Failure of the SS520 No. 4 experiment and measures to be taken

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6. Summary

Terminology

Terminology	Abbreviation
Telemetry	TLM
Command decoder	CMD
Miniature radar transponder	MRT
On-board battery	28 V battery
Sun aspect sensor	SAS
Inertial measurement unit	IMU
Rhumb line controller	RLC
On-board computer	OBC
Data acquisition unit	DAU
Pyrotechnic controller	PYRO
Rhumb line controller	RCS
Active nutation control	ANC
3rd-stage flight-path monitor	FPM
Fault tree analysis	FTA

1. Overview of experiment results

1-1 Experiment plan
- 1-2 Outline of the experiment
- 1-3 Spacecraft flight trajectory and impact point
- 1-4 Sequence of events

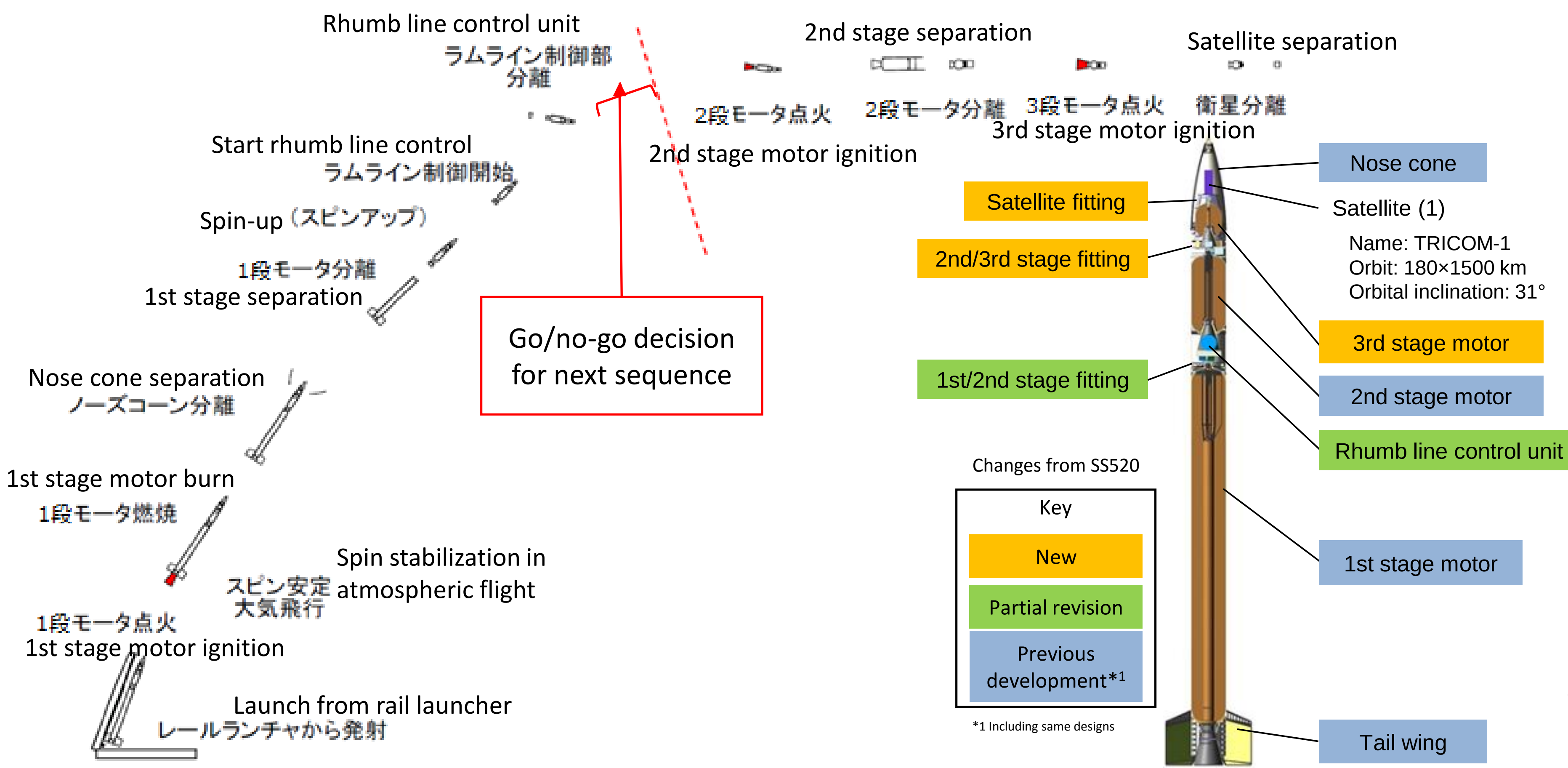
1-1. Experiment plan

Flight experiment plan

Purpose of the experiment: Development of rockets and satellites using civil engineering technology; demonstration of a microsatellite of about 3 kg mass.

Rocket description: The SS520 No. 4 is a three-stage rocket that is a modification of the SS520 two-stage rocket.

Flight plan: The 1st stage flight is performed with tail-wing spin stabilization. After 1st stage motor separation, attitude is changed by rhumb line control. Following determination of flight safety, 2nd stage motor ignition command is transmitted from the ground and subsequent sequences continue. Satellite is separated after completion of the 3rd stage motor burn out.



1-2. Outline of the experiment

Launch conditions

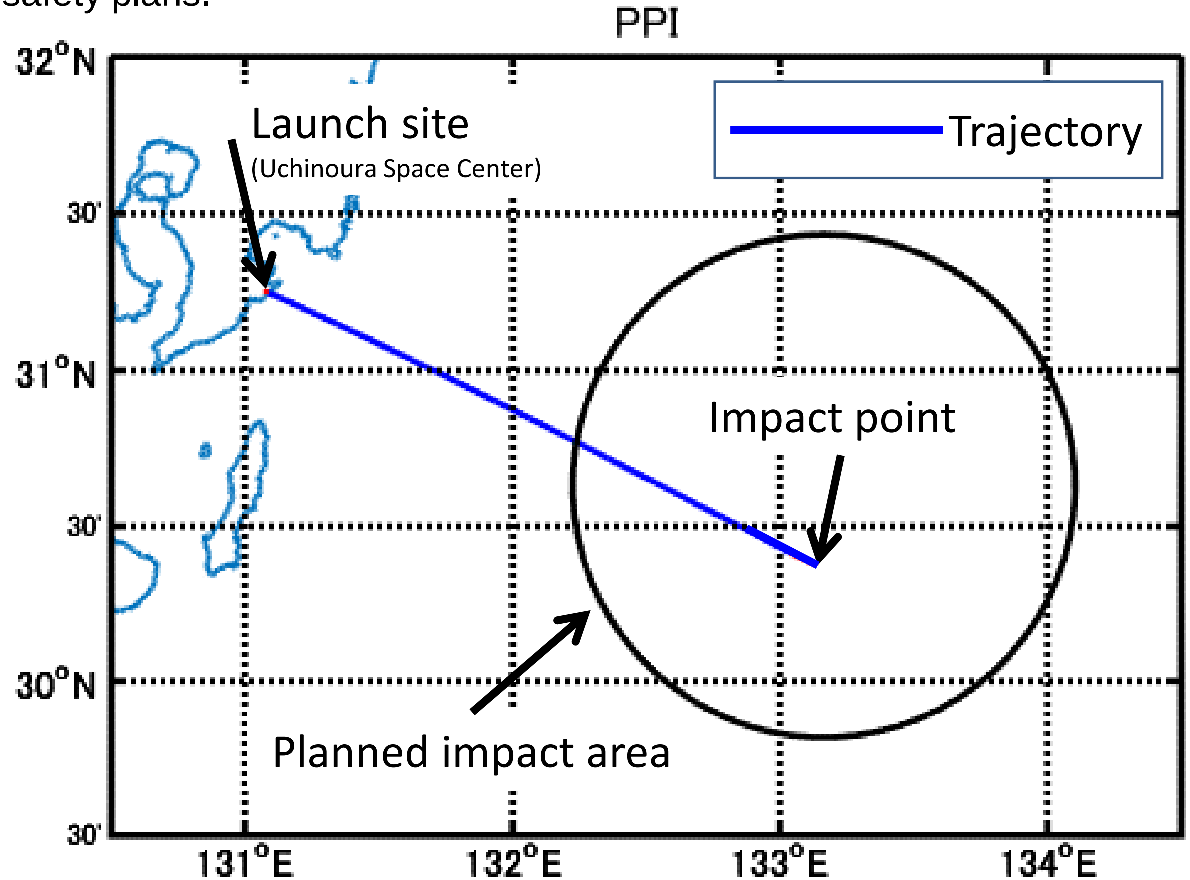
- The launch experiment was conducted at 8:33 AM on 15 January 2017 [JST].
- Weather, wind direction, wind speed, and other atmospheric conditions were acceptable for launch.
- Time schedule started at 5:00 AM [JST]. Launch preparations (security measures including evacuation of residents) were implemented as planned.

