



H3

FLIGHT No.10

PRESS KIT

ENGLISH Version



1 What is H3?



H3 is Japan's new large mainstay launch vehicle, jointly developed by the Japan Aerospace Exploration Agency (JAXA) and Mitsubishi Heavy Industries, Ltd. (MHI).

As the successor to H-IIA, H3 is designed to ensure that Japan retains its own means of access to space while offering a launch vehicle that meets the needs of satellite operators and other customers. To achieve this, H3 combines flexibility, high reliability and low cost. It builds on Japan's accumulated launch vehicle expertise while incorporating newly developed technologies, helping pass this knowledge on to the next generation.

The program has renewed the launch vehicle, its ground facilities and operating methods. The first stage uses the newly developed LE-9 main engine, based on Japanese technology. The vehicle, including its second stage, is larger and offers greater payload capacity to support a wider range of missions. Different combinations of LE-9 engines, SRB-3 solid rocket boosters and payload fairings provide multiple configurations, allowing efficient launches tailored to payload size, mass and destination orbit.

Cost reduction extends beyond vehicle and engine design to manufacturing and operations. Initiatives include more efficient production, new manufacturing methods such as 3D printing, and shared technology and components with the Epsilon S small solid-propellant launch vehicle. At Tanegashima Space Center, many H-IIA launch pad facilities have been reused, while selected facilities have been newly built. Upgraded tracking stations and automated ground equipment inspections improve operational efficiency and reliability.

H3 Test Flight No. 1 launched on 7 March 2023. Its second-stage engine did not ignite. Once it was determined that the Advanced Land Observing Satellite-3, DAICHI-3 (ALOS-3), could no longer reach its designated orbit, a flight termination command was transmitted to the vehicle. The launch failed.

JAXA established a special task force and worked with MHI and other partner companies to investigate the failure. The investigation identified multiple possible causal scenarios. Corrective actions addressed both the potential direct causes and underlying organizational factors, alongside broader reliability improvements.

With these measures implemented, Test Flight No. 2 launched on 17 February 2024, carrying the Vehicle Evaluation Payload-4 (VEP-4) and two small secondary satellites. It flew as planned, successfully placing the second stage into the designated orbit—a milestone not achieved on Test Flight No. 1—and separating both

secondary satellites. After one orbit of Earth, the second stage performed a controlled reentry to reduce orbital debris. VEP-4 also enabled verification of the payload separation interface.

Flight No. 3 launched DAICHI-4 (ALOS-4), the Advanced Land Observing Satellite-4, on 1 July 2024. Flight No. 4 launched the X-band defense communications satellite KIRAMEKI-3 on 4 November 2024. Flight No. 5 launched MICHIBIKI No. 6 on 2 February 2025. All three missions succeeded, using the H3-22 configuration with two LE-9 engines and two SRB-3 boosters.

On 26 October 2025, Flight No. 7 successfully launched the first HTV-X cargo spacecraft, HTV-X1. This was also the first flight of the H3-24 configuration, equipped with four SRB-3 boosters.

Flight No. 8 launched on 22 December 2025. The second-stage engine failed to build up normally during its second burn and shut down prematurely. MICHIBIKI No. 5 of the Quasi-Zenith Satellite System (QZSS) could not be placed into its planned orbit, and the launch failed. A special task force investigated the cause and corrective actions. In April 2026, the Ministry of Education, Culture, Sports, Science and Technology (MEXT) task force for H3 Flight No. 8 accepted the technical validity of the investigation and proposed measures for resuming launches.

Incorporating these measures, Flight No. 6 was launched on 12 June 2026. It was the test flight of the H3-30 configuration, which lifts off using three LE-9 engines without SRB-3 boosters. This flight successfully placed the second-stage vehicle into its designated orbit and deployed six small secondary satellites. The successful demonstration of H3-30 completed the full H3 configuration lineup.

Flight No. 9, an H3-22 incorporating the same measures, launched on 11 August 2026 and placed MICHIBIKI No. 7 into its designated orbit, marking the resumption of steady H3 launch operations.

The program will continue development of the LE-9 Type 2 engine and improve vehicle performance and launch site capabilities through block upgrades. It will also transition toward private-sector-led launch operations, maintaining Japan's independent access to space, improving efficiency and strengthening international competitiveness.

2 From H-IIA to H3

2.1 What was H-IIA?

H-IIA, Japan's principal large launch vehicle until its retirement in June 2025, first launched in 2001, more than 20 years ago. It was designated a mainstay launch vehicle: a nationally retained transportation system essential to independent access to space and to the fulfillment of government missions, particularly those related to national security.

