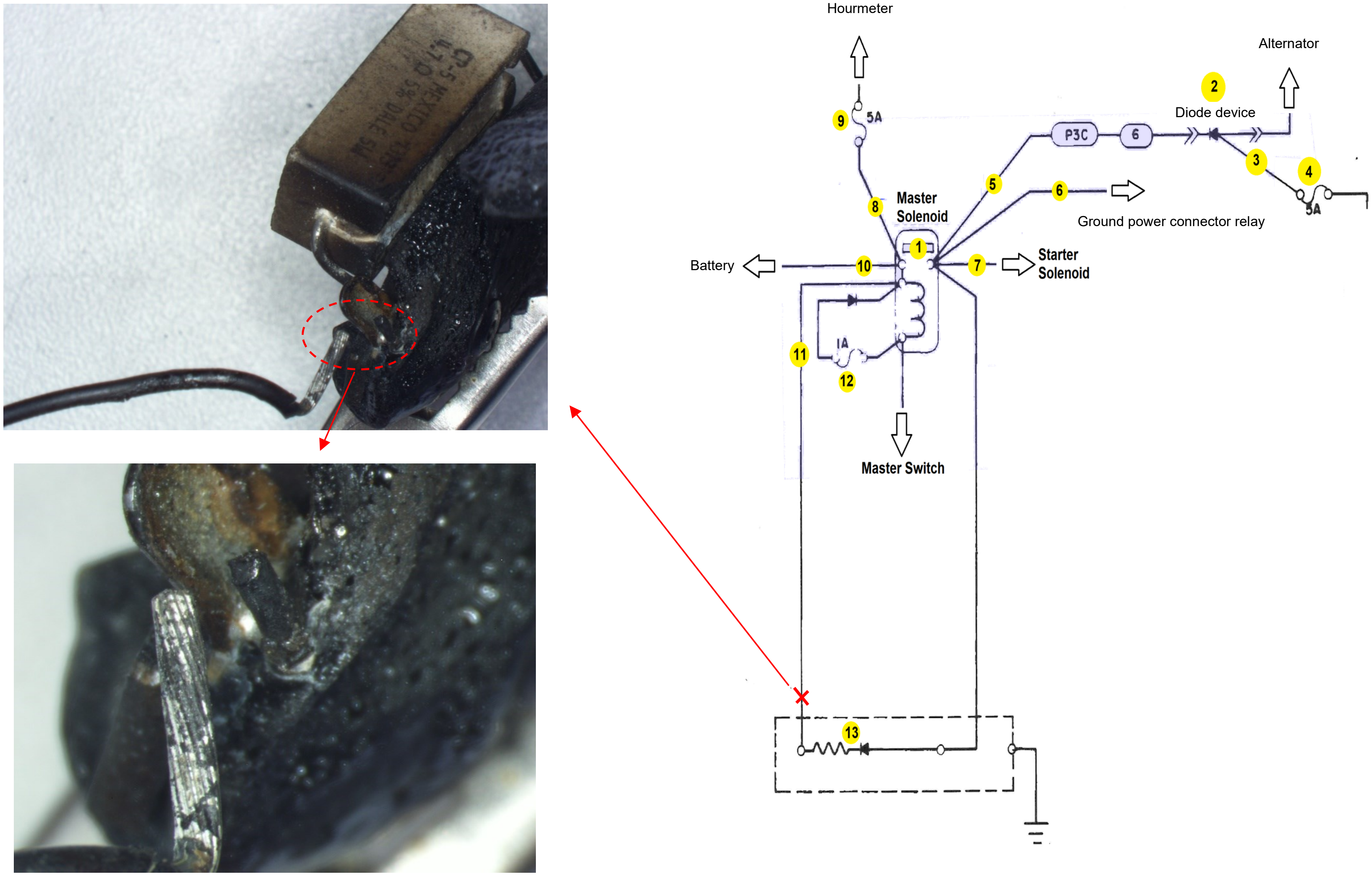


Figure 143: Conductor (item 11)
Source: BEA

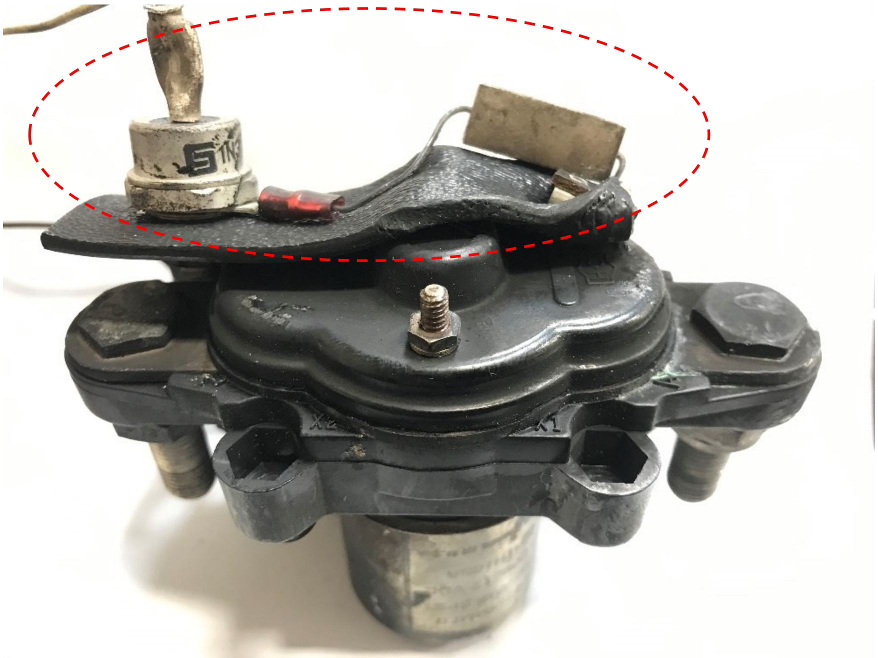


Figure 144: Device (item 13)
Source: BEA

The fuse holder (item 12) was found open, an area of the fuse holder had ruptured (**Figure 145** and **Figure 146**). The 1A fuse was found whole in the aircraft, nearby (**Figure 147**). What caused the fuse holder to rupture remains unknown. The detailed examination of the break did not provide more information because of the nature of the fuse holder material.

The conductors associated with the fuse holder were in position and secured to the master solenoid terminals.

<u>Note:</u> Piper was asked about the role of this fuse and the consequences in the event of discontinuity.

According to Piper, the diode with the 1 A fuse is intended to suppress surges in the relay coil that blocks the electromagnetic fields that can induce electrical noise that might interfere with nearby electrical signals, noise in communication radios or other sensitive avionics. The system will operate in the same way if the fuse is open, except for the possible electrical noise/interference mentioned above.

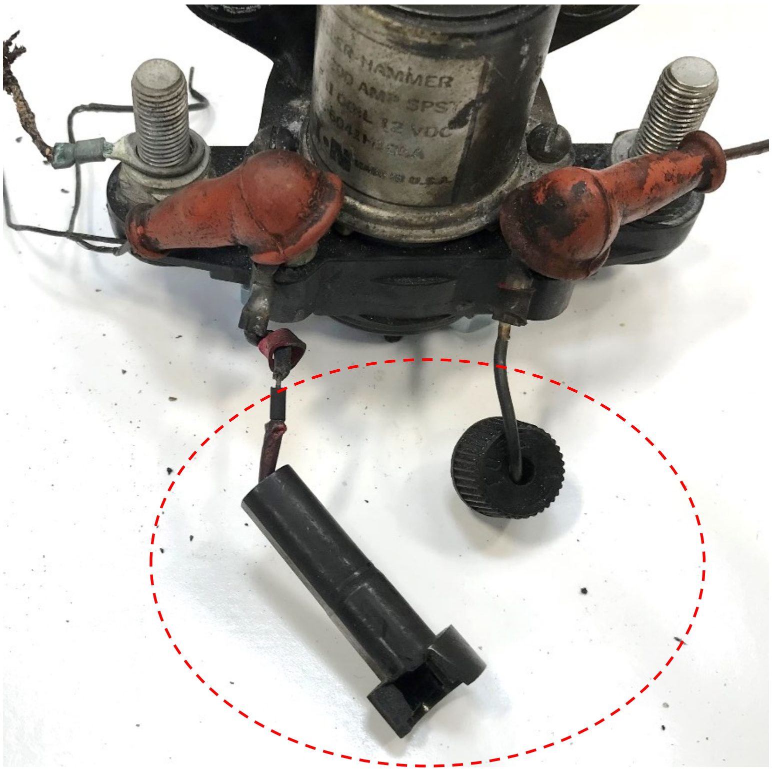


Figure 145: Fuse holder (item 12)
Source: BEA

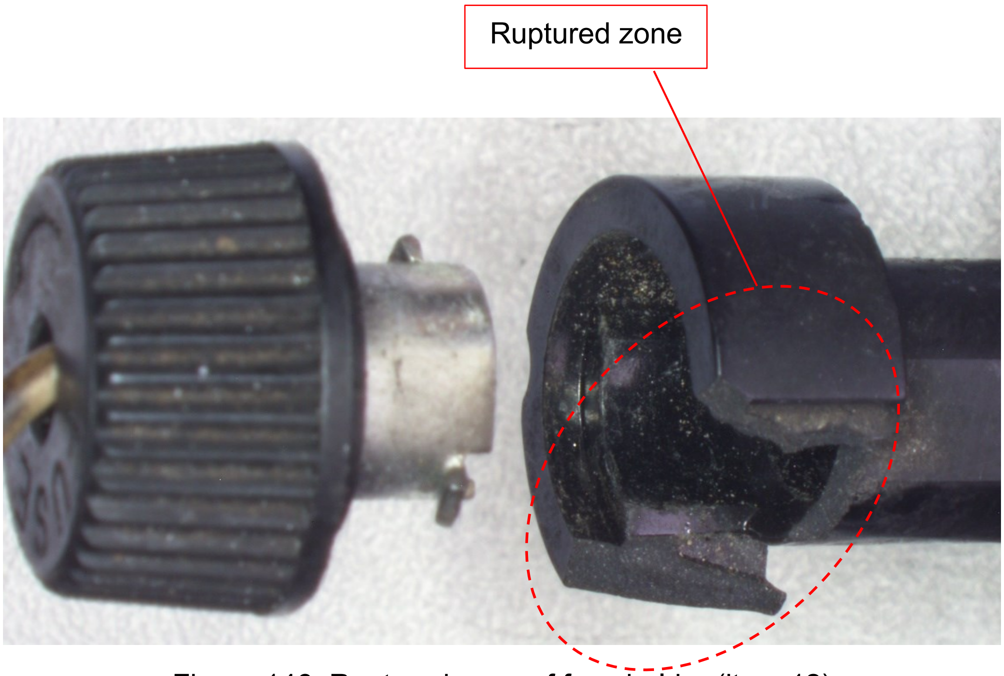


Figure 146: Ruptured zone of fuse holder (item 12)
Source: BEA

Figure 147: Fuse associated with fuse holder (item 12)
Source: BEA

2.4.10. Conclusion of phase B

Three singularities were identified:
- The very degraded state of the positive terminal of the battery.
- The damage to the conductor connected to this terminal and the substantial force required to bring the end of this conductor in line with the battery terminal.
- The separation of the diode device into two parts and the melting of the associated metal cooler in line with one of the points where it is attached to the aeroplane floor.

The functionality of the following components was checked and validated:
- alternator;
- alternator control;
- contactors;
- controls on instrument panel.

The detailed examination of the positive terminal of the battery found that the two main components of this terminal had separated due to the melting of the solder which normally ensures the connection.

The negative terminal of the battery was intact.

The diode device had also separated due to the melting of a soldered connection. The four identical cells making up the device had all short circuited.