Post-launch conditions

- The ignition time was as planned, and the overall system including ground systems for telecommunications and tracking such as the telemeter, radar, etc. were normal immediately after the launch.
- Data transmission from the telemeter ceased at X + 20.4 s after launch. Information on the data display screen of the control room was lost, and telemeter information could not be confirmed, even for the flight safety control system.
- At around the same time, answer-back from the command decoder (CMD) became 100% error.
- Because the subsequent data reception situation did not improve, we could not ascertain the flight status of the spacecraft, so the 2nd stage motor ignition command was not sent to ensure safety.
- Although the 2nd stage motor did not ignite, the radar maintained a lock-on state, so we continued tracking the spacecraft and confirmed that it impact in the predicted fall area.
- Because 2nd stage motor ignition was not performed, it was impossible to place the microsatellite TRICOM-1 into the predetermined orbit.

1-3. Spacecraft trajectory and impact point

- The rocket was launched at a launch elevation angle of 75.1° and an azimuth angle of 125.0°.
- According to radar tracking data, 1st stage flight was normal (vertex altitude 190 km, ground speed 0.918 km/s). Corrections to launch elevation angle and azimuth angle were properly implemented, based on wind observed by balloon and Doppler lidar.
- It is estimated that following 1st stage flight, nearly median values for the trajectory plan would have been achieved for 2nd and 3rd stages, had they occurred.
- The spacecraft landed in the ocean in the planned area, indicating that the launch was conducted according to safety plans.



1-4. Sequence of events

Time after ignition	Event	Success	Basis
X+0.0	1st stage motor ignition	○	Flight data, etc.
X+31.7	1st stage motor burn out	○	Visual records
X+53.0	Pyro valve release	△	
X+54.0	Initiate attitude command reception	×	No answer-back
X+62.0	Satellite separating system activation command	○	Satellite separation at X+450 s, as planned
X+67.0	Nose cone separation	○	Visuals, FPM, satellite data
X+68.0	1st stage separation	△	
X+73.3	Initiate rhumb line control	×	Radar reception level analysis
X+117.6	Rhumb line control complete	×	
X+121.2	Initiate active nutation control (ANC)	×	
X+145.0	ANC complete	×	
X+147.0	Rhumb line control unit separation	△	
X+150.0	Initiate time command reception	△	
X+157.0	Initiate go/no-go decision on 2nd stage motor ignition	△	
X+164.0	Initiate 2nd stage motor ignition signal reception	×	
X+180.0	2nd stage motor ignition	×	
X+235.0	2nd stage motor separation	×	
X+238.0	3rd stage motor ignition	×	
X+263.8	3rd stage motor burn out	×	
X+450	Satellite separation	○	Radio reception from satellite

Key
○: Successful
×: Unsuccessful
△: Unknown

2. Factual relations of anomalous events

2-1 Known facts
2-2 Event in TLM
2-3 Output anomaly of 2nd stage motor strain sensor
2-4 Analysis result of attitude change implementation
2-5 Reproduction test results (TLM event reproduction)
2-6 Other reproduction test results

2-1. Known facts

Summary time-series of events

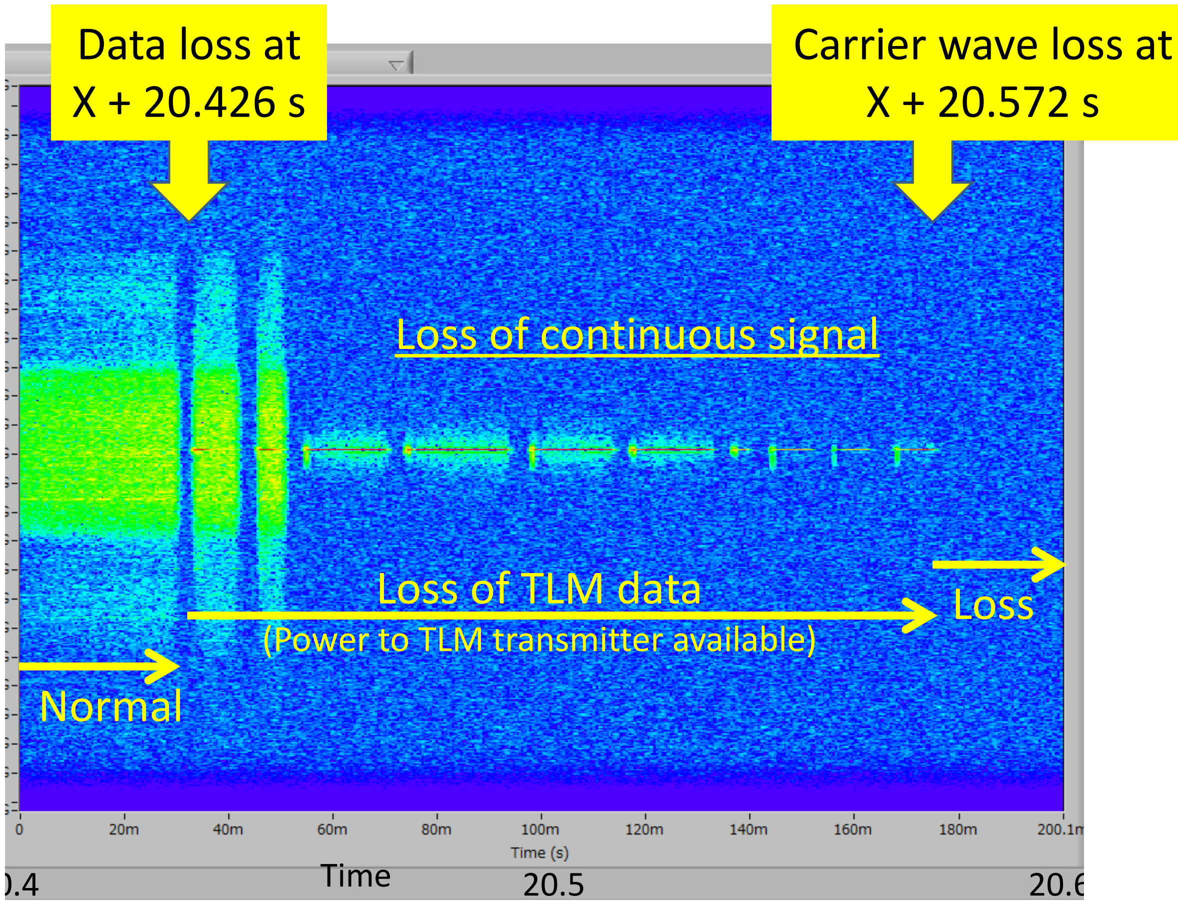
X-relative time [s]	Device showing anomaly	Phenomenon
X+20.015–20.020	2nd stage motor strain sensor	Anomalous output value. Sensor power supply system broken or ground fault (from analysis results).
X+20.426	TLM transmitter	Ten short interruptions of about 3 ms each occurred. Period between interruptions was 14.5 ms on average.
X+20.446	TLM transmitter	Data transfer interruption
X+20.572	TLM transmitter	Transmission interruption (lock off)
X+20.831	CMD	100% answer-back errors

Other analysis results

1. Attitude corrections by rhumb line control were not implemented.
    - Attitude change by rhumb line control not seen from variation in the radar reception level.