H-IIA was a large launch vehicle with substantial payload capacity. It could deliver up to approximately 5.1 tonnes to sun-synchronous orbit (SSO), a north–south orbit several hundred kilometers above Earth commonly used by Earth observation satellites, and up to approximately 6.0 tonnes to geostationary transfer orbit (GTO), used for weather, communications and broadcasting satellites. Its versatility also allowed it to launch two satellites into different orbits on a single mission and to support specialized trajectories for lunar and planetary exploration.

From its maiden flight to its final launch on 29 June 2025, H-IIA flew 50 missions. All succeeded except Flight No. 6, giving a success rate of approximately 98.0%. Flights Nos. 7–50 achieved 44 consecutive successes, demonstrating a reliability rate among the highest in the world.

H-IIA also had a high on-time launch rate: excluding weather and other circumstances beyond its control, technical problems involving the vehicle or ground systems rarely caused postponements, allowing launches at their predetermined dates and times.

In 2007, JAXA transferred launch vehicle technology to MHI. MHI subsequently provided an integrated launch service, taking responsibility for orders, payload integration and launch.

H-IIA's performance and reliability supported numerous important Japanese government missions, including weather satellites, QZSS satellites and Information Gathering Satellites, as well as science missions such as the Hayabusa2 asteroid explorer and the Smart Lander for Investigating Moon (SLIM). MHI's private-sector operation also secured overseas orders from the Korea Aerospace Research Institute (KARI), Telesat of Canada, the Mohammed Bin Rashid Space Centre (MBRSC) in Dubai, United Arab Emirates, and Inmarsat of the United Kingdom.

H-IIB, an enhanced-capacity derivative of H-IIA, debuted in 2009. Nine H-IIB vehicles launched through 2020, delivering the KOUNOTORI H-II Transfer Vehicle (HTV) to the International Space Station (ISS) with supplies and scientific equipment.



2.2 Challenges

The environment surrounding Japanese launch vehicles has continued to change, presenting new challenges.

2.2.1 International price competition

H-IIA reduced launch costs relative to its predecessor, H-II, and entered the commercial launch market, accepting orders from domestic and overseas satellite operators. Since the 2010s, however, many new launch vehicles have emerged worldwide, promoting lower prices and ease of use. H-IIA faced increasing difficulty winning orders.

A prominent example is Falcon 9, developed by SpaceX. Since its first launch in 2010, Falcon 9 has undergone successive upgrades that increased payload capacity. Recovery and reuse of its first stage and fairings have substantially reduced launch costs. First-stage recovery and reuse are now routine, and Falcon 9 alone conducted 165 orbital launches in 2025. Its services are reportedly offered in some cases at close to half the price of other launch vehicles with comparable capacity.

SpaceX is also developing the next-generation Starship/Super Heavy system. By repeatedly reusing the vehicle in a manner similar to aircraft, the company aims to reduce the cost per launch to approximately US\$1 million–US\$10 million.

Other companies are also active. In January 2024, United Launch Alliance (ULA), a Boeing–Lockheed Martin joint venture, successfully launched its new Vulcan vehicle for the first time. Vulcan succeeds Atlas V and Delta IV, which supported US large satellite launches. Its configuration can be adjusted to payload mass and orbit, supporting large satellites and constellations.

Blue Origin successfully launched its large New Glenn vehicle for the first time in January 2025. Designed for recovery and reuse, its first stage lands on an offshore landing vessel. New Glenn's second launch, in November 2025, successfully recovered the first stage. In April 2026, that stage was flown again and recovered once more. Although New Glenn has not yet matched Falcon 9's launch record or reuse experience, it could become a significant competitor.

In Europe, Ariane 6, the successor to Ariane 5, successfully made its maiden launch in July 2024. It offers greater operational flexibility and lower costs than Ariane 5. Arianespace provides launch services,

positioning Ariane 6 as a means of restoring the competitiveness of European providers that once held a major share of the commercial market.

India is also a significant participant. The Indian Space Research Organisation (ISRO)'s LVM3 can launch approximately 10 tonnes to low Earth orbit and approximately 4 tonnes to GTO. In addition to national missions, it conducts commercial communications satellite launches, increasing its presence in the international launch market.

Startups worldwide are also actively developing small and medium launch vehicles. These may become competitors in the small satellite launch market in the future.

Against this background, Japan needs a launch vehicle with internationally competitive pricing and lower launch costs.