The metal cooler associated with the diode device had melted in line with one of the points where it is attached to the aeroplane floor. The analyses carried out in this melted area found that an insulating washer and a screw were present before the melting of the cooler.

The observation of a diode device on an identical aeroplane found that the cracking of insulating washers was possible.

Tests carried out by Concorde Aircraft Batteries showed that:
- the damage found in line with the positive terminal of the battery was compatible with that found after being subject to a strong external heat source (such as a flame);
- the diode device bore a passing current of around 70 A for 60 min;
- the presence of a short-circuit in line with one of the points where the metal cooler is attached to the aeroplane floor led to a very local, extremely high increase in temperature and the inflammation of the ABS cover.

2.5. PHASE C: engine start-up system

The purpose of phase C of the investigation process was to assess the engine start-up system. According to the manufacturers consulted, a fault in this system could explain the abnormally high current in the general electrical power supply system during the accident.

2.5.1. Information provided by Bogert Aviation Inc and Concorde Aircraft Batteries

The following information was given to the BEA in January 2022 after it had transmitted a complete summary of the work carried out on HB-PNP translated into English.

The three hypotheses given by Bogert Aviation Inc (Concorde Aircraft Batteries agrees with these hypotheses) to explain the abnormally high current in the system and then the fire, are the following:

- Hypothesis 1: starter solenoid blocked in closed position

    => If the starter solenoid stays in the closed position, the starter will be continually powered. This situation would lead to a very high current consumption which could explain the abnormal overheating of the system associated with the battery.

- Hypothesis 2: a malfunction of the start-up switch leading to a continuous start-up procedure.

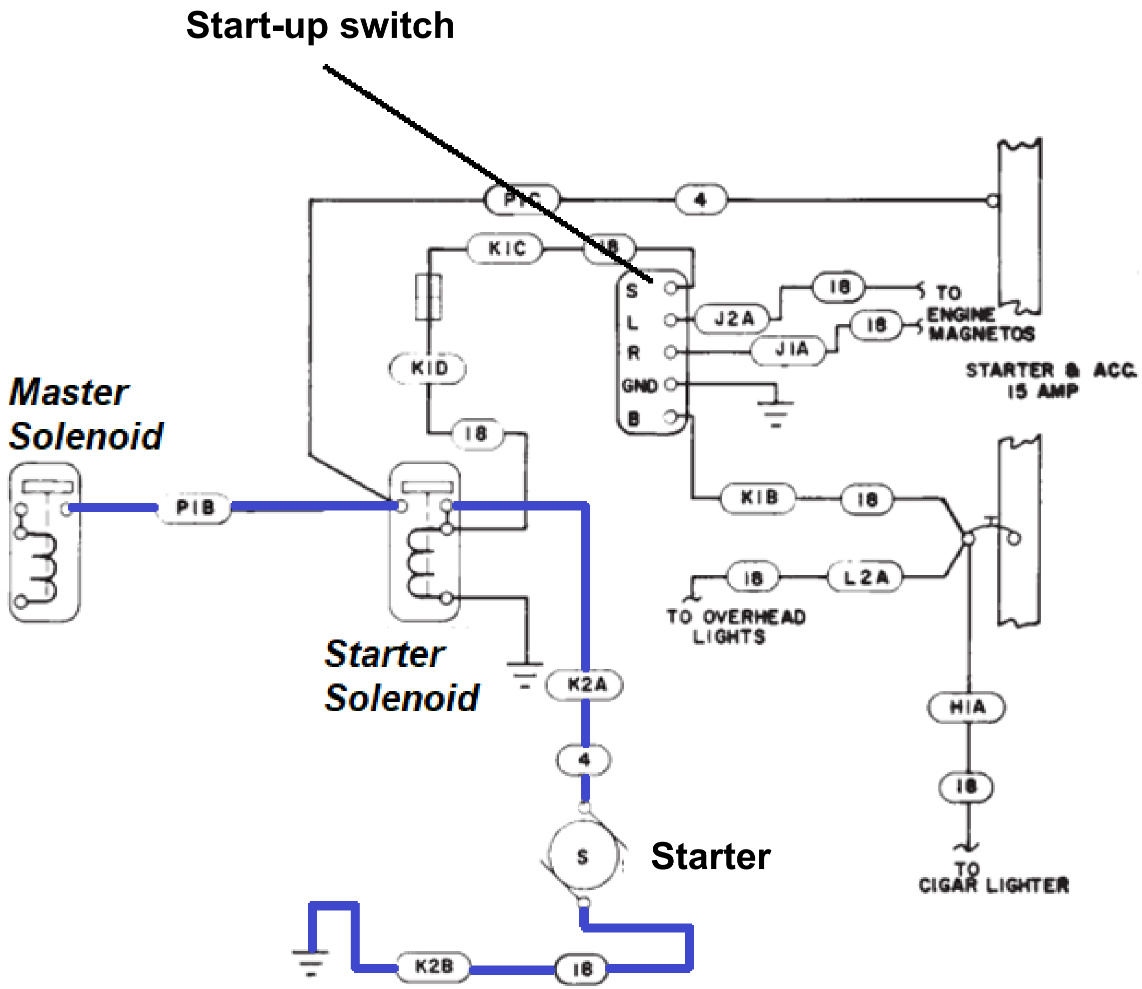
    => The situation and consequences would be identical to those in hypothesis 1.

- Hypothesis 3: a short-circuit between the master solenoid and the starter solenoid.

To validate or invalidate these hypotheses, Bogert Aviation Inc recommended examining the following components:
- engine start-up pinion;
- starter;
- starter solenoid;
- start-up switch;
- the various associated conductors.

2.5.2. Composition of system associated with starter

The diagram of the system associated with the starter is given in **Figure 148**. The names of the main components are also specified.



Start-up switch	Teledyne Continental Motors – FAA-PMA – 10-357200-1 L J0797--H
Starter solenoid	Lamar Technologies – P/N: X61-0030
Starter	Teledyne Continental Motors – Energizer 12 – P/N: 646238– S/N: G-099146

Figure 148: Electrical system associated with starter

Source: Maintenance Manual

Note: The conductors shown in blue are those taken into account in STC SA3531NM (paragraph 2.4.3).

2.5.3. Condition of conductors

All the conductors were whole, with no damage observed on their insulation. Their continuity was checked and validated.

2.5.4. Start-up switch

The start-up switch has five positions:

Positions	Comments
OFF	The two magnetos are grounded. Neither magneto is energized.
R	The left magneto is grounded. Only the right magneto is energized.
L	The right magneto is grounded. Only the left magneto is energized.
BOTH	Neither magneto is grounded. Both magnetos are energized.
START	Neither magneto is grounded. Both magnetos are energized. The starter is electrically powered.

The START position is an unstable position. To use this position, the pilot must hold the key in this position until the engine starts up. When the pilot releases the key, the switch automatically returns to the BOTH position and the electrical power supply to the starter is instantaneously cut off.

Figure 149: Start-up switch plate indicating the five possible positions
Source: BEA

2.5.4.1 – Condition and position in aeroplane

The start-up switch was in position on the instrument panel. The ignition key was not present. The start-up switch was in the OFF position (**Figure 150**).

Figure 150: Start-up switch in position on instrument panel
Source: BEA

Behind the switch, the conductors were in position and no damage was observed (**Figure 151**). The insulation of these conductors was of a brown colour due to the smoke and heat reached during the accident.

The conductor connections to the switch and the other components were consistent with the diagram specified by the manufacturer.

Figure 151: Rear face of start-up switch
Source: BEA

2.5.4.2 – Test

The start-up switch was functional. When the START position was released, the starter's electrical power supply was instantaneously cut off as specified.

2.5.4.3 – Destructive examination

The internal components of the switch were in position.

Black colouring and incipient damage to various electrical contact zones could be observed (**Figure 152**).

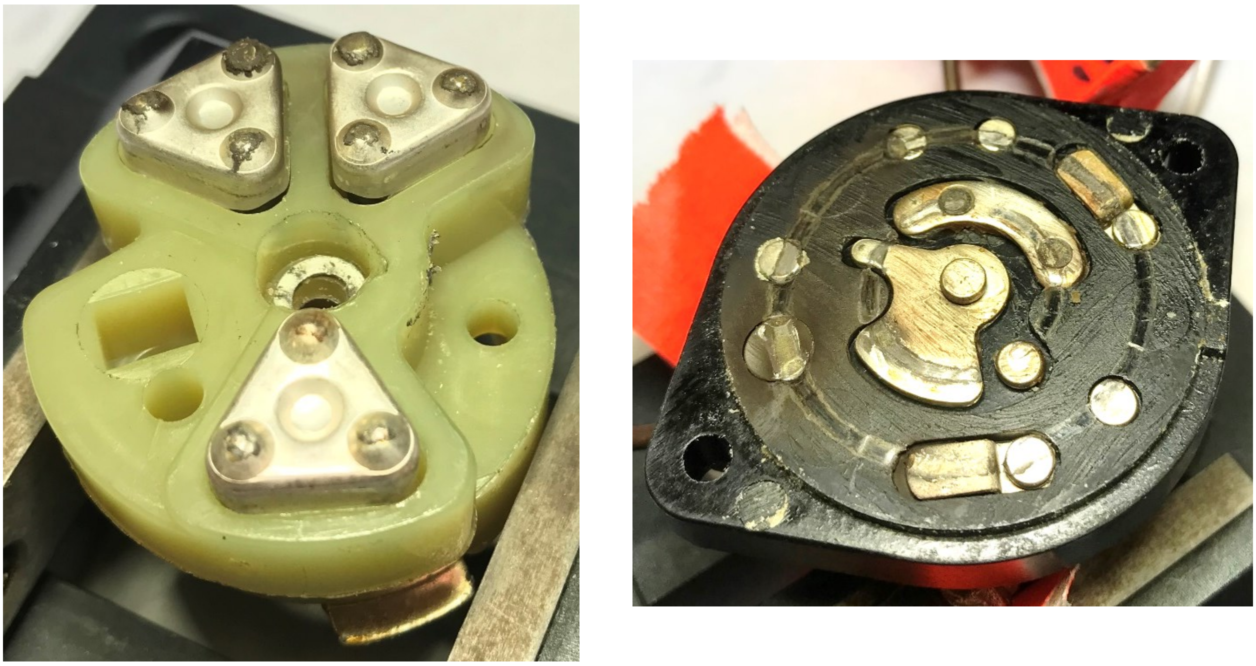


Figure 152: Electrical contact zones in start-up switch
Source: BEA

2.5.5. Starter solenoid

2.5.5.1 – Condition and position in aeroplane

The starter solenoid was in position in the aeroplane with the various associated conductors in position and secured (**Figure 153**). On the wreckage, it was observed that the starter solenoid was electrically open as would be expected, the start-up switch being in the OFF position.

Figure 153: Starter solenoid in position in the engine compartment on the firewall
Source: BEA

The starter solenoid showed no damage on its outer surface nor unusual colouring (**Figure 154** and **Figure 155**).

Figure 154: Starter solenoid removed from aircraft
Source: BEA

Figure 155: Starter solenoid removed from aircraft
Source: BEA

2.5.5.2 – Non-destructive examination

The starter solenoid was subject to a tomographic examination. Several images taken during this examination are shown on the following pages.

This examination did not find any internal damage.