2-2. Event in TLM

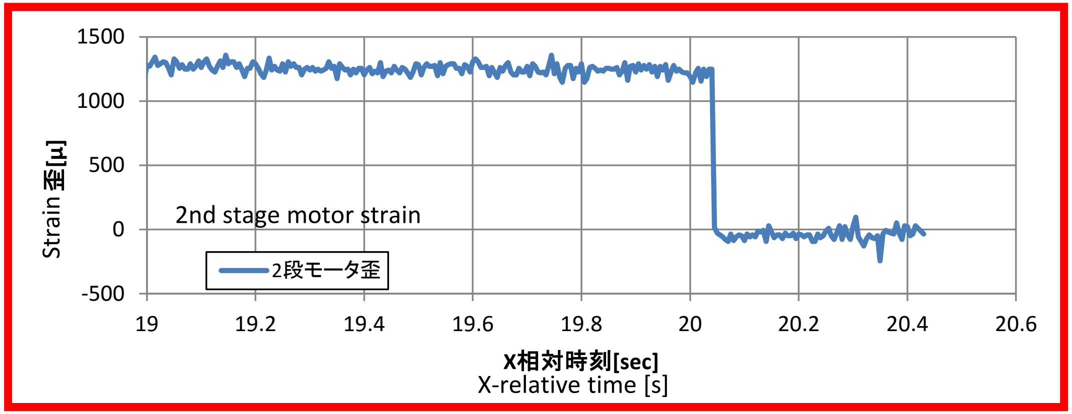
Received status of transmitted radio waves around X + 20.4 to 20.5 s

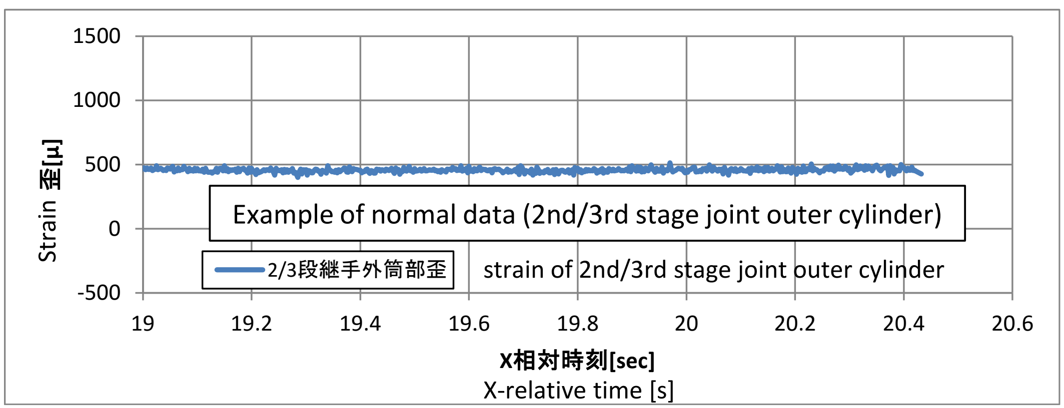


Change over time in TLM spectrum

2-3. Output anomaly of 2nd stage motor strain sensor

Recognized as necessary data for relation with other phenomena due to time proximity (X+20.015 to 20.020 s) to TLM data loss (X+20.426 s).





Cable duct between 2nd stage motor

2nd stage motor strain sensor

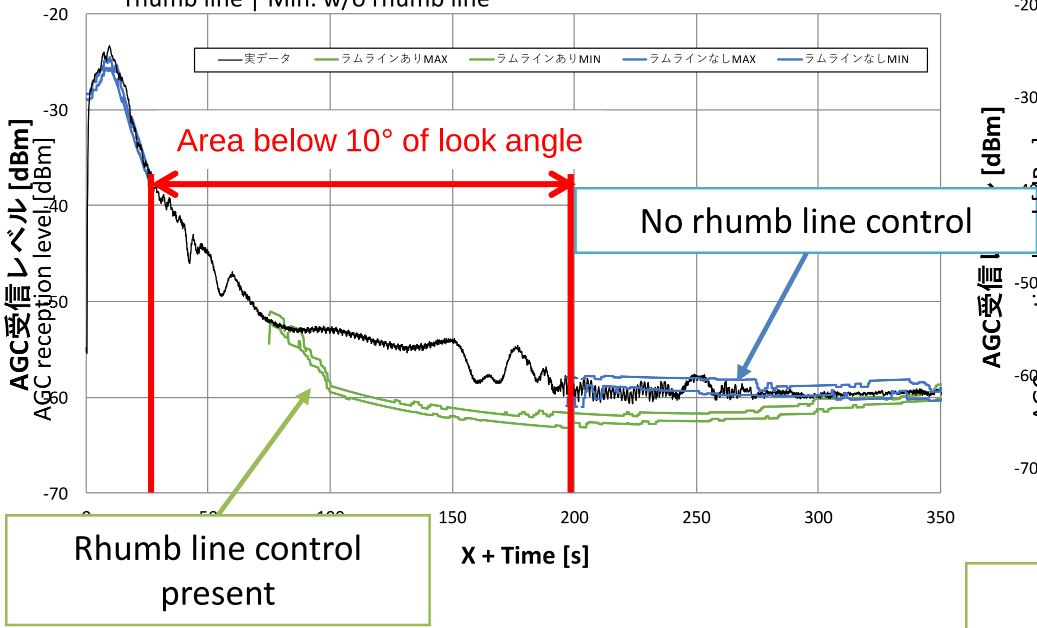
Reference strain sensor for bridge circuit

2.4. Analysis result of attitude change implementation

Reception level analysis for Miyabaru Precision Radar (Uchinoura) and Masuda Precision Radar (Tanegashima)

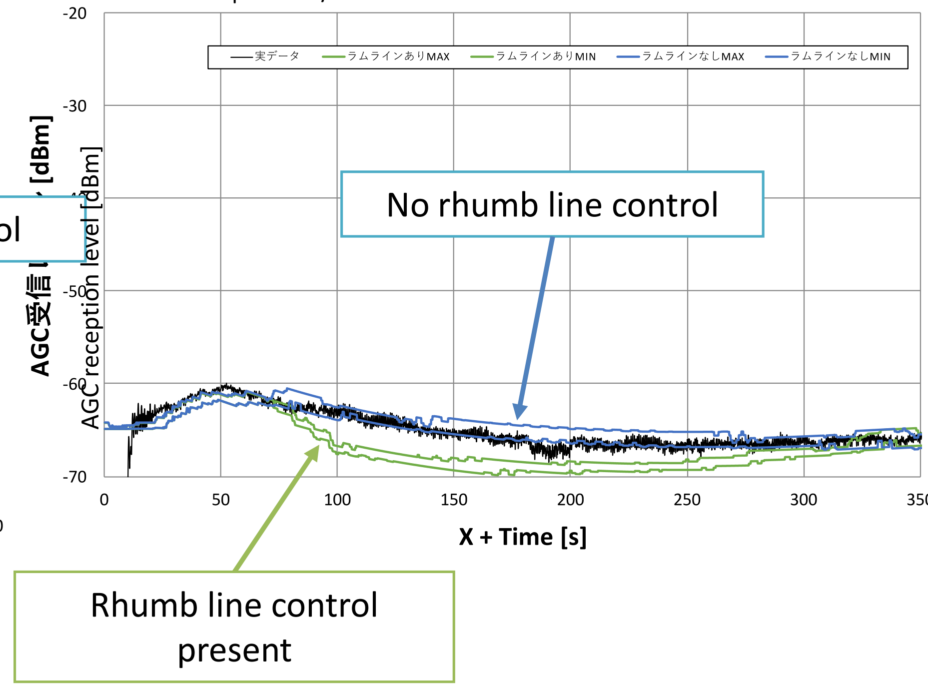
Flight data agree with the finding that attitude change by rhumb line control did not occur.

Actual data | Max. with rhumb line | Min. with rhumb line | Max. w/o rhumb line | Min. w/o rhumb line



Miyabaru Precision Radar

Actual data | Max. with rhumb line | Min. with rhumb line | Max. w/o rhumb line | Min. w/o rhumb line

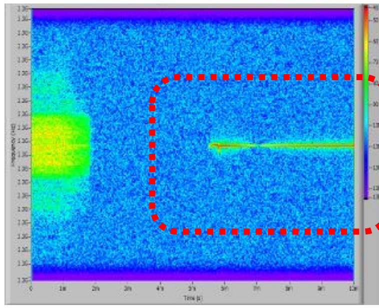


Masuda Precision Radar

2-5. Reproduction test results (TLM event reproduction)

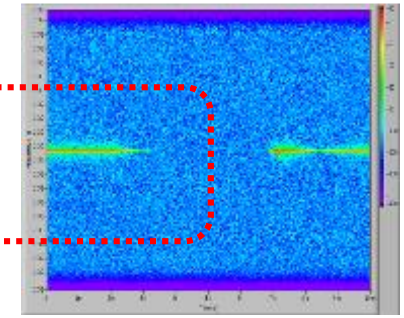
Based on features seen in the flight data, we investigated the spectrum waveform at TLM transmitter power ON/OFF.