Major current and next-generation competitors to H3

Item	Falcon 9 Block 5	Vulcan Centaur	New Glenn	Ariane 6	LVM3 (GSLV Mk III)
Manufacturer/operator	SpaceX, US	ULA, US	Blue Origin, US	ArianeGroup/Arianespace, Europe	ISRO, India
LEO payload, tonnes*1	22.8	25.6	45.0	21.6	10.0
SSO payload, tonnes*2	Not published; SSO capability stated	22.3	Not published; SSO capability stated	15.5	Not published; SSO capability stated
GTO payload, tonnes*3	8.3	14.4	13.0	11.5	4.2
First launch	2018; original vehicle 2010	2024	2025	2024	2017; first orbital launch
Published launch price*4	US\$74 million, recovery configuration	From US\$110 million	Not published	Not published	Not published
Reuse	First stage and fairing	First-stage engine section; future concept	First stage	None	None

*1 LEO: low Earth orbit. Altitude and inclination assumptions differ between vehicles; values are representative or maximum figures published by their manufacturers or agencies.

*2 SSO: sun-synchronous orbit. Altitude and inclination assumptions differ; values are representative or maximum published figures.

*3 GTO: geostationary transfer orbit. The required velocity increment to reach geostationary orbit is not standardized here; values are representative or maximum published figures.

*4 Nominal figures from company or agency websites and similar sources.



SpaceX Falcon 9. © SpaceX.



Maiden launch of Ariane 6. © ESA – S. Corvaja.

2.2.2 Changes in the satellite market: larger and more diverse payloads

Although H-IIA was a large launch vehicle, demand has grown for satellites beyond its capacity. Demand is also increasing for launching dozens of small satellites together. Launch vehicles must therefore accommodate greater payload mass and size.

Recent geostationary satellites have grown in size and mass as communications capacity increases and electric propulsion enables longer operational lifetimes. Some weigh 7–8 tonnes. H-IIA's maximum GTO capacity was approximately 6 tonnes, meaning some satellites were beyond its launch capability.

For communications and broadcasting satellites, larger antennas generally improve performance. Larger fairings are therefore needed to accommodate them.

Market analyses and forecasts for satellites planned for launch through 2028 also predict substantial growth in satellite constellations: large groups of small satellites operating together. Launch vehicles need to carry dozens at once and provide dedicated payload structures, such as dispensers.

2.2.3 Increasing transportation costs

Large satellites and multiple satellite launches require larger launch vehicles. Larger vehicles generally cost more, yet competitive pressures demand lower prices. Providers must meet the seemingly conflicting requirements of launching heavier payloads at lower cost.

2.2.4 Fewer new development opportunities and declining technical capability

More than 20 years have passed since H-IIA's first launch, and development began in 1994. H-IIA was a derivative of H-II, while H-IIB was a derivative of H-IIA. Japan had therefore not developed an entirely new large launch vehicle for more than 30 years.

This creates a risk that the transfer of technology and know-how from experienced engineers to younger generations will be interrupted, jeopardizing future development and manufacturing capability.

2.2.5 Aging facilities

At Tanegashima Space Center, some H-IIA launch and range facilities were built more than 20 years ago; even relatively recent facilities are over 10 years old. The coastal location accelerates corrosion of piping and valves. Regular repairs maintain these facilities but consume a substantial share of the space development budget. Development of a new vehicle offered an opportunity for fundamental refurbishment.

2.2.6 Insufficient launch opportunities

H-IIA typically launched only one to four times a year, a limited rate compared with other countries. Low launch rates make it difficult to maintain and develop a stable industrial base. More launches are needed.

2.3 Maintaining independent access to space and international competitiveness

By the mid-2010s, these challenges raised concerns that Japan might be unable to sustain its launch vehicle capabilities a decade later. H3 was developed to address them and achieve two major objectives: maintaining independent access to space and securing international competitiveness.

2.3.1 Maintaining independent access to space

Satellites are now essential to communications, broadcasting, weather observation and positioning. Launch vehicles are indispensable for placing them in space.

Because launch capability is directly linked to national security, retaining the freedom to launch a country's own vehicles from its own territory is critically important. Space utilization is expected to expand across many fields, requiring efficient and reliable launch vehicles for security, business and international contributions.

As H-IIA's successor, H3 aims to retain, develop and strengthen Japan's independent access to space. Sharing SRB-3 technology with Epsilon S, developed in parallel, also helps maintain Japan's indigenous solid-propellant rocket technology.