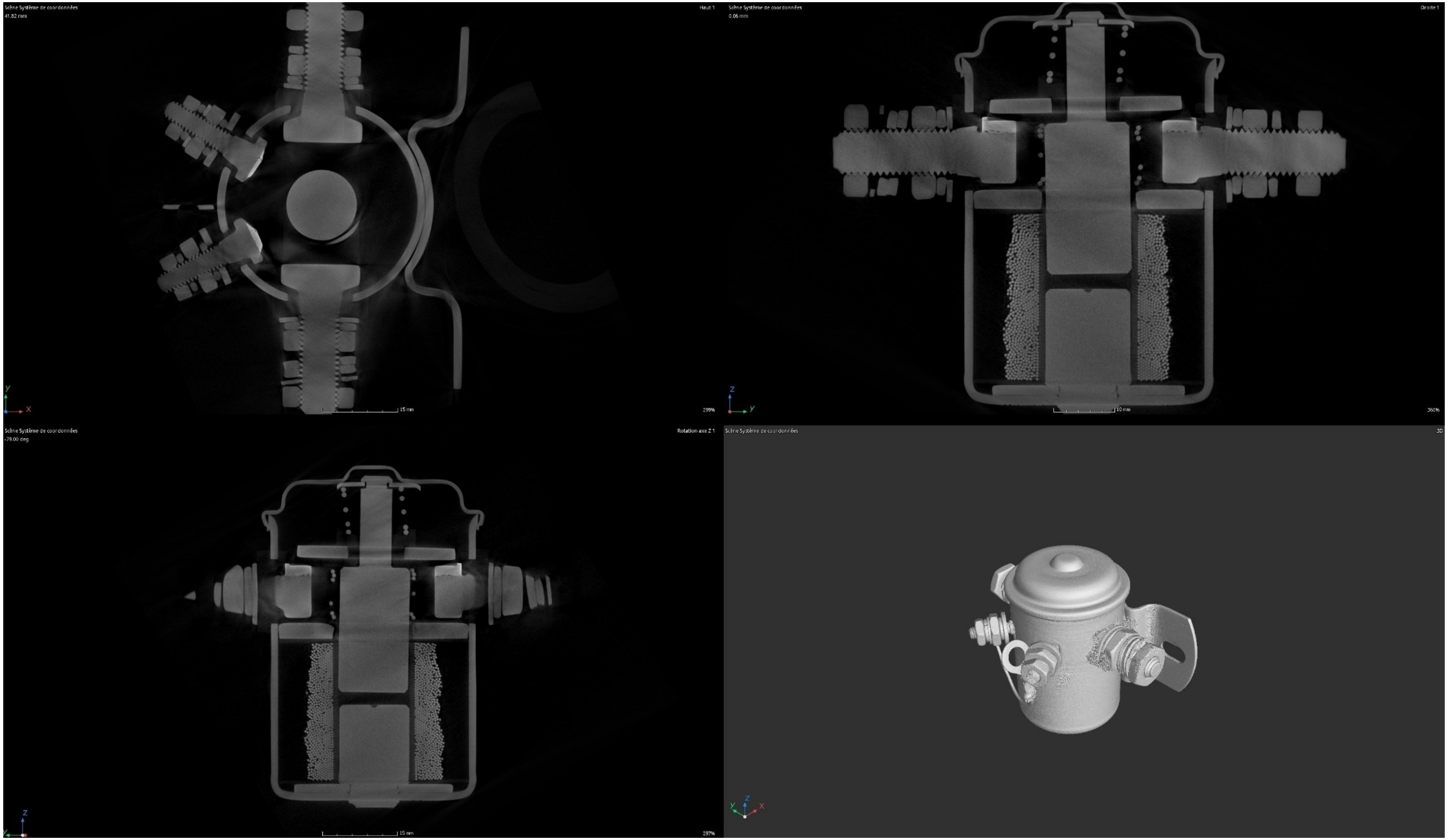


Figure 156: Images from tomographic examination
Source: BEA

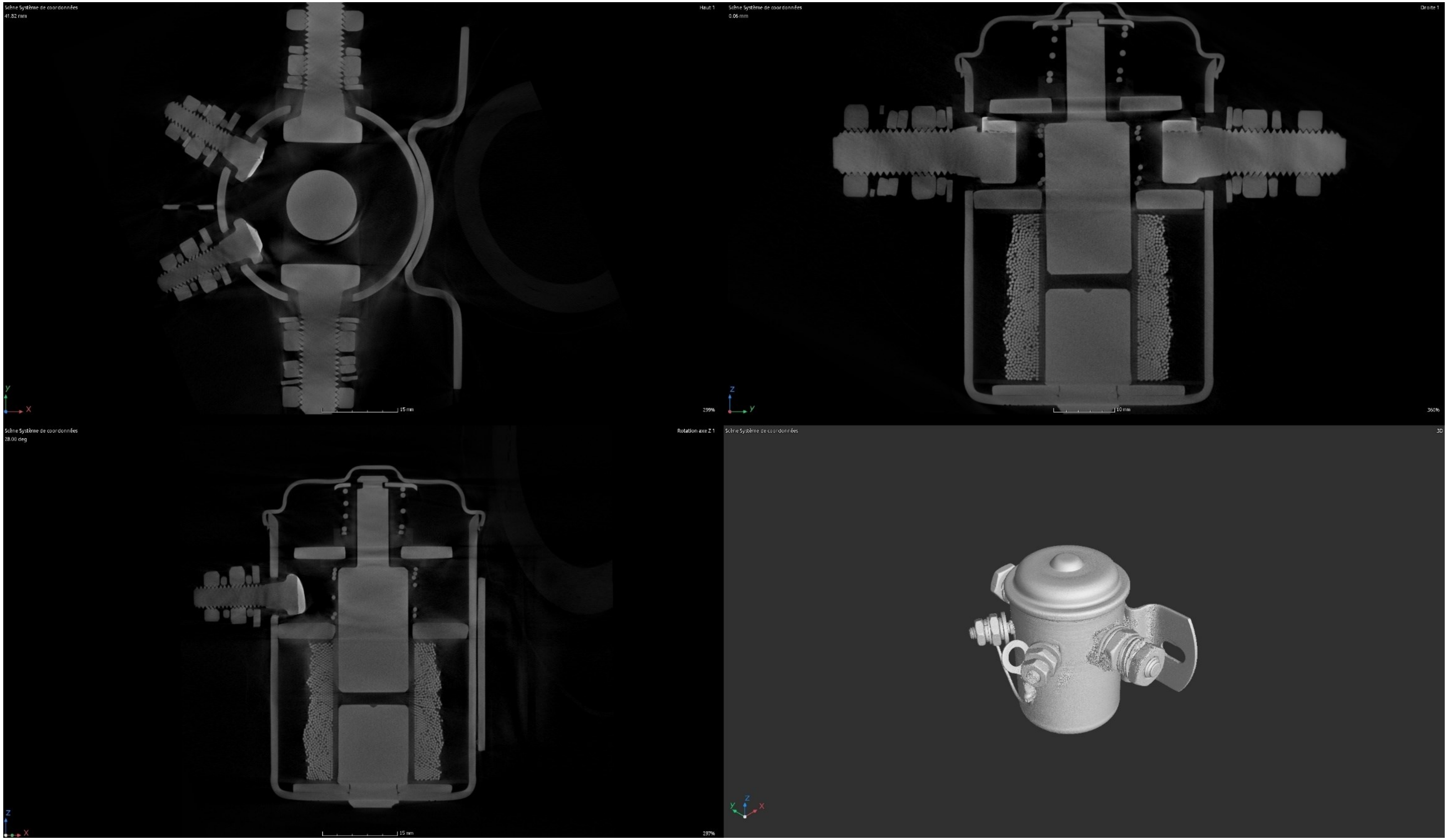


Figure 157: Images from tomographic examination
Source: BEA

2.5.5.3 – Test

Note: Lamar Technologies indicated to the BEA that there was no specific procedure for testing this starter solenoid. The only document available was a troubleshooting document.

The test consisted of repeatedly electrically controlling the starter solenoid and systematically checking for its activation and deactivation.

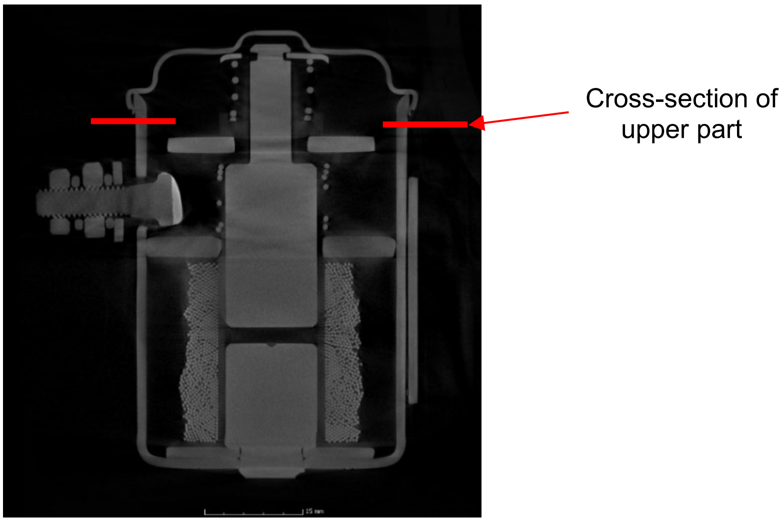
This test was carried out at various external temperatures between 20°C and 90°C.

No malfunction was found.

2.5.5.4 – Destructive examination

This examination consisted of opening the upper part of the starter solenoid.

The components were in position. Signs of the creation of small electrical arcs were observed. This type of damage is common on this type of equipment. This damage did not affect the functionality of the equipment.