Flight data



Revival behavior of transmissions

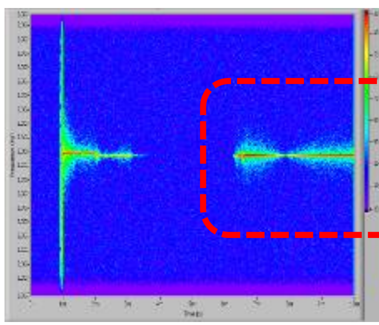
Waveform features



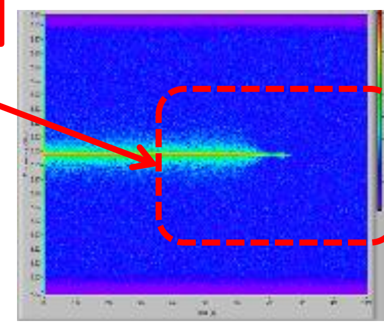
Interruption behavior of transmissions

Waveform features (reproduced)

Reproduction test



Waveform at power on



Waveform at power off

From the appearance of the reproduced spectrum waveform, it is possible that TLM transmitter power supply input was continuously intermittent.

2-6. Other reproduction test results

To clarify the cause of TLM data interruption events, the cause of failure in the power supply circuit of on-board equipment was extracted and the following tests were conducted.

1. Switch open fault test
   → Voltage profiles for the power supply and equipment were acquired with the power supply equipment switch turned off. We found that the power was turned off in a shorter time than the TLM data sampling time (5 ms).
2. Connector drop test
   → Power supply voltage profile due to connector drop was acquired. We found that the power supply was turned off in a shorter time than the TLM data sampling time.
3. Short circuit test
   → A short-circuit simulation test was performed on the load-side power input unit, and a voltage profile of each part was acquired. We found that the power supply was turned off in a shorter time than the TLM data sampling time.

From the above results, we conclude that no sign of power failure remained in the telemetry data.

From considerations including the above results, we have determined that continuous TLM signal interruptions were a result of a TLM transmitter power supply anomaly.

3. Analysis of the anomaly

3-1 Investigative review
3-2 FTA summary
3-3 Mounting system power supply system diagram (function block diagram)
3-4 Avionics power system diagram
3-5 Results of verification and test
3-6 Results of FTA factor analysis
3-7 Exterior appearance of cable ducts

3-1. Investigative review

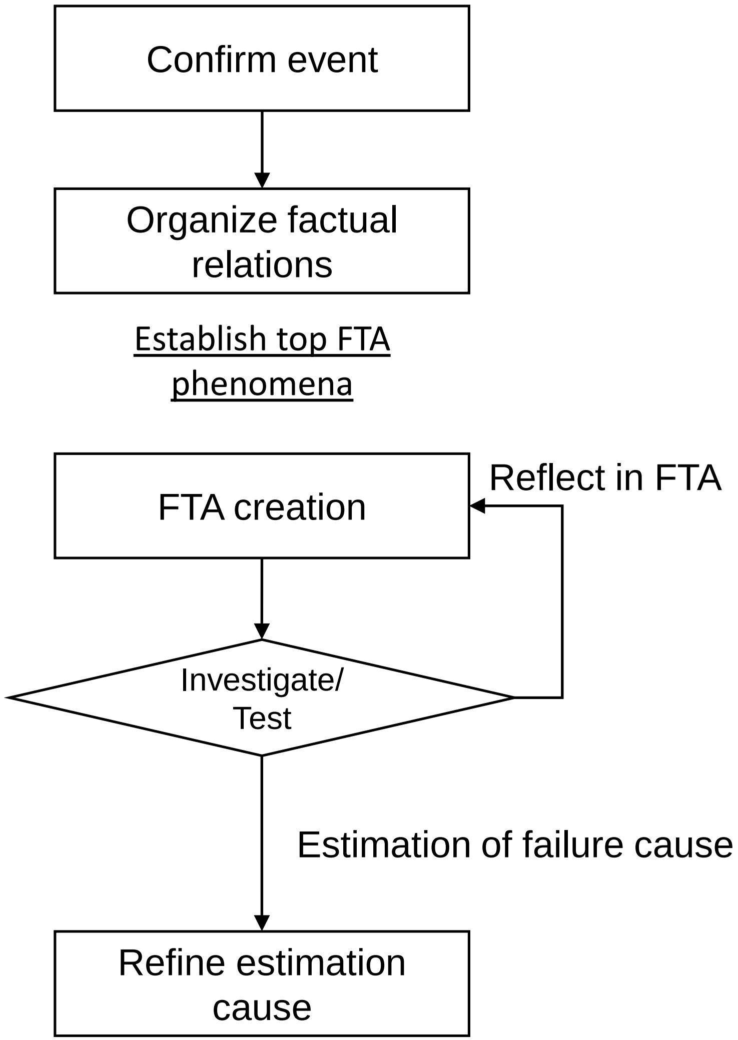
1. Confirmation of event

2. Organizing factual relations
   →Ascertain factual relations between flight data, ground system data, etc.

3. Fault tree analysis (FTA) of the failure
   →Extract presumed cause of failure from the ascertained factual relations

4. Investigation and testing
   → Of the estimated factors extracted by FTA, conduct element tests for those that cannot be bench confirmed. Reflect test results in FTA (evaluate the possibility of impacts)

5. Estimate scenario
   → Narrow down estimated cause from facts, verification results, etc.



*Discuss with countermeasures team

3-2. Fault tree analysis (FTA) summary

Top events established in FTA are as follows:

1. Loss of TLM data
2. Output anomaly of 2nd stage motor strain sensor

The following were extracted as a result of FTA analysis of factors causing these events.

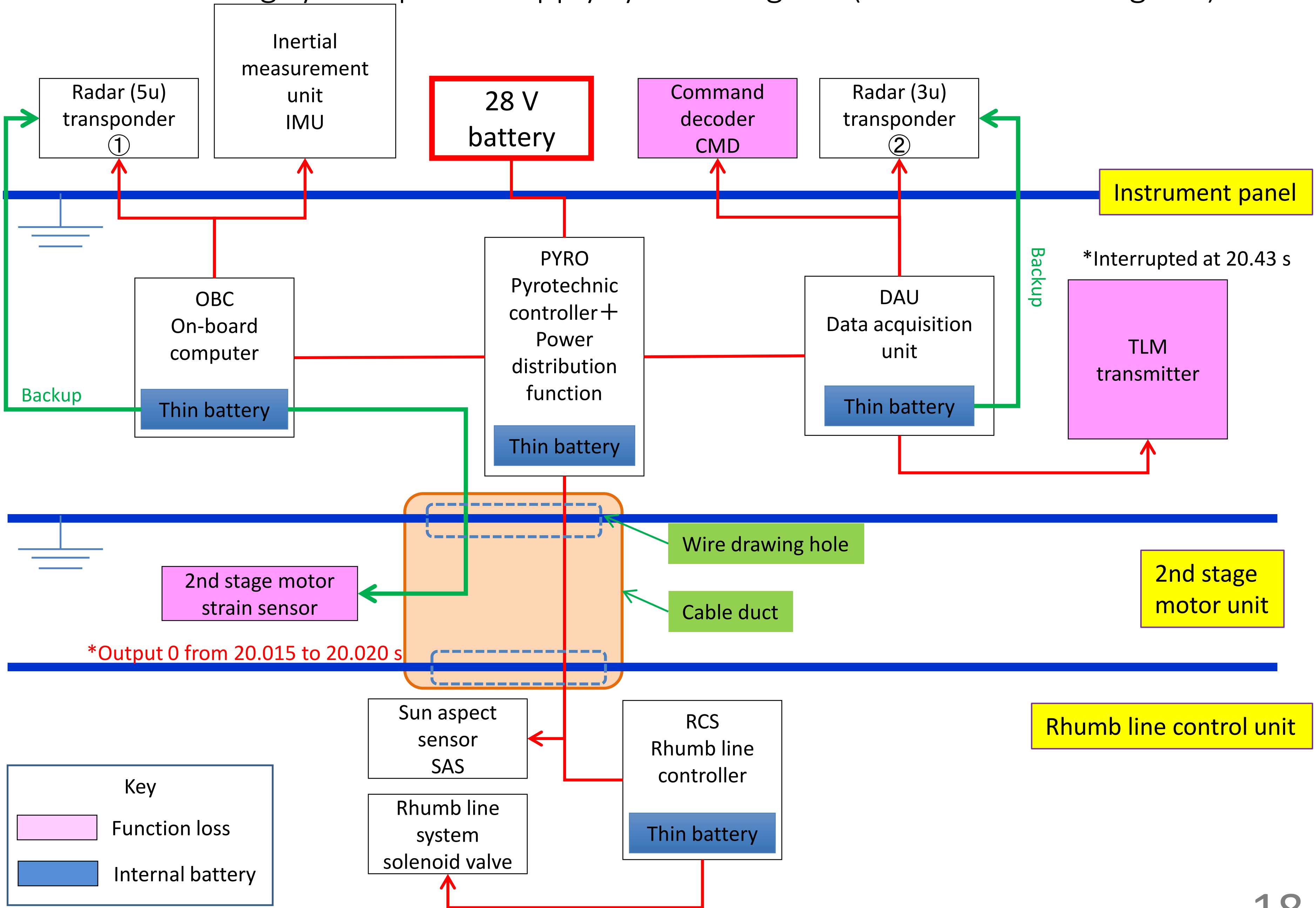
1) Disruption of TLM data was caused by a power supply system anomaly (from explanation in Section 2).