2.3.2 Securing international competitiveness

Independent access requires a stable, continuing launch program that maintains and strengthens the technological and industrial base. This means expanding commercial launches for domestic and overseas private satellite operators, in addition to Japanese government missions.

H-IIA pursued commercial orders through improved reliability and reduced costs but faced strong competition. H3 aims to provide high reliability and competitive pricing based on customer needs. Combining commercial orders with government missions, the program plans a stable rate of six launches per year. This supports further reliability gains and cost reductions, reinforcing Japan's independence in space transportation.

MHI participates actively as a private-sector partner to help develop an internationally competitive system.

3 H3's objectives: flexibility, high reliability and low cost

Development began by analyzing market trends, competing launch vehicles and customer requirements. Customers identified launch price and reliability as their most important selection criteria, followed by flexibility and certainty in launch scheduling. H3 therefore targets three characteristics: flexibility, high reliability and low cost.

3.1 Flexibility

Payload size, mass and orbit vary widely. H3 uses a common vehicle design with different numbers of engines and boosters and different fairing sizes. This lineup supports smaller payloads more efficiently than H-IIA and larger payloads beyond H-IIA's capacity, covering a wide range of payloads and destination orbits.

H-IIA manufacturing was close to individually tailored, build-to-order production. This increased costs and limited launch frequency. H3 shifts toward line production, similar to cars and aircraft, with the aim of reducing the order-to-launch period from approximately two years to approximately one year.

H3 also aims to halve the minimum launch interval from two months to one month and shorten launch-site vehicle assembly and payload integration from one month to half a month.

Geostationary satellites are generally launched into GTO and then use their own propulsion to reach geostationary orbit. Customers prefer transfer orbits that minimize the satellite's required velocity increment, or ΔV . A GTO requiring $\Delta V = 1,500$ m/s has become the global de facto standard.

Tanegashima's relatively high latitude meant that H-IIA either inserted satellites into transfer orbits requiring larger ΔV or suffered a payload-capacity penalty when targeting $\Delta V = 1,500$ m/s. H3 aims to deliver payloads of at least 6.5 tonnes to a GTO requiring $\Delta V = 1,500$ m/s, matching the market standard while reducing the propulsion burden on the satellite.

3.2 High reliability

H3 aims to inherit H-IIA's high reliability: a 98% launch success rate, 44 consecutive successful missions from Flight No. 7 onward, and a high on-time launch rate, excluding weather and other uncontrollable factors.

Reliability is further enhanced by applying a high-reliability development approach to the new LE-9 first-stage engine and adopting fault-tolerant avionics architectures.

3.3 Low cost

H-IIA launch prices were relatively high compared with newer foreign launch vehicles. H3 therefore pursues cost reductions at every stage, from concept and development to operations.

Measures include commercial components from Japan's automotive and other industries, revised manufacturing processes and supply chains, additive manufacturing (AM, or 3D printing), and a shift toward line production. Automated ground-system inspections also aim to reduce the number of launch-site personnel at Tanegashima to one-quarter of the previous level.

4 Development organization

H3 development began in 2014. MHI is the prime contractor, responsible for coordinating vehicle development. JAXA is responsible for the integrated system, key technologies and ground facilities, and develops the engine systems together with MHI.

4.1 Key technology contractors

Companies are listed in the order used in the Japanese source.

Company	Responsibilities
IHI Corporation (IHI)	Liquid oxygen and liquid hydrogen turbopumps for LE-9 and LE-5B-3
IHI Aerospace Co., Ltd. (IA)	SRB-3 solid rocket boosters; reaction control system (RCS) gas jets for vehicle attitude control
Japan Aviation Electronics Industry, Ltd. (JAE)	Inertial sensor unit with ring laser gyros and accelerometers; provides navigation and attitude-control measurements including position, attitude, azimuth, velocity and acceleration
Mitsubishi Heavy Industries, Ltd. (MHI)	LE-9 and LE-5B-3 engine systems and combustion assemblies, excluding turbopumps
Mitsubishi Electric Software Corporation (MESW)	Mission analysis, flight simulation, navigation and guidance software, etc.

4.2 Development cost

Total development cost: ¥239.3 billion, as of March 2026 (FY2025).

4.3 Development schedule

2012

- System concept studies and technology demonstrations for LE-X, the precursor to LE-9.

2013

- Continued system concept studies and LE-X technology demonstrations.
- 28 May: At its sixth meeting, the Cabinet Office Space Policy Committee's Space Transportation Systems Subcommittee decided to begin development of a new mainstay launch vehicle in FY2014.