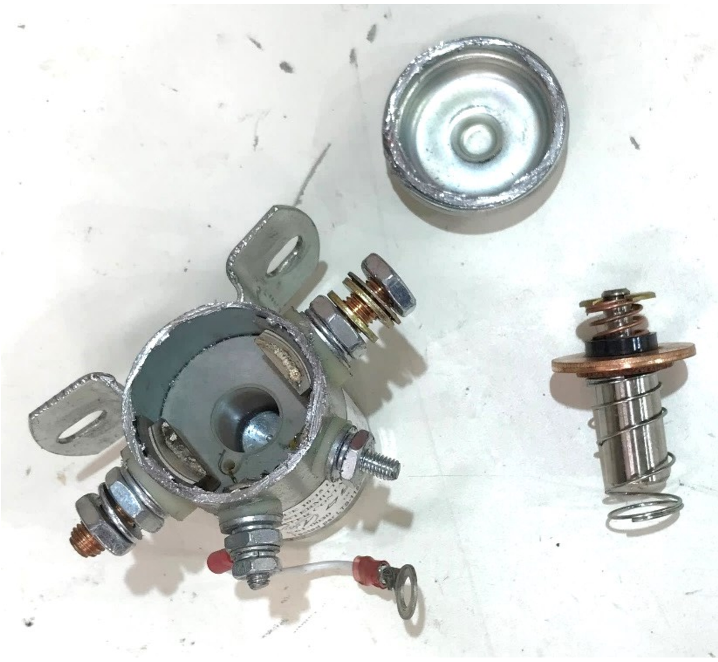


Figure 158: Destructive examination of starter solenoid
Source: BEA

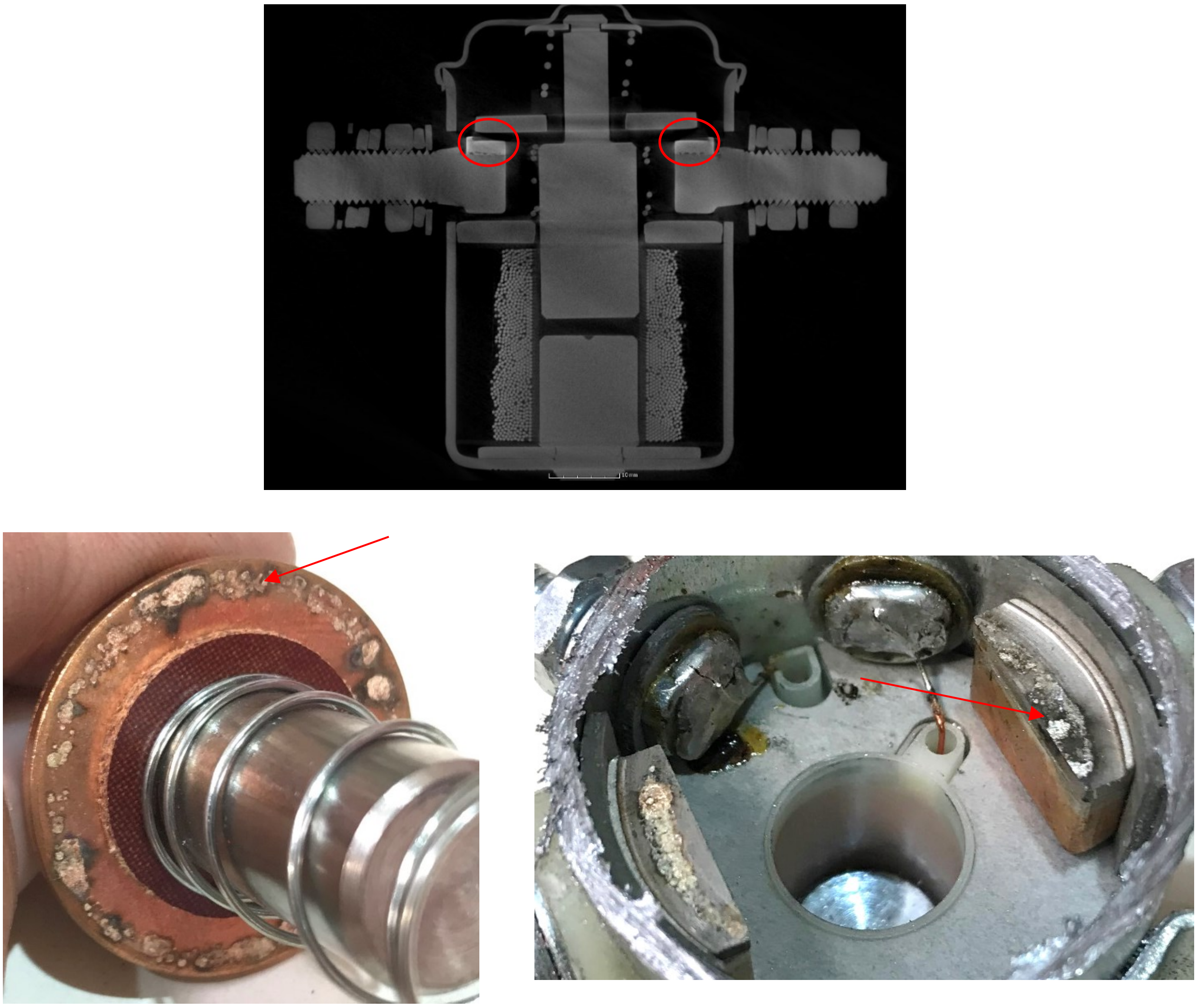


Figure 159: Signs of electrical arcs in operation
Source: BEA

2.5.6. Starter

2.5.6.1 – Condition and position in aeroplane

The starter was in position on the engine (**Figure 160**), assembled and secured to the starter adapter.

The electrical power supply conductor was in position and secured.

After disassembling it, it was observed that the starter drive shaft was whole and in position (**Figure 161**).

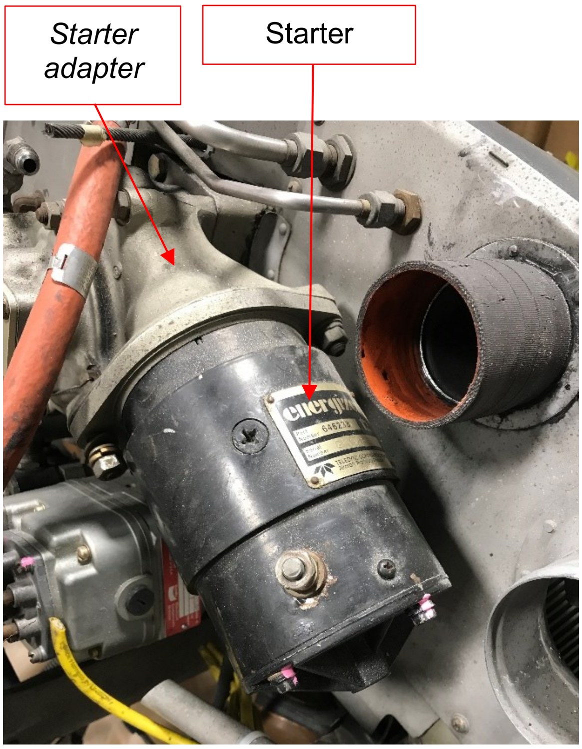


Figure 160: Starter in position on the starter adapter. Electrical power supply cable removed before photo taken
Source: BEA

Figure 161: Starter drive shaft
Source: BEA

2.5.6.2 – Test

The starter was tested on a dedicated bench at the ENAC at Castelnaudary.

This test consisted of electrically powering the starter with a voltage of 13 V. The current was measured. The following curves (**Figure 162**) show the data measured on the starter of HB-PNP.

Continental Motors, the manufacturer, was asked for its opinion via the NTSB; no reply was given.

According to the ENAC specialists at Castelnaudary, the curves obtained did not call for any particular comment.

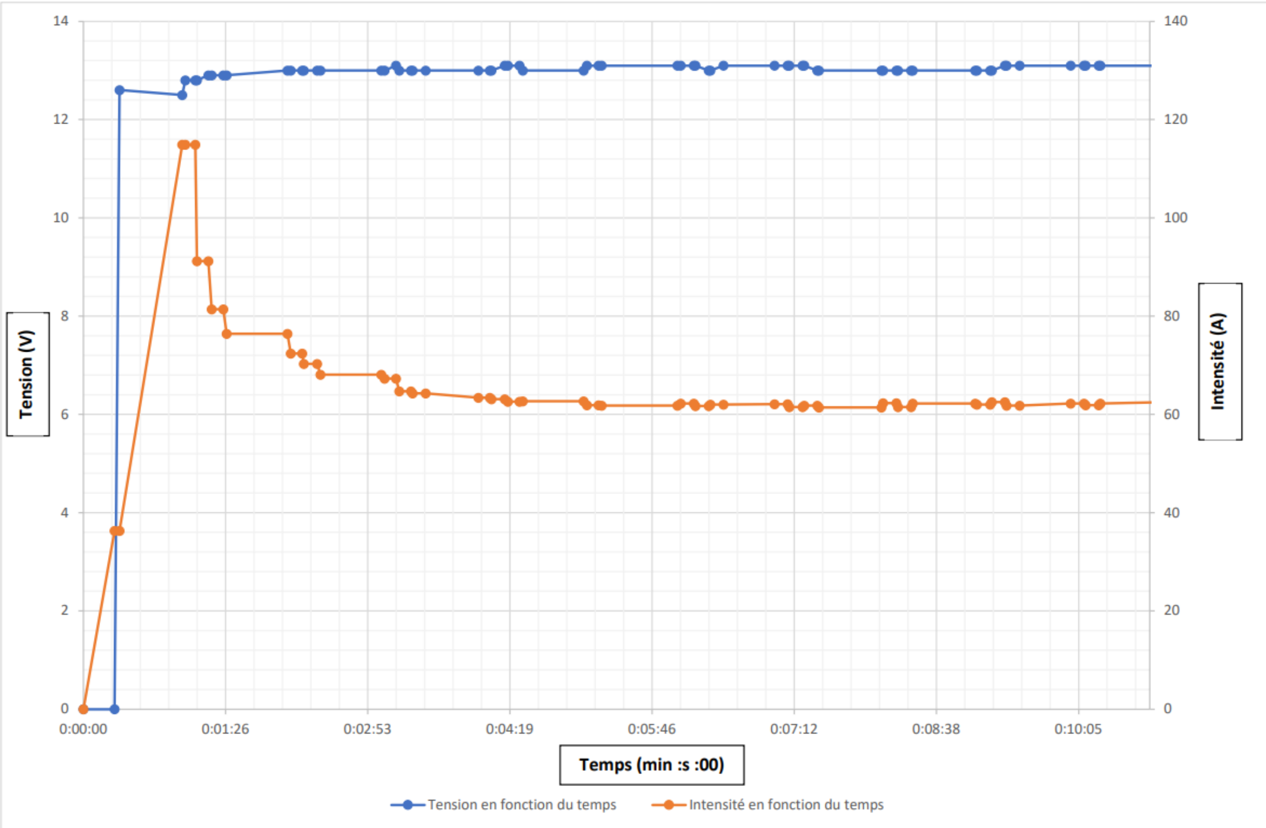


Figure 162: Curves obtained during starter test
Source: BEA

2.5.7. Starter adapter

2.5.7.1 – Composition and operating principle

The starter adapter is the mechanical link between the starter and the engine.

Figure 163 shows the composition of the starter adapter An arbitrary parts list is proposed to simplify comprehension in the rest of the document.

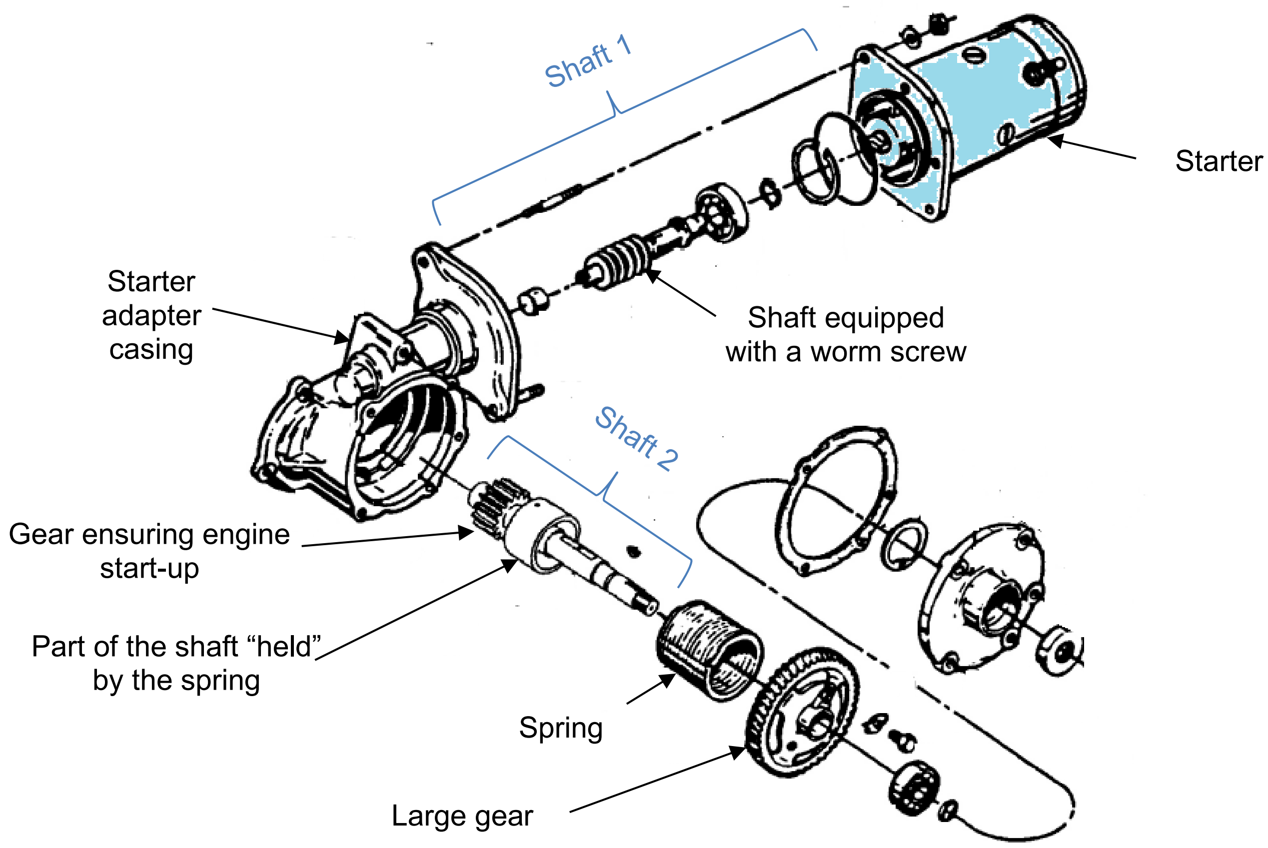


Figure 163: Starter adapter schematic diagram
Source: BEA

The starter is mechanically connected to shaft 1. The latter is equipped with a worm screw which allows the large gear to start rotating when the starter is operated.