From FTA, the following factors were extracted:

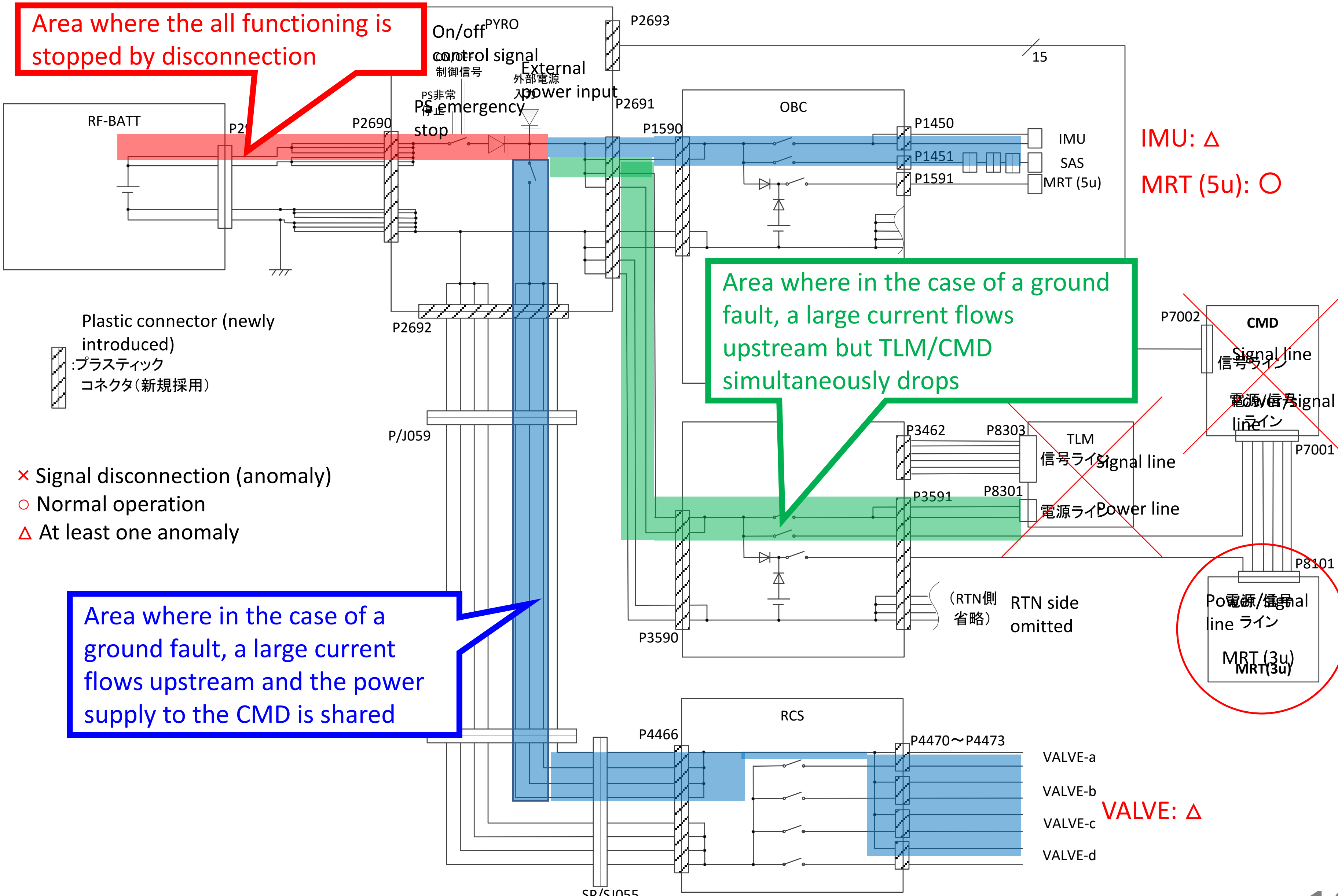
A) Part failure on the primary side of power regulator in a component
B) Short circuit, ground fault, or disconnection due to power line damage
C) Disconnection or breakage of a power line connector

2) From analysis of the sensor circuit, etc. the cause of the output anomaly of the 2nd stage motor strain sensor may have been disconnection or a ground fault in a power line.

3-3. Mounting system power supply system diagram (function block diagram)



3-4. Avionics power system diagram



MRT (3u) and MRT (5u) operated normally with power supply from the internal battery.

3-5. Results of verification and test

1. Ground fault effect check test for 28 V battery interface circuit

No anomaly in semiconductor switch (MOSFET) or ceramic capacitor.

2. Wire connector durability test

Even if the lock mechanism of the wire connector is partially damaged, it does not fall off due to tensile force or vibration.

3. Wire break test

Protective tape damage was detected at contact with metal parts.

4. Confirmation of avionics device internal controller program

No anomaly was detected in related software operations.

5. Influence of SAS internal failure

No anomalous propagation to other equipment due to SAS internal failure.

6. Internal failure of avionics equipment

Avionics equipment are controlled by programed software, so failure will not occur unless there is a hardware failure (see 4.). As MRT (3u, 5u) were normal, OBC was determined to be normal. This is unrelated to the 2nd stage motor strain sensor anomaly.

7. TLM anomaly investigation

→ TLM and CMD anomalies are unrelated to the 2nd stage motor strain sensor anomaly.