2014

- January: JAXA conducted the Mission Definition Review (MDR).
- 25 March: MHI selected as the development lead.
- April: H3 project launched.

2015

- April: System Definition Review (SDR) completed, enabling transition from conceptual to preliminary design. Preliminary design began for vehicle systems and subsystems—structures, electrical systems, engines and SRB-3—and ground facilities.

2016

- April: JAXA conducted the integrated-system Preliminary Design Review (PDR), concluding that transition to detailed design was feasible.
- 27 December–26 January 2017: Seven LE-9 turbopump component tests, Series 1, at Kakuda Space Center.

2017

- 30 March–31 October: Twenty LE-5B-3 qualification tests, Series 1, at Kakuda.
- 24 April–12 July: Eleven LE-9 flight-type engine #1-1 firing tests at Tanegashima.
- 5–30 June: Six LE-9 turbopump component tests, Series 2, at Kakuda.
- December: JAXA conducted the integrated-system Critical Design Review (CDR), concluding that transition to manufacturing and testing was feasible.
- 12 December–25 June 2018: Eight LE-9 flight-type engine #2 firing tests at Tanegashima.

2018

- 20 February–9 March: Four LE-9 turbopump component tests, Series 3, at Kakuda.
- 23 August: LE-9 flight-type engine #3 firing test at Tanegashima.
- 24 September: LE-9 flight-type engine #4 firing test at Tanegashima.
- 25 September–5 October: Three LE-9 turbopump component tests, Series 4, at Kakuda.
- 26 August: SRB-3 flight-type motor static firing at Tanegashima.
- 10 November–18 February 2019: Fifteen LE-5B-3 qualification tests, Series 2, at Kakuda.
- 25 December–29 May 2019: Eight LE-9 flight-type engine #1-2 firing tests at Tanegashima.

2019

- 18 January–12 April: Four first-stage battleship tank firing tests (BFT), with two clustered LE-9 engines, at MHI's Tashiro Test Facility.
- 20 May: SRB-3 separation test, Series 1, at IHI Aerospace's Tomioka facility.
- 28 August: SRB-3 qualification motor static firing, Series 1, at Tanegashima.
- 12–21 October: Two LE-9 flight-type engine #1-3 firing tests at Tanegashima.
- 17 October–13 February 2021: Four three-engine LE-9 cluster BFTs at Tashiro.
- 15–20 November: Two LE-9 turbopump component tests, Series 5, at Kakuda.
- 17 December: Fairing separation and jettison test at Kawasaki Heavy Industries' Harima Works.

2020

- 13 February–26 May: Eight LE-9 qualification engine #1 firing tests at Tanegashima.
- 29 February: SRB-3 qualification motor static firing, Series 2, at Tanegashima.
- 2 July–29 August: Three second-stage flight-type tank stage firing tests at Tashiro.
- 24 July: SRB-3 separation test, Series 2, at IHI Aerospace's Tomioka facility.
- 9–15 August: Two LE-9 turbopump component tests, Series 6, at Kakuda.
- 11 September: Announced that Test Flight No. 1 would not launch in FY2020.
- 28 September–3 October: Two LE-9 turbopump component tests, Series 6-2, at Kakuda.
- 19 November–8 April 2021: Nine LE-9 technical-data firing tests at Tanegashima.

2021

- 12 March–1 April: Two LE-9 turbopump component tests, Series 7, at Kakuda.
- 17–18 March: Cryogenic test at Tanegashima.
- 26 June–13 July: Two LE-9 turbopump component tests, Series 7-2, at Kakuda.

- 30 June–19 October: Five LE-9 qualification engine #2 firing tests at Tanegashima.
- 9–18 December: Three LE-9 turbopump component tests, Series 8, at Kakuda.

2022

- 21 January: Announced that Test Flight No. 1 would not launch in FY2021.
- 23 March–29 June: Six LE-9 blade-vibration measurement and technical-data tests at Tanegashima.
- 3–9 July: Two LE-9 turbopump component tests, Series 9, at Kakuda.
- 20 July–14 August: Five LE-9 qualification engine #3 firing tests at Tanegashima.
- 6 and 12 September: Acceptance firing tests of the first LE-9 engine for Test Flight No. 1 at Tanegashima.
- 29 September: LE-9 turbopump component test at Kakuda to collect data for selecting specifications for Test Flight No. 2 and subsequent vehicles.
- 3 October: Acceptance firing test of the second LE-9 engine for Test Flight No. 1 at Tanegashima.
- 23 October–10 November: Four LE-9 qualification engine firing tests at Tanegashima.
- 6 November: First-stage captive firing test (CFT) at Tanegashima.
- 10 December: LE-9 turbopump component test at Kakuda to collect data for selecting specifications for Test Flight No. 2 and subsequent vehicles.
- 23 December: Announcement of the Test Flight No. 1 launch carrying DAICHI-3 (ALOS-3).