When the large gear starts rotating, the spring tightens around shaft 2. The latter then rotates and the engine starts up.

This interface between the spring and shaft 2 is a “clutch” mechanism which allows the engine to start turning after being started up without driving the starter via the starter adapter.

2.5.7.2 – Test

The test consisted of reassembling the starter on the starter adapter and electrically powering the starter.

The starter adapter was functional.

2.5.8. Conclusion of phase C

The examinations carried out on the engine start-up system did not bring to light any singularity which could explain a malfunction of one or more components during the occurrence flight.

2.6. PHASE D: autopilot system

The purpose of phase D of the investigation process was to assess the system associated with the aeroplane's autopilot. The accident flight was the aeroplane's first flight following the completion of work to modify the aircraft's avionics and to install this autopilot.

The autopilot was added under FAA GARMIN STC SA01866WI or EASA STC 10060846.

According to Work report No 11687 (document written during installation), the work was carried out from 19 June to 16 July 2020.

2.6.1. Content of GARMIN STC SA01866WI

The following equipment was installed in the scope of the Garmin STC SA01866W on HB-PNP:

- Three servocontrols positioned as shown in **Figure 164** and **Figure 165**.
  - a servocontrol for the roll axis (**Figure 167**);
  - a servocontrol for the pitch axis (**Figure 166**);
  - a servocontrol for the pitch axis trim (**Figure 168**).

  These three servocontrols were identical. Their part number is:
  - Garmin GSA28 (**Figure 170**);
  - FAA PMA 011-02927-11.

  The servocontrol is electrically controlled. The action on the controls is then mechanical via metal cables.
- Two Garmin G5s, providing the EADI[24] and EHSI[25] functions (**Figure 171**), positioned on the instrument panel.
- Garmin GFC 507 interface (**Figure 172**), positioned on the instrument panel.
- Garmin GAD 29B adapter, positioned behind the instrument panel (**Figure 173**).
- Garmin GMU 11 magnetometer, positioned at the tip of the wing (**Figure 174**).

The embodiment of the Garmin STC SA01866WI on HB-PNP also led to the removal of the following former equipment:

- Century 31 autopilot system (Piper system);
- Bendix/King KR87 ADF receiver;
- Bendix/King KA 44B ADF antenna;
- Bendix/King KI229 RMI;
- Century NSD 360 compass;
- Century IB 495-4 flux valve;
- ID 755 control accessory;
- Airborne 211CC vacuum pump;
- Airborne 1G10-1 gyro suction gauge;
- Airborne 2H3-19 vacuum regulating valve;
- Airborne 1J7-1 vacuum filter.

[24] EADI: Electronic Attitude Director Indicator
[25] EHSI: Electronic Horizontal Situation Indicator

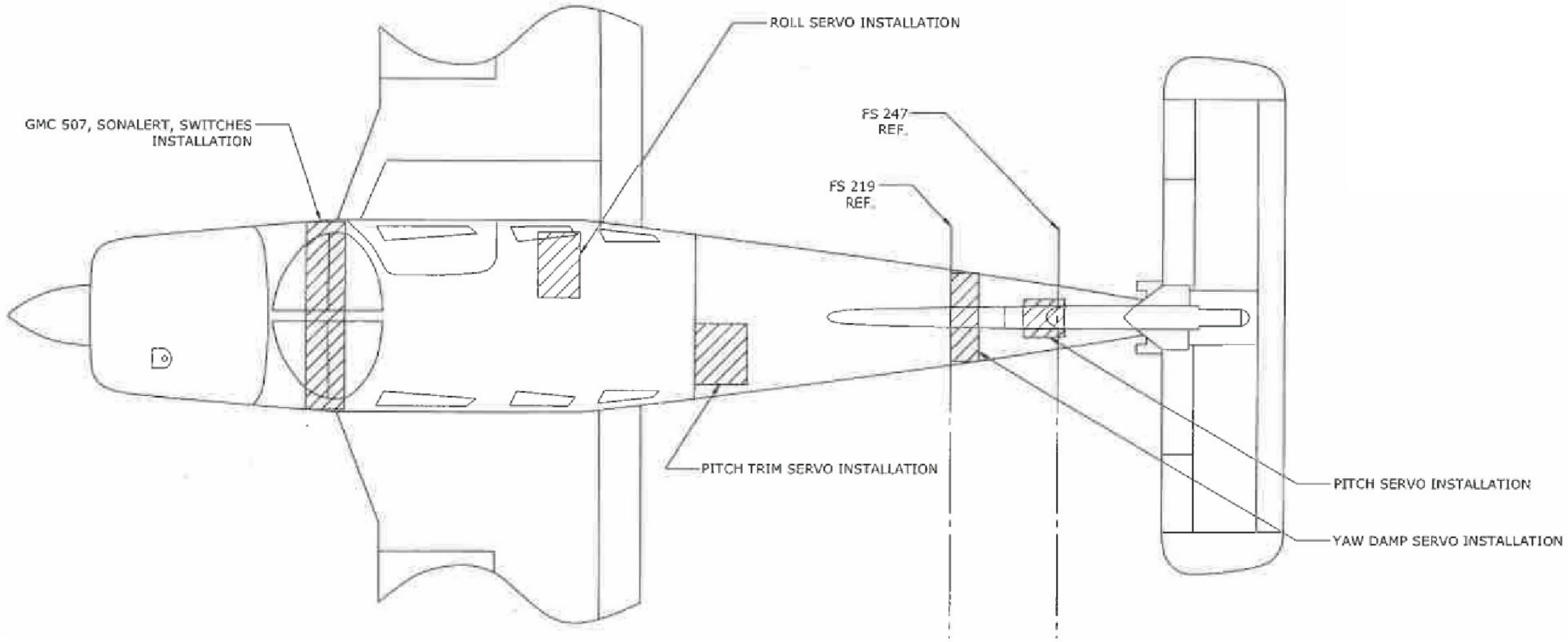


Figure 164: Position of servocontrols
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

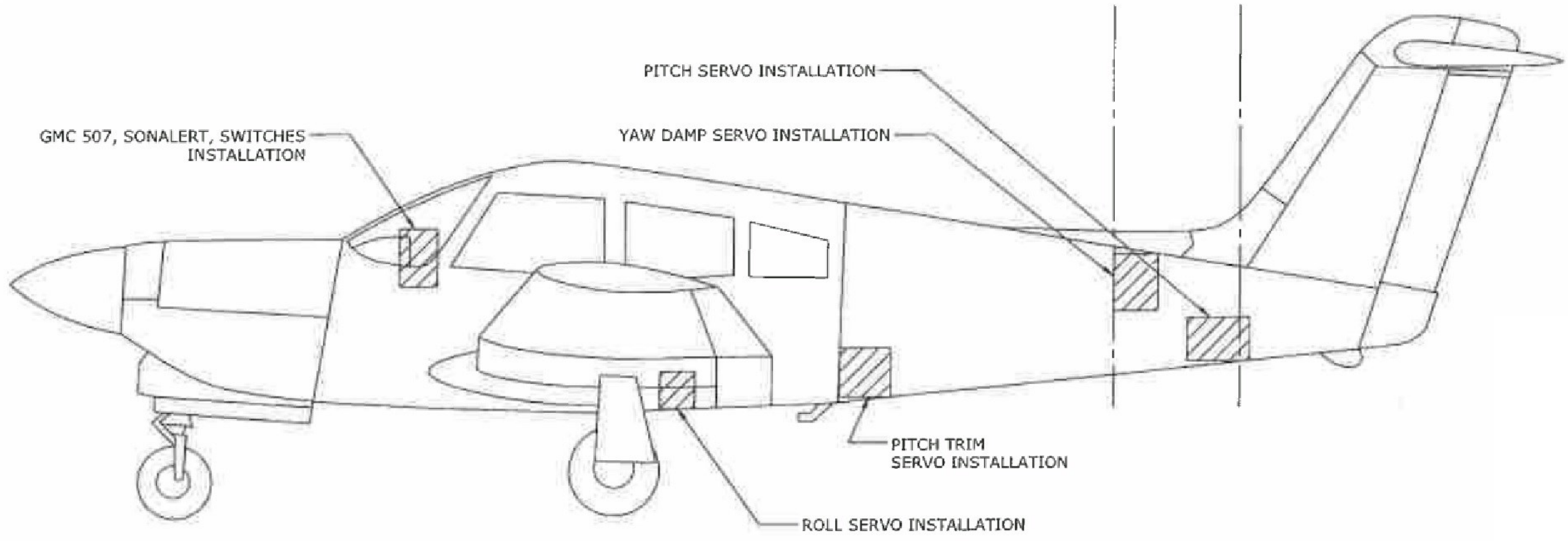


Figure 165: Position of servocontrols
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

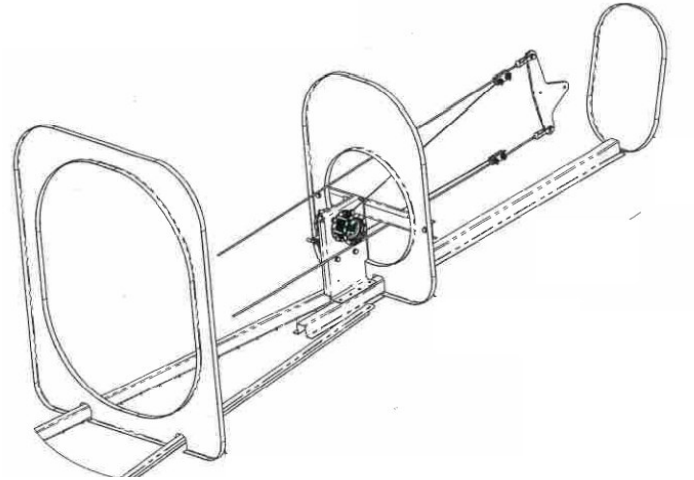


Figure 166: Installation of servocontrol on pitch axis
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

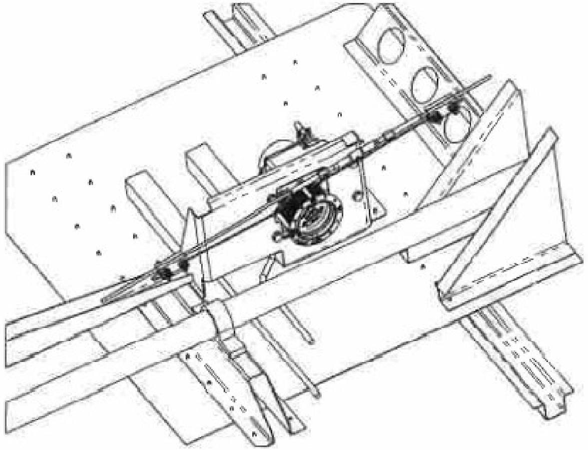


Figure 167: Installation of servocontrol on roll axis
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