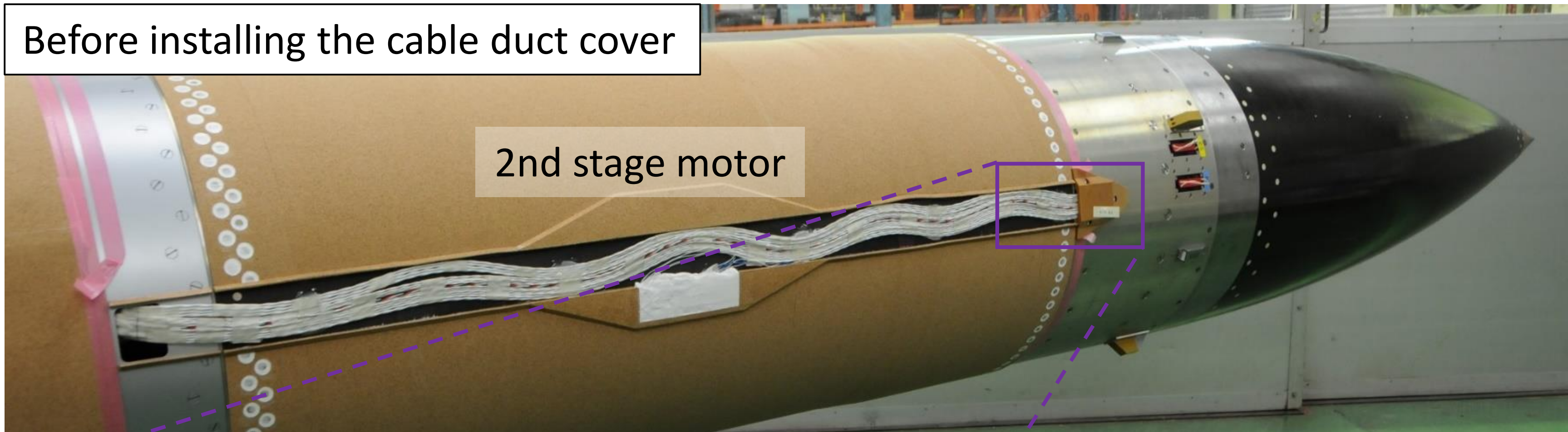
3-6. Results of FTA factor analysis

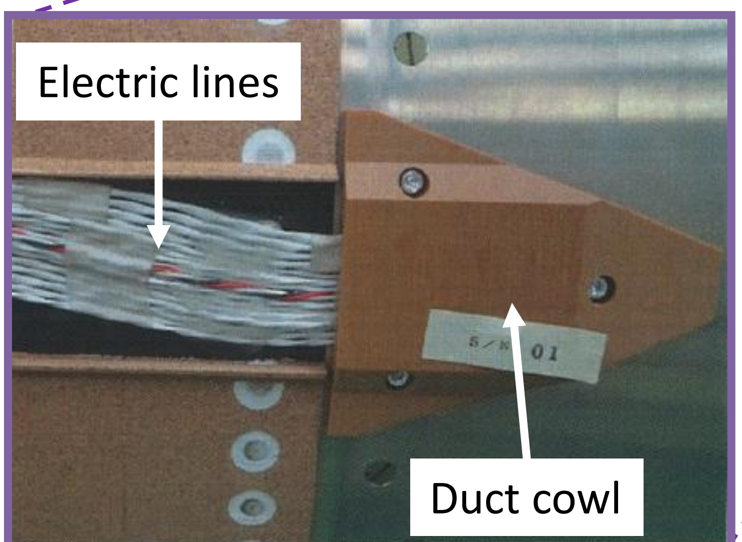
1. From the results of analysis of the event in the TLM transmitter, we recognized that there was an anomaly in the power supply system, and the possibility of a short circuit or ground fault was extracted (as described in Section 2).
2. The main factors for the top events by FTA were as follows:
   A) Part failure on the primary side of power regulator in a component
   B) Short circuit, ground fault, or disconnection due to wire damage
   C) Disconnection or breakage of a power line connector
3. Based on the following facts, the possibility that a malfunction occurred in the vicinity of the cable duct was extracted.
   1. A device powered by a 28 V battery with no backup power supply malfunctioned (loss of power).
   2. The circuit was configured such that a short circuit or ground fault occurs in one place in the 28 V battery power supply system will affect the entire device.
   3. An electric wire of the 2nd stage motor strain sensor and the power line of a 28 V battery system were close to the inter-stage routing, which crossed the 2nd stage motor. (⇒ Relevance to the event)
   4. An electric wire was attached passing through a hole in the metal structure (inter-stage joint structure) used for electric grounding.

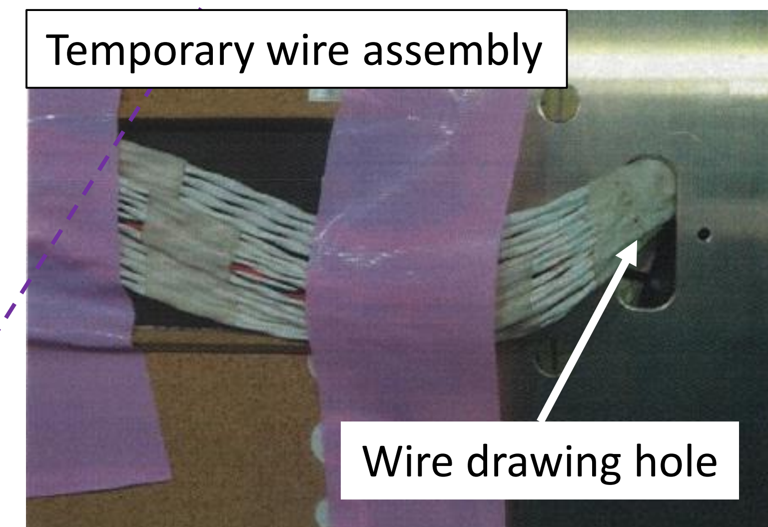
3-7. Exterior appearance of cable ducts

Cable duct: Protector of electric line used for electrical connection between the 1st/2nd stage and 3rd stage.

- Because it is necessary to lay electric wires outside the 2nd stage motor casing, those wires are protected against the flight environment (acceleration, aerodynamic force, and aerodynamic heating).





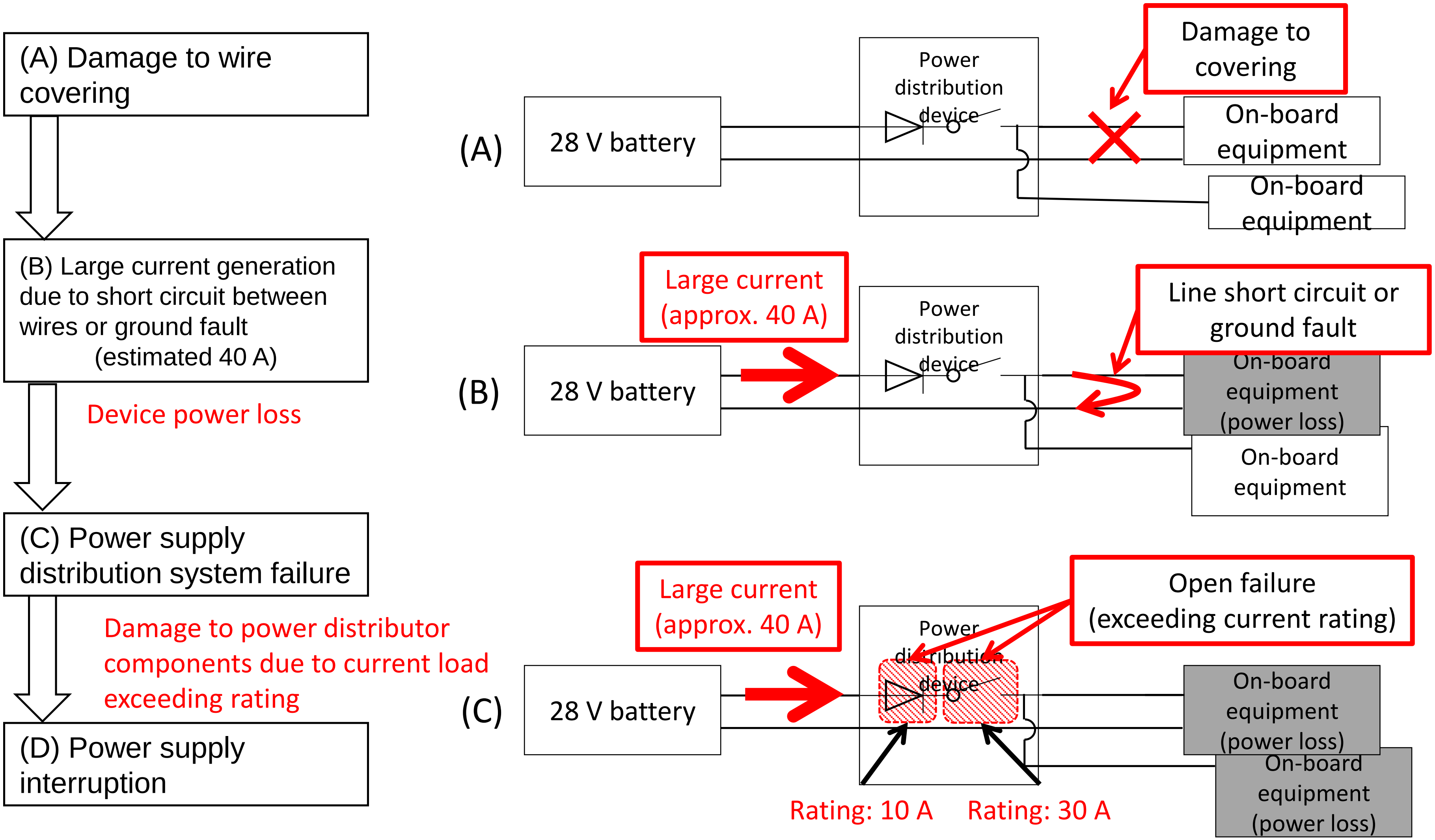


4. Probable causes

4-1 Estimated mechanism of power supply anomaly
4-2 Estimation of causes
4-3 Rationale for estimation of anomaly location
4-4 Probable cause (1)
4-5 Probable cause (2)

4-1. Estimated mechanism of power supply anomaly

From the FTA-based factor investigation, the mechanism leading to the power source anomaly is estimated as follows.