2023

- 8 January: Acceptance firing test of the first LE-9 engine for Test Flight No. 2 at Tanegashima.
- 17 February: Test Flight No. 1 launch aborted. During the automatic countdown, the first-stage system detected an anomaly and did not send the SRB-3 ignition signal.
- 21 February–8 April: Five LE-9 blade-vibration measurement and technical-data tests at Tanegashima.
- 7 March: Test Flight No. 1 launched. A flight termination command was transmitted after it was determined that the mission could not be achieved; the launch failed.
- 7 March: Special task force established for the Test Flight No. 1 failure.
- 8 and 16 March; 27 April: Failure reports to the expert meeting on space-development investigation and safety.
- 28 April: Investigation status reported to the 74th meeting of the Committee on Space Development and Utilization.
- 24 May: Proposed direction for Test Flight No. 2 reported to the committee's 75th meeting.
- 25 May: Investigation status reported to the 47th meeting of its Investigation and Safety Subcommittee.
- 1–11 June: Two LE-9 blade-vibration measurement tests at Tanegashima.
- 22 June: Investigation status reported to the subcommittee's 48th meeting.
- 27 June: Small-satellite rideshare plan for Test Flight No. 2 reported to the committee's 76th meeting.
- 13–25 July: Three LE-9 turbopump component tests at Kakuda.
- 31 July: Investigation status reported to the subcommittee's 49th meeting.
- 23 August: Investigation status reported to the subcommittee's 50th meeting.
- 6 August–12 December: Eight LE-9 qualification engine firing tests at Tanegashima.
- 25 September: Investigation status reported to the subcommittee's 51st meeting.
- 15 and 26 October: Acceptance firing tests of the second LE-9 engine for Test Flight No. 2 at Tanegashima.
- 26 October: The special task force compiled its failure investigation report and presented it to MEXT's Investigation and Safety Subcommittee, 52nd meeting. The Committee on Space Development and Utilization adopted it that day.
- 28 December: Test Flight No. 2 launch announced.

2024

- 17 and 24 January: LE-9 turbopump component tests at Kakuda.

- 17 February: Test Flight No. 2 launched.
- 26–27 May: H3 cryogenic test.
- 21 June; 6 July; 7, 14 and 21 November: LE-9 firing tests for blade-vibration measurements and technical data at Tanegashima.
- 1 July: Flight No. 3 launched DAICHI-4 (ALOS-4).
- 4 November: Flight No. 4 launched KIRAMEKI-3.

2025

- 2 February: Flight No. 5 launched MICHIBIKI No. 6.
- 15, 20 and 27 May; 3 June: LE-9 firing tests for blade-vibration measurements and technical data at Tanegashima.
- 23–24 July: Flight No. 6 (H3-30 test vehicle) first-stage CFT at Tanegashima.
- 26 October: Flight No. 7 launched HTV-X1.
- 22 December: Flight No. 8 (H3-22S) launched. The second-stage engine failed to build up normally during its second burn and stopped prematurely; MICHIBIKI No. 5 did not reach its planned orbit, and the launch failed.
- 22 December: Special task force established for the Flight No. 8 failure.

2026

- 11 March: Second first-stage CFT for Flight No. 6 (H3-30 test vehicle), completed successfully.
- 22 April: Investigation and Safety Subcommittee presented its interim report on the Flight No. 8 failure to the 105th meeting of the Committee on Space Development and Utilization.
- 12 June: Flight No. 6 (H3-30 test vehicle) launched.
- 23 July: Investigation and Safety Subcommittee presented its report on the Flight No. 8 failure to the committee's 107th meeting.
- 27 March; 7 and 17 April; 20 June; 1, 10, 19 and 27 July: LE-9 Type 2 qualification engine firing tests at Tanegashima.
- 11 August: Flight No. 9 launched MICHIBIKI No. 7.

5 H3 overviews

5.1 Vehicle name

- Japanese name: H3 rocket, pronounced “H-three rocket,” with standard-width alphanumeric characters and no space between H3 and the Japanese word for rocket. Abbreviation: H3.
- English name: H3 Launch Vehicle. Abbreviation: H3.