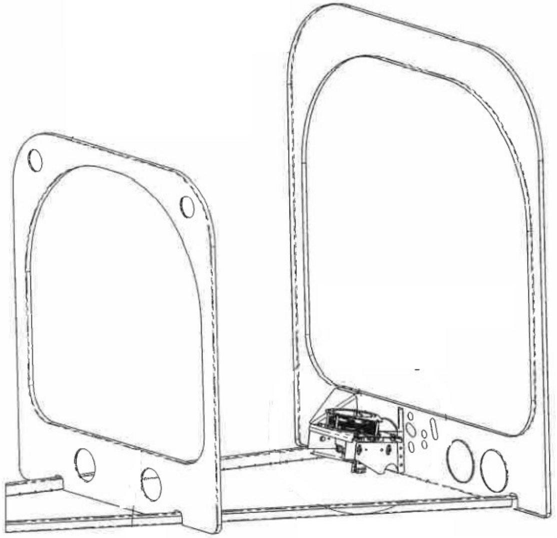


Figure 168: Installation of servocontrol on pitch axis trim
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

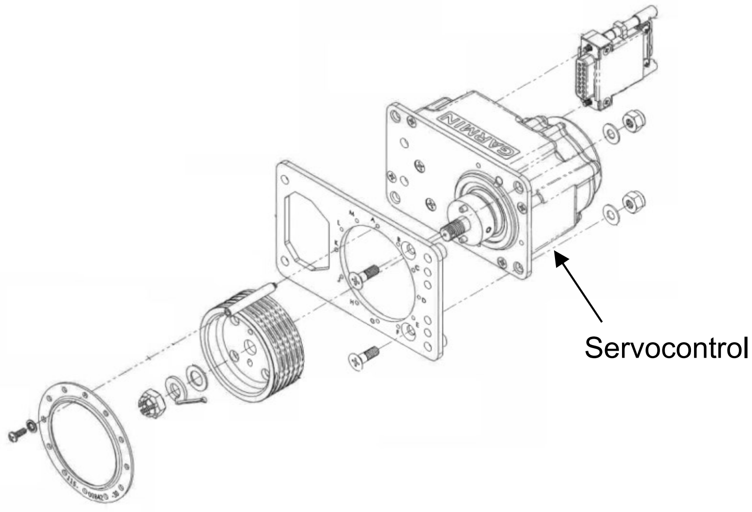


Figure 169: Servocontrol
Source: GARMIN GFC 500 Install Manual Addendum, Piper PA-28RT _ DWG No 190-02291-49

Figure 170: Garmin GSA 28 servocontrol
Source: https://sarasotaavionics.com/avionics/gsa28-certified

Figure 171: Garmin G5
Source: https://www.garmin.com/fr-FR/p/514383

Figure 172: Garmin GFC 500 interface
Source: https://www.garmin.com/en-US/p/604257

Figure 173: Garmin GAD 29 B adapter
Source: https://www.seaerospace.com/sales/product/Garmin/GAD-29B

Figure 174: Garmin GMU 11 magnetometer
Source: https://www.opaleaero.com/garmin-gmu11-magnetometre-pro1504316.html

2.6.2. Situation on HB-PNP

The servocontrols were in position on the aircraft and nominally attached (**Figure 175** to **Figure 177**). The electrical connector was in position on each servocontrol, no damage was found on the connector. The ground of each servocontrol was attached and secured to the airframe.

No damage was found on the conductors of the associated electrical system. These conductors were continuous. The outer surface of these conductors was just of a dark colour locally due to the hot smoke released during the accident.

The interfaces were in position on the instrument panel. Their outer face was damaged by the hot smoke during the accident (**Figure 178**). Behind the equipment, the conductors were in position and no damage was found.

The Garmin GAD 29 B adapter was in position behind the instrument panel, along the right side. The adapter did not show any external visual damage (**Figure 179**). The associated conductors were in position, no damage was found.

The electrical diagram associated with the new autopilot system on HB-PNP is given in **Figure 180**.