⇒ (D) As a result of a short circuit, a component failure or disconnection in the power distribution device occurred, eventually disconnecting the power supply function.

4-2. Estimation of causes

From the confirmed events and the occurrence factors extracted by FTA, **we estimate the damaged part and causes to be as follows, considering the relevance of the occurring events.**

Events
A) TLM data interruption, CMD answer error (28 V power supply anomaly)
B) 2nd stage motor strain sensor measurement anomaly

Related to event occurrence (occurrence of short circuit or ground fault)

1. Electric wire damage in the cable duct periphery

(1-1) Cover damage near the wire drawing hole

(1-2) Wire exposure and covering damage due to cable duct damage

*When A and B are considered to be independent events

No relation to event (power loss)

2. Damage to internal and external equipment

(2) Simultaneous damage or disconnection of two line connectors

4-3. Rationale for estimation of anomaly location

From the following rationales, we estimated the point of anomaly occurrence.

(1 - 1) Cover damage near the wire drawing hole
- Both the 28 V power supply system and the 2nd stage motor strain sensor system are included in the wire bundle (considering the relevance of the event).
- The part in question (wire drawing hole) is a metal structure and electrically grounded, and the electric wire bundle (with partial glass cloth tape protection) is in direct contact.
- There is a possibility that the electric wire was subjected to friction by vibration during flight at the relevant part.
- The temperature rises due to aerodynamic effects on the metal structure.
- The electric wire bundle is pressed against a corner of the relevant part under the influence of acceleration.
- In No. 4, the metal structure was changed from stainless steel to aluminum, which has a thermal conductivity about 10 times higher.
- In No. 4, the shape of the cowling and the position of the wire drawing hole were changed.
- In No. 4, thinner wires were used for weight reduction.
- In No. 4, the adhesive that fixes the wire was changed to an epoxy adhesive with a low glass transition temperature.

(1 - 2) Wire exposure and covering damage due to cable duct damage
- The part is molded with a cork sheet, and there is a possibility that the electric wire will be exposed if it breaks.
- The part is directly subjected to aerodynamic influences (airflow, aerodynamic heating, etc.) during flight.
- In No. 4, design changes to the cable duct (addition of a cowling, removal of internal filler) were carried out for weight reduction and reduction of aerodynamic drag.
- Unusual fluctuation was observed in the pressure monitor (fluctuation in the measurement range is small and the noise is large, so it is difficult to judge the relevance to the event)

* When A and B are considered to be independent events

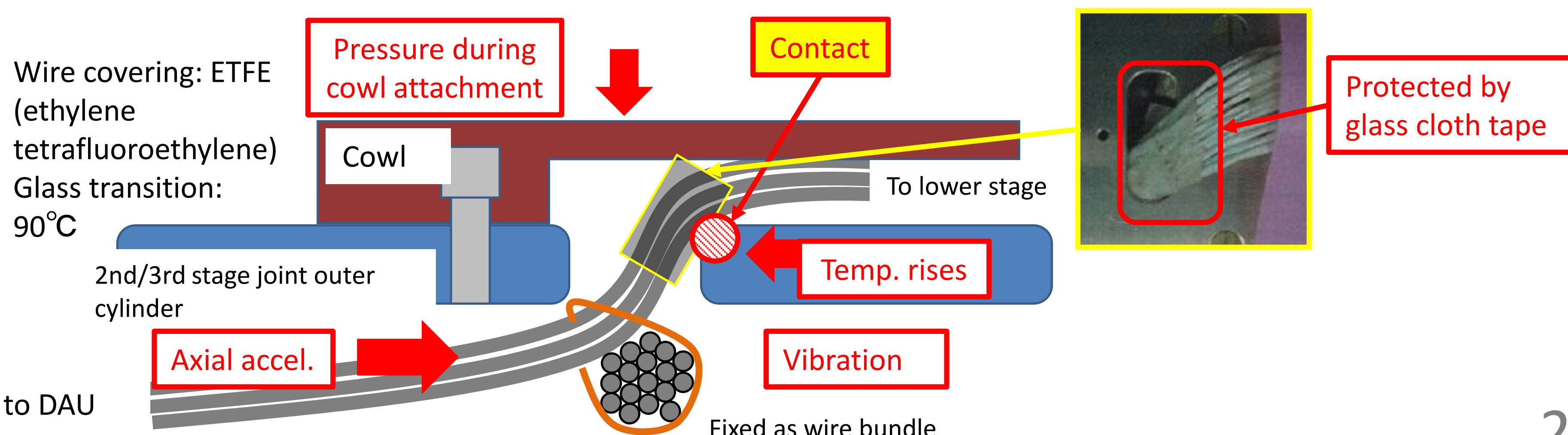
(2) Electric line connecter damage or disconnection
- No. 4 uses devices developed using many consumer parts (wire connectors, etc.)
- The circuit design must be such that power cannot be supplied to all on-board equipment (in particular, places close to the 28 V battery of the power distributor) when the part is damaged or disconnected.

4-4. Probable cause (1)

Short circuit due to wire cover damage at the wire drawing hole.

- The electric wire attached near the wire drawing hole (at the upper part of the cowl) was influenced by acceleration in the direction of the vehicle axis and by spacecraft vibration during flight, causing the surface cover on the electric wire to experience friction.
- The periphery of the wire drawing hole was heated by aerodynamic heating, and the temperature of the metal (aluminum) part increased.
- As a result, the wire covering was damaged due to a combination of effects:
  - Contact between the electric wire and metal structure by pressing force during cowl attachment
  - Static load due to axial acceleration
  - Friction due to spacecraft vibration
  - Difference in the degree of temperature rise in the wire contact area (*Changed to aluminum in No. 4)
  - Differences in covering protection measures from previous unit (* Changes in hole position and cowling shape from No. 4)
- As a result of the above effects, core wires of the 28 V power supply system were exposed and a wire with a damaged covering contacted the metal structure, resulting in a short circuit.

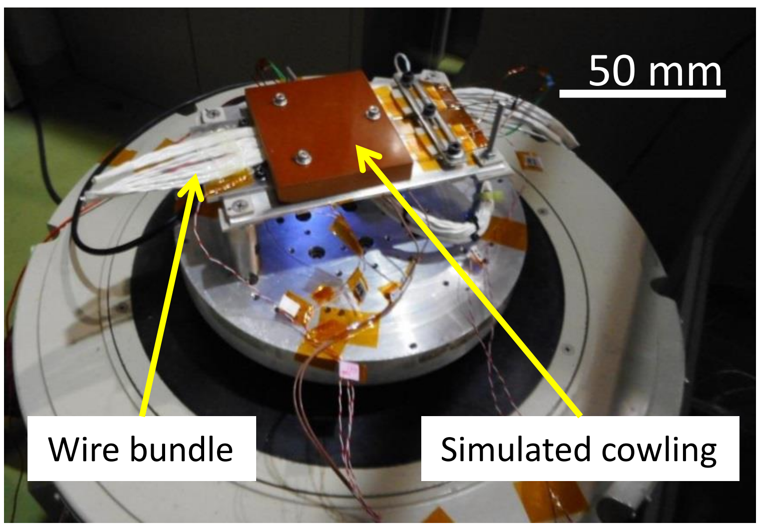
  *Because this part is constructed in the final assembly process, it cannot be verified in comprehensive testing.