The name retains “H” to continue the heritage of large liquid-oxygen/liquid-hydrogen launch vehicles and preserve the recognition and trust important to international competitiveness. Unlike H-IIA, which was an H-II derivative, H3 fundamentally revisits the launch vehicle concept and is therefore not named H-IIC. The Arabic numeral “3” was selected instead of the Roman numeral “III” to avoid confusion with “II” and ensure clear public and media recognition.

5.2 Vehicle overview

H3 is approximately 63 m long and 5.2 m in diameter, roughly equivalent in length to two leading cars of a Tokaido Shinkansen train. Its common vehicle design supports different numbers of LE-9 main engines and SRB-3 boosters, and different fairings, accommodating a broad range of payload sizes and masses, with configurations tailored to customer mission requirements.

Configurations are designated H3-abc:

Character	Meaning	Options
a	Number of first-stage LE-9 engines	2 or 3
b	Number of SRB-3 boosters	0, 2 or 4
c	Fairing size	L: Long; S: Short; W: Wide

The smallest configuration is H3-30S, pronounced “H-three, three-zero-S.” It can deliver at least 4 tonnes to a 500 km × 500 km SSO. This orbit is important for government missions such as Information Gathering Satellites and Earth observation satellites and is seeing growing commercial demand. Whereas H-IIA 202 provided excess capacity for such missions, H3-30 offers an appropriate capacity at lower cost.

H3-22L and H3-24L are intended for commercial missions. H3-22 primarily supports medium geostationary satellites. H3-24 supports large geostationary satellites in the 6-tonne class and multiple small satellites, extending coverage beyond H-IIA's capacity.

For GTO missions, H3 can provide an orbit requiring $\Delta V = 1,500$ m/s to reach geostationary orbit, comparable to vehicles such as Ariane 5. H3-24 also launches HTV-X cargo spacecraft to the ISS, using a dedicated wide fairing.

Payload attach fittings (PAFs) support standard clamp bands from Japanese and overseas manufacturers, with interface diameters of 937, 1,194 and 1,666 mm. This enables single launches of a wide range of large satellites. Dedicated payload structures also allow multiple small satellite launches.



H3 lineup: H3-30S, H3-22S, H3-22L, H3-24L and H3-24W.

5.3 Specifications

Item	Specification
Overall length	63 m with long fairing; 57 m with short fairing; 64 m with wide fairing
Liftoff mass	575 t: H3-24L/24W; 420 t: H3-22L; 419 t: H3-22S; 270 t: H3-30S
Payload fairings	Short (S), Long (L), Wide (W)
PAF interface diameters	937/1,194/1,666 mm

Element	Engines/motors	Vacuum thrust	Specific impulse
First stage, 5.2 m diameter	Two or three LE-9 engines	1,471 kN per engine at 100%; throttle capability	422 s
Second stage, 5.2 m diameter	One LE-5B-3	137 kN	448 s
Solid boosters, 2.5 m diameter	Zero, two or four SRB-3 motors	2,300 kN	283.6 s

5.4 Launch capability

By selecting the number of boosters and fairing, H3 can efficiently launch small or large payloads into a variety of orbits. H-IIA's capacity to a GTO requiring $\Delta V = 1,500$ m/s was limited, in part by Tanegashima's latitude. H3 increases that capacity to at least 6.5 tonnes.

Target orbit	Payload capability
SSO	At least 4 tonnes
GTO	At least 6.5 tonnes, with geostationary insertion $\Delta V = 1,500$ m/s

Low Earth orbit (LEO) is a general term for orbits several hundred kilometers above Earth. Launch performance to LEO often refers to a due-east launch from the launch site, corresponding to the vehicle's maximum payload capability.

Sun-synchronous orbit (SSO) is a near-polar low Earth orbit whose plane maintains an approximately constant relationship to the Sun. A satellite passes a given location at approximately the same local solar time, producing consistent illumination and facilitating analysis of Earth observation images.

Geostationary transfer orbit (GTO) is an intermediate orbit used when launching geostationary satellites. The satellite uses its own propulsion to transfer to geostationary orbit. Customers prefer transfer orbits that minimize this required ΔV .

5.5 Launch price

For H3-30S, the target vehicle price is half that of H-IIA, under specified conditions in the steady operational phase once manufacturing has stabilized, based on economic conditions in 2014.

5.6 Tanegashima Space Center

H3 launches from JAXA's Tanegashima Space Center, on Tanegashima Island south of Kagoshima. Facing a coastline of coral reefs and emerald-green water, it is often described as the world's most beautiful launch site.