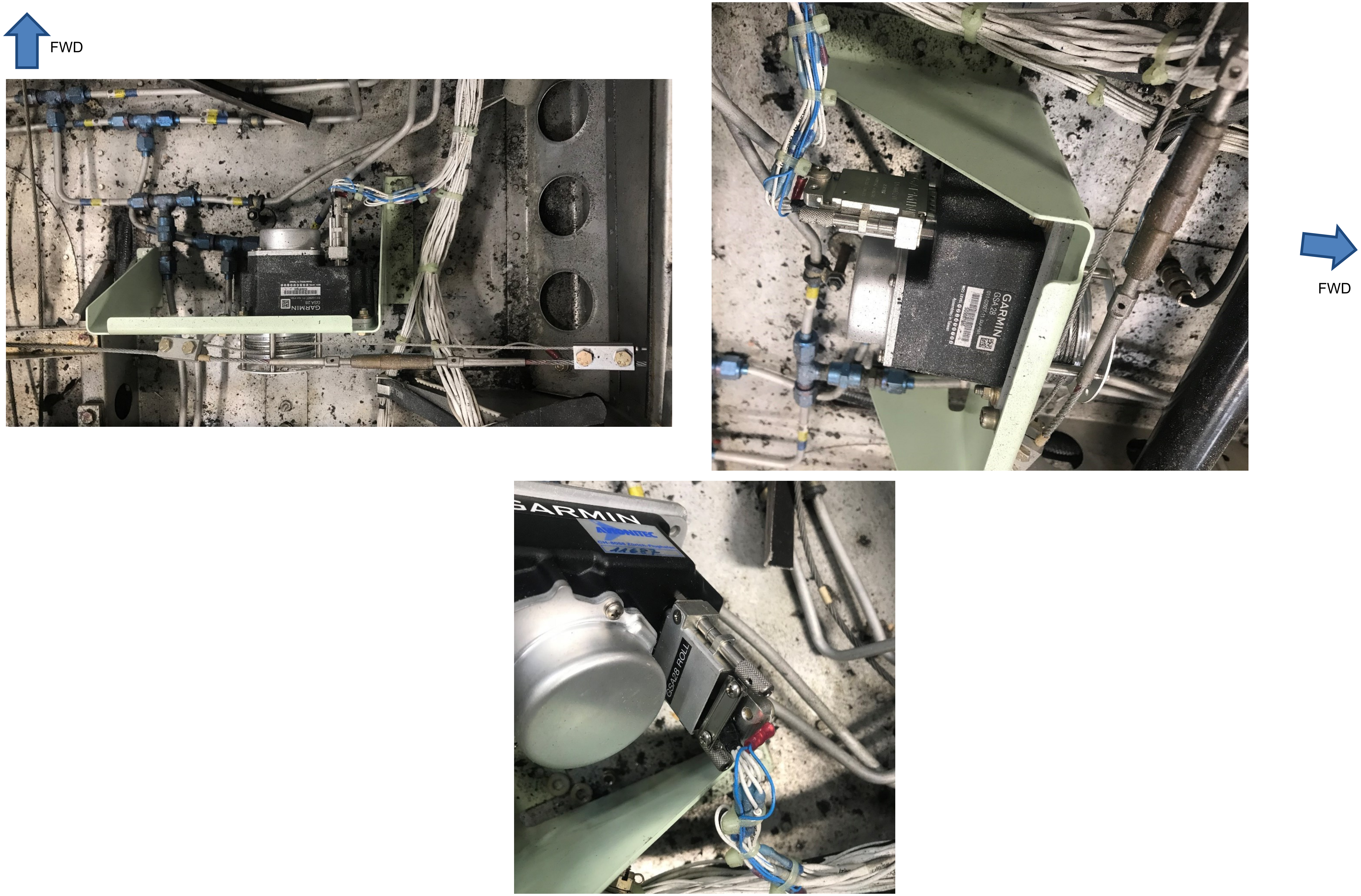


Figure 175: Servocontrol on roll axis
Source: BEA

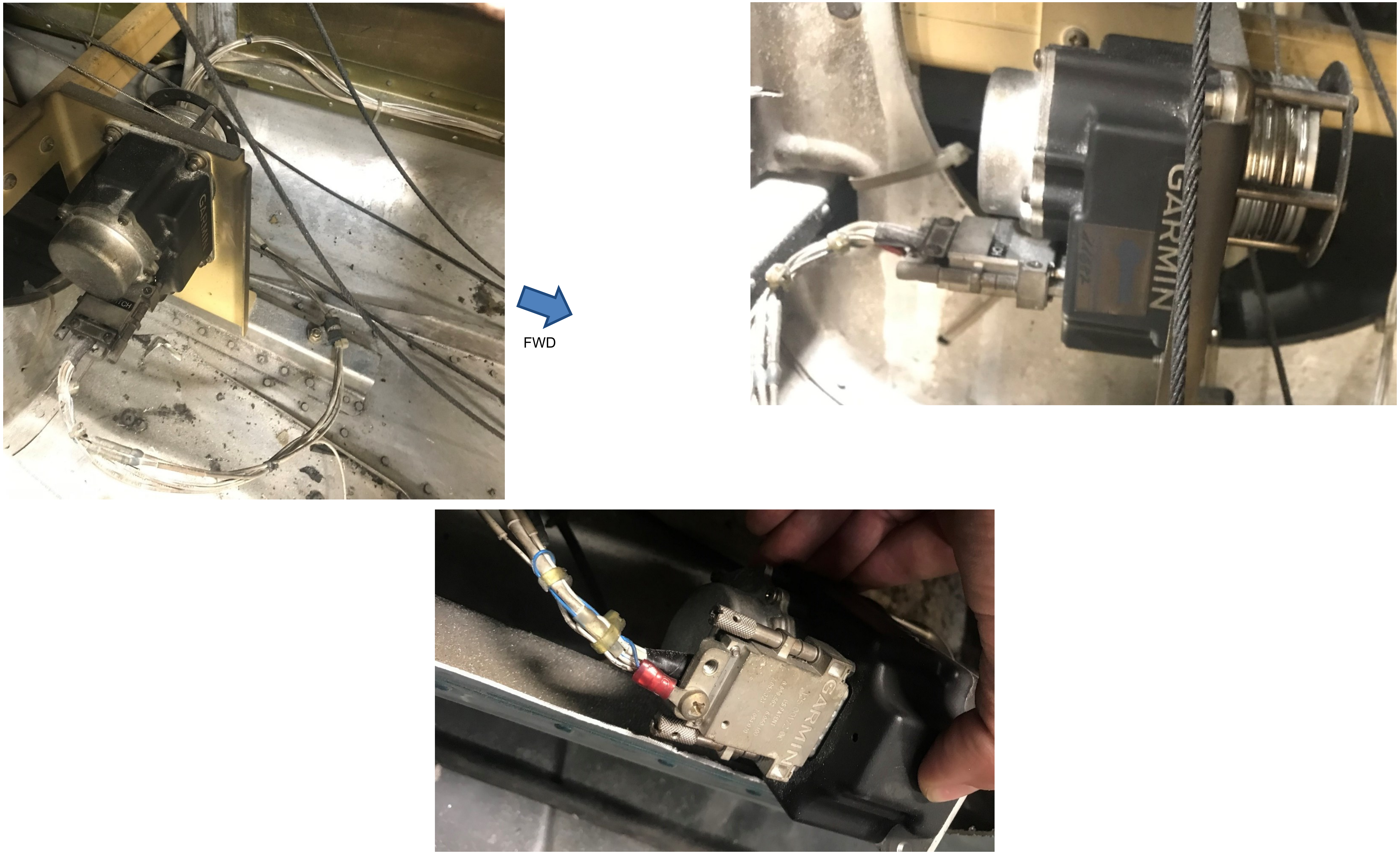


Figure 176: Servocontrol on pitch axis
Source: BEA

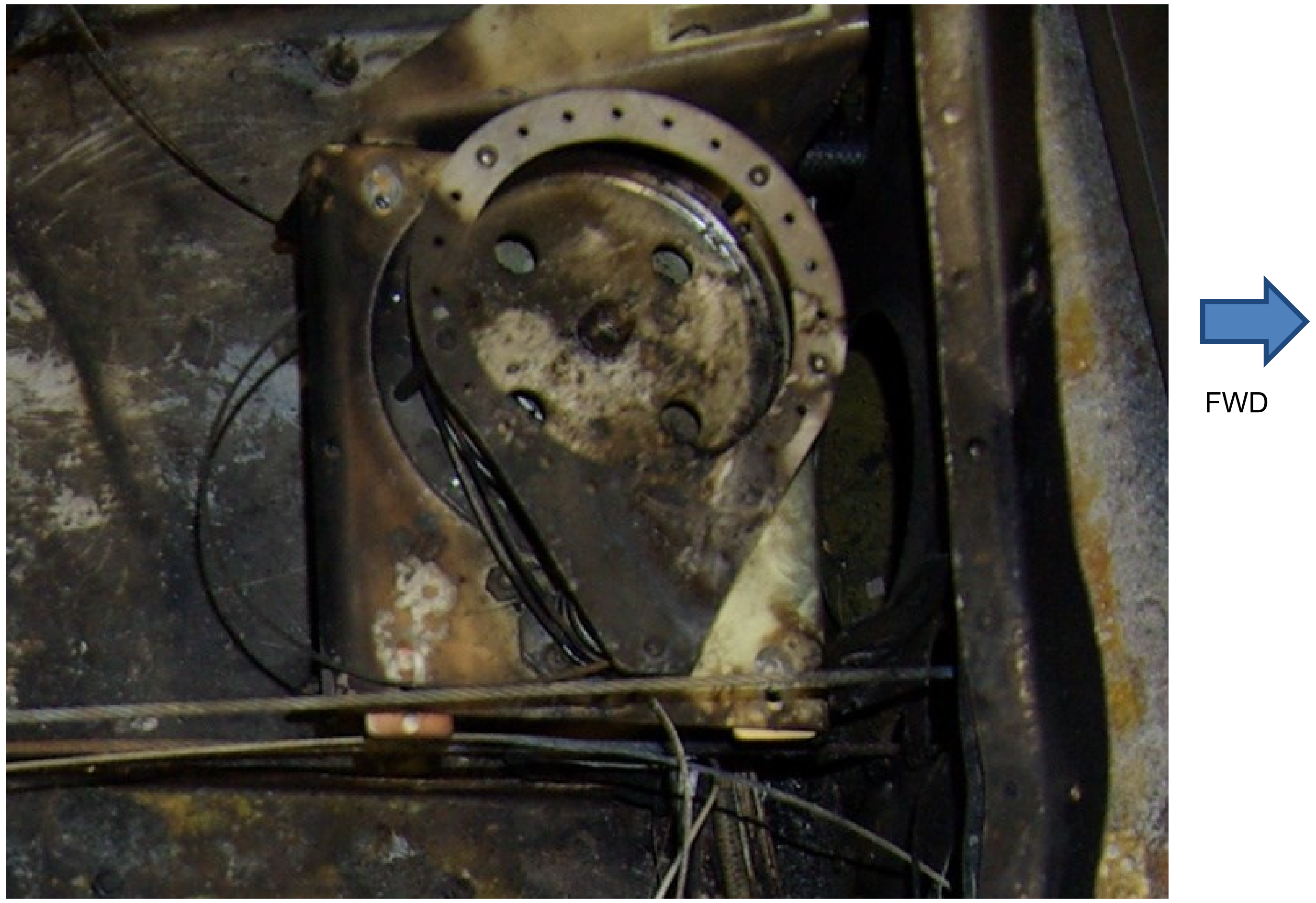


Figure 177: Servocontrol on pitch axis trim
Source: BEA

Garmin G5

Garmin GFC 500

Figure 178: Interfaces (Garmin G5 and Garmin GFC 500) on instrument panel
Source: BEA

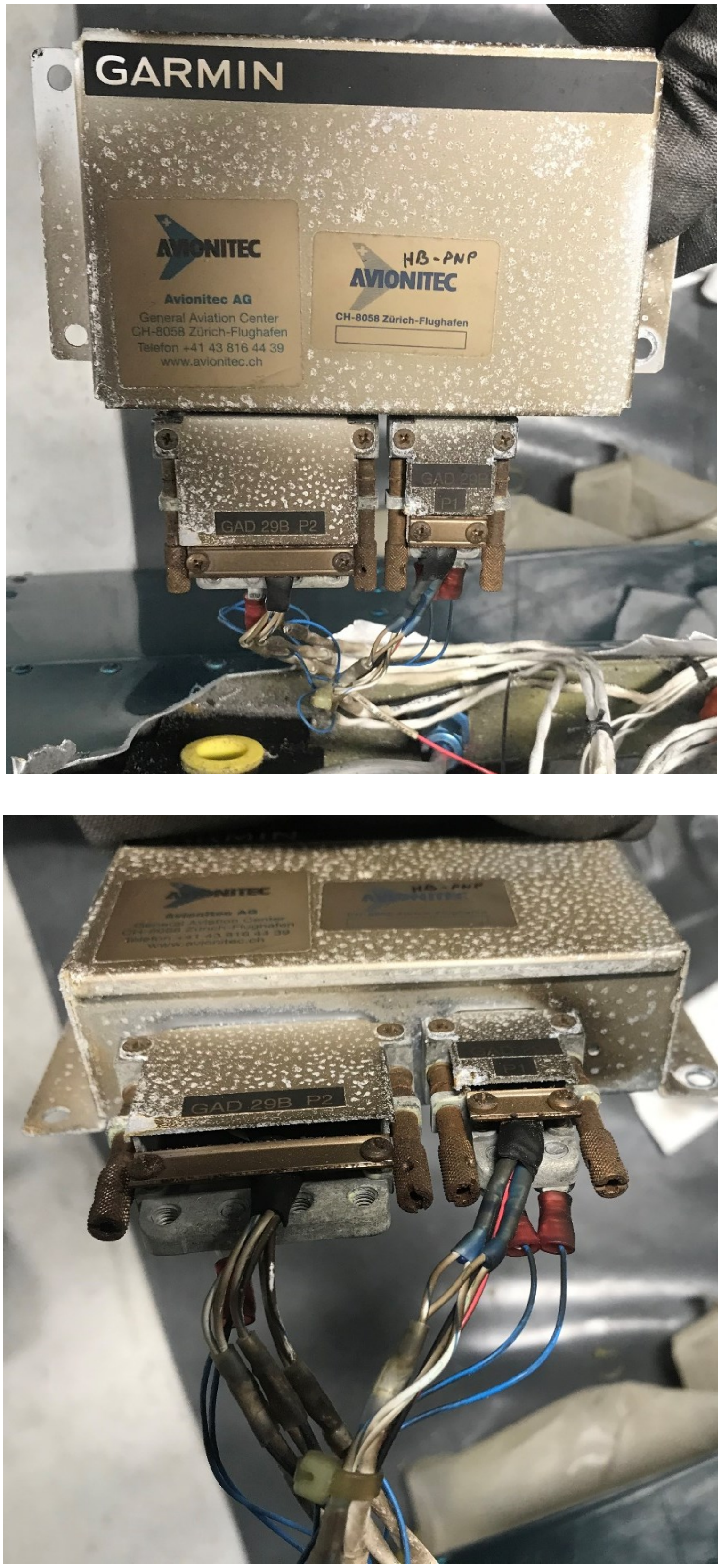


Figure 179: Garmin GAD 29 B adapter
Source: BEA

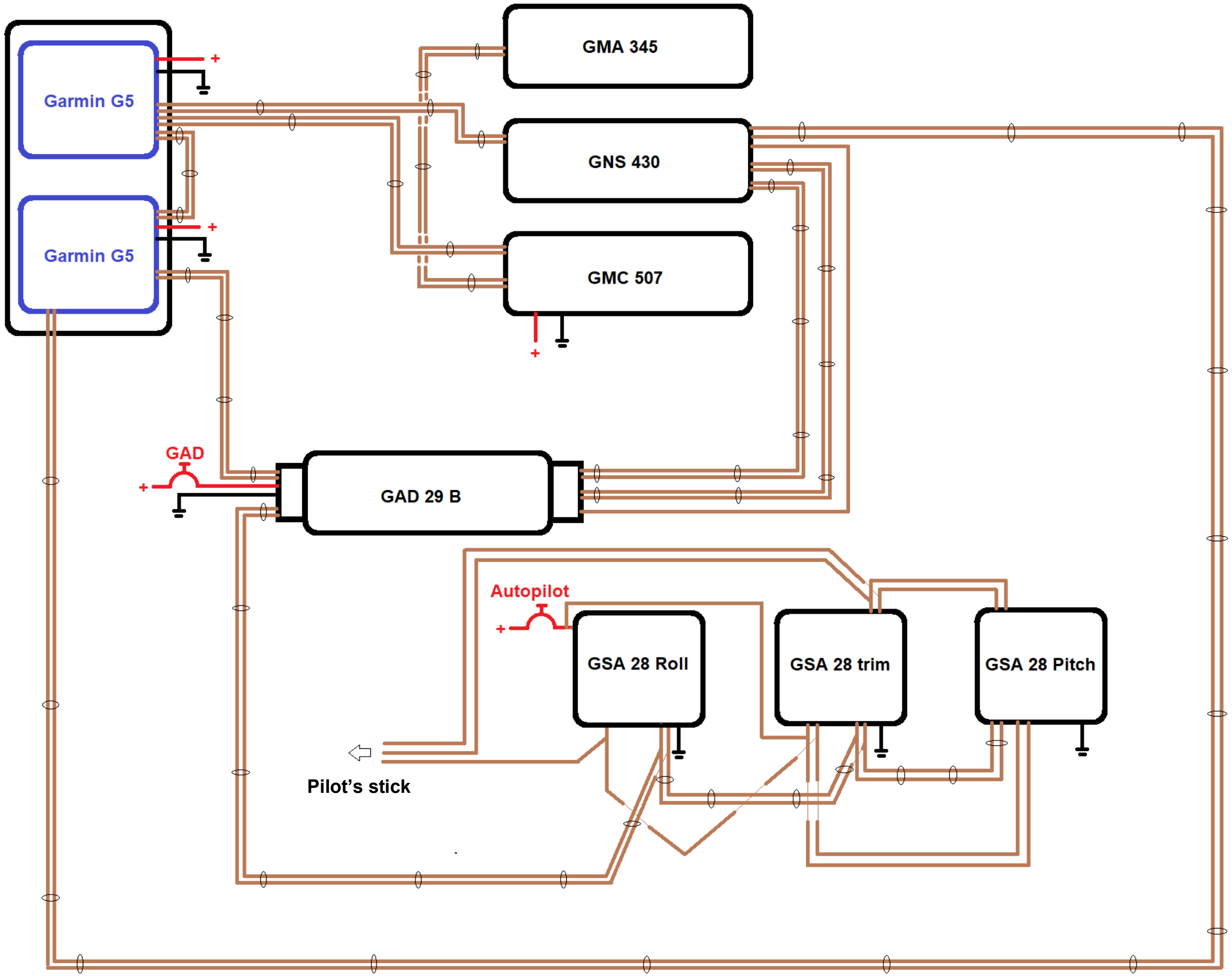


Figure 180: Electrical system associated with autopilot on HB-PNP (system drawn by the BEA as identified on HB-PNP)
Source: BEA

2.6.3. Conclusion of phase D

The examinations carried out on the system associated with the autopilot did not reveal any damage.

3 - CONSULTATION OF MANUFACTURERS AND EXPERTS

3.1. PIPER – aeroplane manufacturer

Piper indicated that it had never heard of a similar case.

It made no comments regarding the BEA's hypotheses due to the numerous modifications made to the aeroplane which were not modifications approved by Piper but approved by the FAA in the scope of STCs.