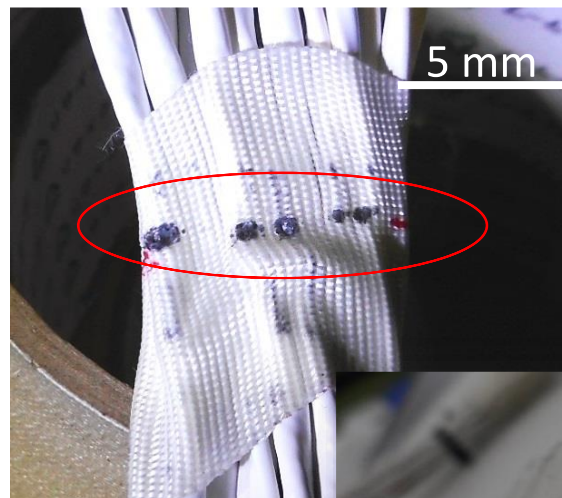
Reference: Electric wire friction tolerance test 1

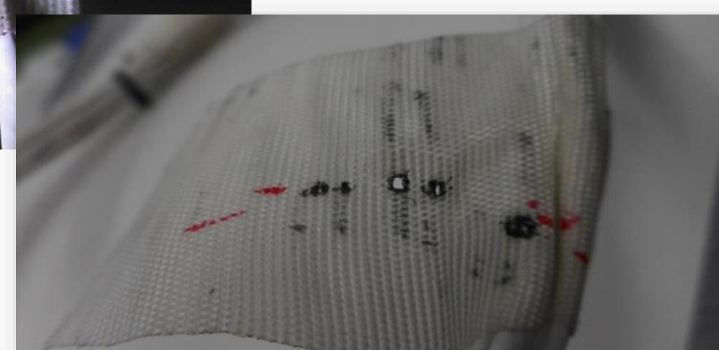
Purpose: Confirmation of friction durability of electric wire coatings in contact with the wire drawing hole.
Implementation: The test is conducted under environmental conditions with temperature and vibration simulating the flight environment.
Results: Damage to the glass cloth tape (mainly contacting parts) used to protect electric wire coverings

Excitation in the orthogonal direction



Expansion of the contact area

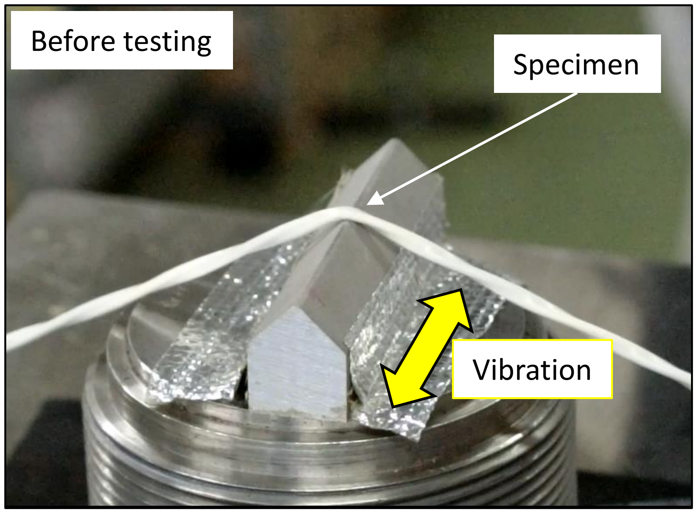




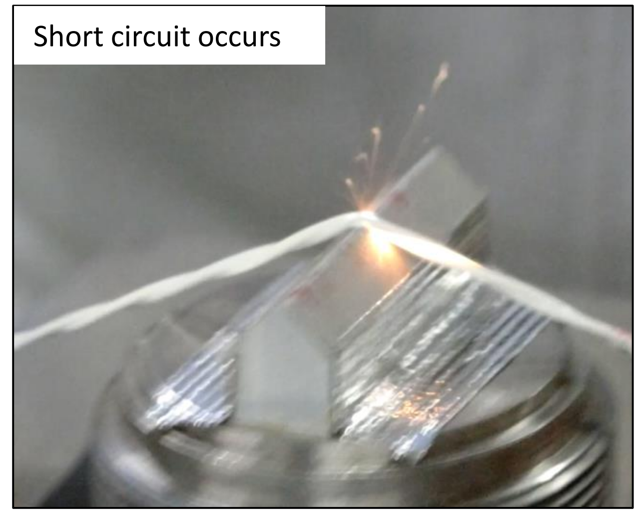
Protective glass cloth tape

Reference: Electric wire friction tolerance test 2

<u>Purpose</u>: Confirmation of friction resistance of aluminum material and electric wire covering simulating a contact area
<u>Implementation</u>: Crossing an electric wire coating on a curved contact surface and moving the contact part at constant speed and amplitude to apply friction to the coating, we confirmed the occurrence of a short circuit.
<u>Results</u>: A short circuit occurred at a total friction distance equivalent to the actual equipment that was assumed.



*28 V applied to wire



Estimated results for total friction distance*

	Vibration (Hz)	Amplitude (mm)	Duration (s)	Total friction dist. (mm)	Notes
Actual equipment (estimated)	80–160	0.5	20	1600 ~ 3200	Frequency refers to TLM instantaneous interruption period
Simulation	3	15	30	2700	A temperature of 130°C was applied.

*Total friction distance: Distance obtained as the product of amplitude, frequency, and time

4-5. Probable cause (2)

While occurrence probabilities are low, we cannot exclude the possibility of causal factors for

A) TLM data interruption, CMD answer errors (28 V power supply anomaly)

B) 2nd stage motor strain sensor measurement anomaly

1-2. Cable duct damage

- The cable duct on the SS520 No. 4 was designed to improve on Nos. 1 and 2 by reducing weight and aerodynamic drag and upgrading heat resistance to realize a microsatellite launch rocket.
- The following factors alone cannot be the cause of damage to the part, but when occurring in combination, there is an undeniable possibility that the design margin of the cable duct part was decreased.
  (1) Local pressure distribution on the fuselage surface
  (2) Uncertainty in structural strength estimations
  (3) Pressure fluctuation due to gusts during maximum dynamic pressure flight

Items that cannot be excluded as causal factors when A) and B) above occur independently:

2. Two electric wire connectors were damaged or disconnection at the same time

- A wire connector is used at the coupling portion of the device having the 28 V power supply and the power distribution function and at the coupling portion between the strain measurement system and the equipment (OBC). When two wire connectors are disconnected or damaged at the same time, equipment related to the power supply system loses power and sensor measurements fail to function normally.
- The development process considers that the occurrence probability of the assumed event is low because mounting involves tests simulating the flight environment (vibration, vacuum, heat) and visual checks of details. However, since it is not possible to verify all cases, the possibility cannot be ruled out.

5. Measures against presumed causes

(1) Countermeasures against damage to the wire covering (probable cause 1-1)

- With regard to wire drawing holes, we will review the position, shape, etc. based on past results and improve reliability.
- We will update the design to avoid direct contact between the wire and metal parts. Friction tests, etc. of the relevant part will be carried out beforehand to confirm protective effects.
- As further measures to improve reliability, we will improve electric wires, covering materials, and protective construction to increase friction resistance.

(2) Countermeasures against cable duct damage (probable cause 1-2)

We made design changes aimed at reducing weight and improving microsatellite launch capability, but we will review the new design specifications based on actual results to ensure reliability.

(3) Countermeasures against damage or disconnection of electric wire connectors (probable cause 2)

- We will review power supply redundancy measures for each device to prevent power loss.

All countermeasures will consider feasibility for a microsatellite launch system.

6. Summary

Our understanding of conditions related to anomalies that occurred during flight, fault tree analysis, analysis of event simulations and demonstration experiments, etc. revealed the points below. We were also able to clarify the design and manufacture of the spacecraft, flight analysis, and causes of the anomalous events.

1. About 20 s after launch, some on-board equipment such as the TLM transmitter and CMD lost power and telemeter communications ceased.

2. As a cause of the above, damage of the electric wire near the cable duct led to a short circuit, immediately damaging internal or external parts of power distribution devices installed upstream on the 28 V power line, and power supply was ultimately lost.

3. We believe there is only a low probability that a malfunction in other equipment caused an anomaly.

4. As a result of narrowing down the causes for this event, we will review details of the rocket design and assembly in detail to produce a highly reliable system.