3.2. Concorde Aircraft Batteries – battery manufacturer

Concorde Aircraft Batteries mentioned an occurrence which took place in France (Perpignan) on 20 March 2018. This occurrence concerned the Cirrus SR22 G3 registered F-HPVP. It happened on the ground. The BEA did not open an investigation.

However, examinations were carried out by the judicial authorities who mandated an expert. The latter shared his findings with Concorde Aircraft Batteries. The BEA was able to exchange with this expert.

The photos transmitted to the BEA show a substantially damaged battery. This damage was visually similar to that found on the battery of HB-PNP. In particular, singular damage to the positive terminal of the battery can be noted.

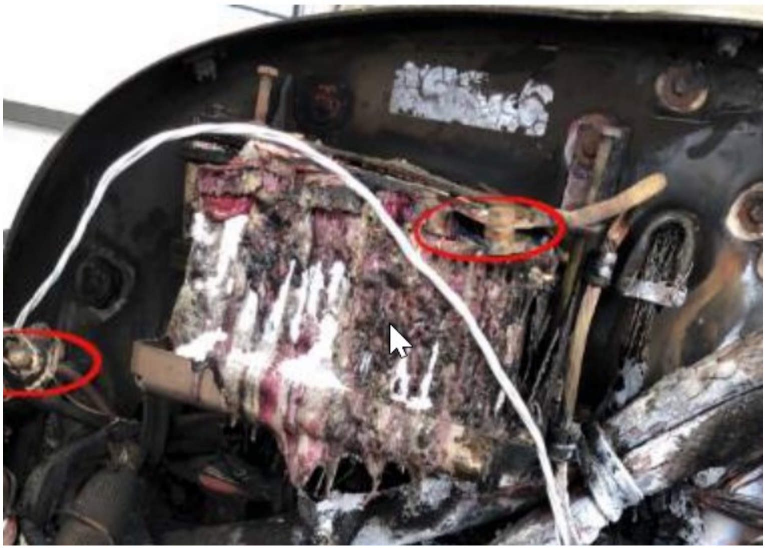


Figure 181: Destroyed battery of F-HPVP
Source: Judicial authority expert

According to the information provided by Concorde Aircraft Batteries, this damage to the battery terminal is thought to be the consequences of excessive overheating of the conductor connected to this terminal due to the circulation of a high current, or external overheating. According to the expert mandated by the judicial authorities, signs of a short-circuit were identified on the conductor associated with the battery.

On HB-PNP, according to Concorde Aircraft Batteries, the damage found on the positive terminal of the battery was effectively the consequences of abnormal overheating of the conductor fixed to this terminal.

3.3. Grenoble CEA

The Grenoble CEA has a department which carries out examinations of “conventional” batteries.

Based on the work carried out on an identical battery to that on HB-PNP, thermal runaway of the battery on HB-PNP cannot be ruled out. However, there are no signs on the battery showing thermal runaway. When this kind of phenomenon occurs, the most common sign is swelling of the battery which was not observed on the battery of HB-PNP.

The CEA expert also specified that overheating of the positive terminal of the battery due to an abnormally high current circulating for a long period was possible. He indicated that for a terminal with a lead section of 133 mm², given that lead has 12 times more resistance than copper (with a safety limit of around 5 A/mm²), a current safety limit at the terminal of about 56 A could be expected.

4 - CONCLUSION

4.1. Reminder of statement made by pilot of occurrence flight

The pilot specified to the BEA that for the accident flight, he had had to make several attempts to start up the engine.

After a flight time of around 15 min, the pilot observed a load indication for the ammeter close to 70 A which corresponded to the needle being close to the limit. This load value was a lot higher than the nominal values of this flight phase of between 10 and 30 A.

The pilot cut off the alternator and observed that the ammeter then indicated a value of 0. On switching the alternator back on, the ammeter once again indicated a constant value of around 70 A. The pilot and his passengers presumed that the battery was discharged and decided to continue with the planned flight programme. This excessive amperage was therefore maintained for about ten minutes.

4.2. Results of examinations

The most substantial damage was located aft of the luggage hold. The main components present in this area were the following:

- cockpit fan;
- hydraulic pump;
- electrical servocontrol;
- battery;
- several electrical contactors;
- diode device;
- ground power connector;
- several converters.

On these components, the main singularities were identified on the battery and the diode device. The diode device was next to the positive terminal of the battery.

On the battery:

- the very degraded state of the positive terminal of the battery: the brazed area ensuring the mechanical connection between the two main components of this terminal had melted (melting temperature of brazed area between 183 and 191°C);
- the damage to the cable connected to this positive terminal (the partial melting of the insulation whose melting temperature is between 254 and 279°C was observed) and the substantial force required to bring the end of this cable in line with the battery post accident;
- the melting of the upper part of the battery without any particular swelling or deformation.

<u>*On the diode device:*</u>

- the destruction of the ABS protective cover;
- the melting of the associated metal cooler (made of aluminium alloy with a melting temperature of around 700°C) at the point where it is attached to the aeroplane floor;
- the damage to the phenolic shield which normally insulates the metal cooler from the aeroplane floor. This damage is greater at the positive terminal of the battery (which does not correspond to the melting area of the cooler);
- the separation of the diode device into two parts due to the melting of a brazed area,
- the four identical cells making up the device in a short-circuit.

The other components had damage which seemed to be the result of the external overheating that occurred during the event.

The damage found in the rest of the cockpit was the consequence of contact with the hot smoke that circulated in the cockpit.

The functionality of the following components in the onboard electrical system was checked and validated:

- alternator;
- alternator control;
- contactors;
- controls on instrument panel.

The examinations carried out on the autopilot system did not bring to light any singularity which could explain a malfunction of one or more components during the occurrence flight.

The examinations carried out on the engine start-up system did not bring to light any singularity which could explain a malfunction of one or more components during the occurrence flight.

4.3. Conclusion regarding work carried out

The value of close to 70 A read by the pilot on the onboard ammeter during the flight indicated that the demand on the power system was abnormally high. This value indicated the presence of an abnormally high amperage in the electrical system for several minutes and as a consequence, of substantial local overheating. These phenomena probably caused the onboard fire.

The alternator and alternator control were tested and found to be functional after the accident which shows that this abnormally high amperage measured in the electrical system was very probably not due to a fault on these components.

The fault in the system is located in the environment comprising the battery and diode device.

The following two elements were identified with respect to the battery and its direct environment during the investigation:

- Use of a Concorde RG-35AXC battery (STC SA01147WI without embodying STC SA01582SE, STC SA4008NM and adapters for the battery in question[26]).

  It should be noted that embodying STC SA01582SE and STC SA4008NM and adapters was not mandatory when using a Concorde battery.

  Concorde Aircraft Batteries informed the BEA that STC SA01147WI had been embodied on around 3,000 aircraft without encountering any in-service difficulties.

[26] The various documents consulted by the BEA do not stipulate that it is mandatory to embody STC SA01582SE, STC SA4008NM and adapters when embodying STC SA01147WI.

- The substantial stresses in the cable connecting the master solenoid to the positive terminal of the battery due to the size of the cable, its section and shape after assembly (cable forming part of STC SA 3531NM). The BEA does not have any factual data to support the hypothesis of the terminal possibly being damaged due to mechanical loads from the cable.

  Furthermore, Concorde Aircraft Batteries informed the BEA that they did not share this questioning and indicated that the strength of the battery terminal was sufficient and that they had never recorded a failure of this type.

  As a consequence, the BEA did not study this assembly in more detail.

The following elements were observed on the diode device and its direct environment during the investigation:

- The local melting of the metal cooler in line with an attaching part which could be explained by contact between this attaching part and the cooler when the device was powered, generating a short-circuit;

  On 1 November 1978, Piper published Service Bulletin No 623 which specified the importance of positioning an insulating washer between each attaching element and the cooler. If this is not the case, it is indicated that a short-circuit is possible leading to a demand for maximum output from the alternator. Piper specified that Service Bulletin No 623 applied to various Arrow models manufactured between 1975 and 1978. PA-28RT-201T was manufactured between 1979 and 1988, which is outside of the dates indicated in SB 623, even though the diode device is similar to previous versions.

  The Piper maintenance manual applying to model PA-28RT-201T does not specify any maintenance requirement regarding the diode device.

  The observation of an assembly on an identical aeroplane found cracking in two of the four insulating washers. Damage to the insulating washers preventing a short-circuit described in Service Bulletin No 623, is possible.

Concorde Aircraft Batteries carried out numerous tests which showed that:

- local overheating of the battery terminal does indeed result in the terminal separating into two parts but with damage that was different to that found on the battery of HB-PNP;
- local overheating of the battery terminal does not lead to an incipient fire, the damage remains local and does not spread;
- incorrect insulation at the attaching point of the diode device leads to local melting of the diode device as observed on HB-PNP;
- intense overheating at the attaching point of the diode device due to incorrect insulation leads to the ABS cover quickly igniting with large flames;
- several minutes are required before the ABS cover is completely burnt.

As a result, a very likely hypothesis has emerged to explain the start of the damage and the fire that subsequently developed on board:

The electrical contact between an attaching item of the diode device and the metal cooler associated with this device led to a permanent short-circuit.

This short-circuit resulted in the aeroplane alternator providing maximum power in the power system for several minutes. The maximum amperage was moreover, observed by the crew during the occurrence.

This short-circuit also led to intense local overheating of the diode device in the short-circuit area and then the nearly-instantaneous ignition of the ABS cover. The flames from the ignition of the ABS cover contributed to the damage found on the battery, and its positive terminal in particular. The fire then spread in the aeroplane.

The publication of SB No 623 shows that faults had already been encountered on the diode device with major consequences on the aircraft.

The observations and tests carried out seem to exclude a primary fault on the battery.

A final point that was not examined by the BEA concerns the consequences of using a ground power unit on the accident aeroplane equipped with a damaged battery. The BEA questions what the amps in the electrical system might have been in such a situation and the possible exacerbation of prior damage to the positive terminal of the battery. Concorde Aircraft Batteries informed the BEA that there were no consequences in using a ground power unit on an aeroplane if the start of the phenomenon is associated with the diode device. If an aeroplane is powered by a ground power unit without the engine running, the battery is recharged directly; the part of the system including the diodie device is not powered.

Appendix 1 Calendar of examinations

Dates **Locations**	**Operations**
29 and 30 July 2020 Bâle-Mulhouse airport	Visual examination and preparation of wreckage for its transfer to the BEA
September and October 2020 BEA	Preparation of aircraft for examination of its electrical system
November 2020 BEA	Examination of most damaged area and electrical system
23 November 2020 BEA	Assessment with a fire investigation expert
24 November to 4 December 2020 BEA	End of general observations on wreckage
January 2021 BEA	Laboratory examinations of battery terminals and diode device
10 May 2021 DGA TA Toulouse	Assessment with “fire” and electricity experts
25 May 2021 BEA	Assessment with expert in damage to F-HPVP
25 May to 6 June 2021 BEA	Work on power system
25 June 2021 ENAC Castelnaudary	Alternator and alternator control test
4 to 6 October 2021 BEA	Examination of components associated with STC SA 3531NM
11 November 2021 BEA	Visual assessment of HB-PPB battery
15 November 2021 Sematec	Tomographic examination of HB-PPB battery terminals
10 December 2021 INES CEA Chambéry	Validation of investigation process with respect to HB-PPB battery

Dates **Locations**	**Operations**
15 January to 14 February 2022 INES CEA Chambéry	Tests and examinations of HB-PPB battery
4 February 2022 BEA	Examination of engine start-up electrical system
16 February 2022 Sematec	Tomographic examination of starter solenoid
7 April 2022 ENAC Castelnaudary	Starter test
19 April 2022 BEA	Destructive examination of start-up switch
19 April 2022 BEA	Destructive examination of starter solenoid
1 September 2022 BEA	Additional examinations of diode device

BEA

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