The center opened in October 1969, initially launching small experimental rockets. It later supported N-I, N-II, H-I, H-II, J-I and H-IIA/B, mainly medium and large vehicles for practical satellite missions.

Covering approximately 9.7 million m²—about 200 Tokyo Domes—the center includes the large launch complex, spacecraft assembly buildings and spacecraft/fairing assembly buildings. It supports the full launch process: final spacecraft checks, payload integration, vehicle assembly, servicing, inspection, launch and tracking. It is the central facility for Japan's satellite launch activities.



Tanegashima Space Center's large launch complex, the launch site for H3.

6 LE-9 engine

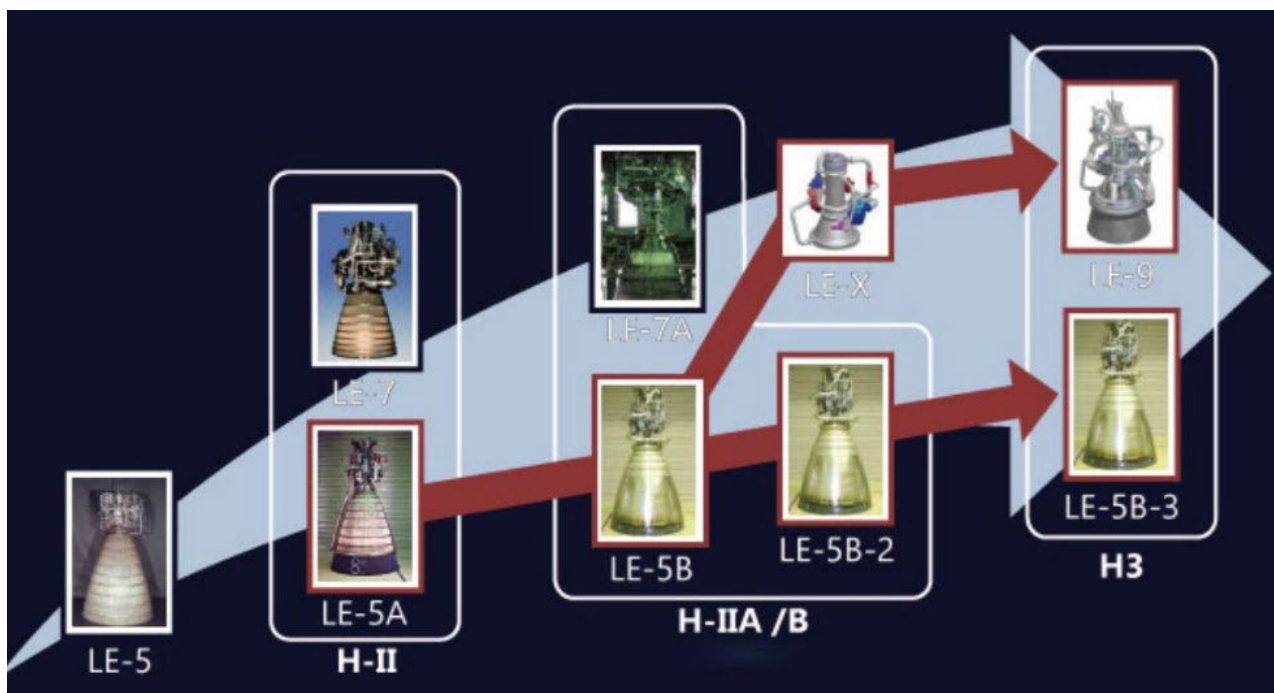
6.1 What is LE-9?

LE-9 is the new main engine developed for H3's first stage. JAXA leads development, with MHI and IHI participating. It combines Japan's accumulated rocket-engine expertise with new technologies to realize H3's objectives of flexibility, high reliability and low cost.

LE-9 uses liquid hydrogen as fuel and liquid oxygen as oxidizer. Each engine produces approximately 1.4 times the thrust of H-IIA's LE-7A first-stage engine. Two or three LE-9 engines are clustered, depending on payload mass and destination orbit, providing greater flexibility and capacity than H-IIA or H-IIB.

To deliver high thrust at low cost and with high reliability, LE-9 uses Japan's expander-bleed cycle, drawing on experience gained through previous engine development and operations. New technologies include additive manufacturing and the world's first use of large electrically actuated valves for engine thrust control (throttling).

6.2 Engine heritage



Evolution of Japanese liquid-propellant rocket engines leading to LE-9. Engine designations in the original artwork identify the LE-5 family, LE-7 family and LE-9.



LE-9 engine.

6.3 Expander-bleed cycle

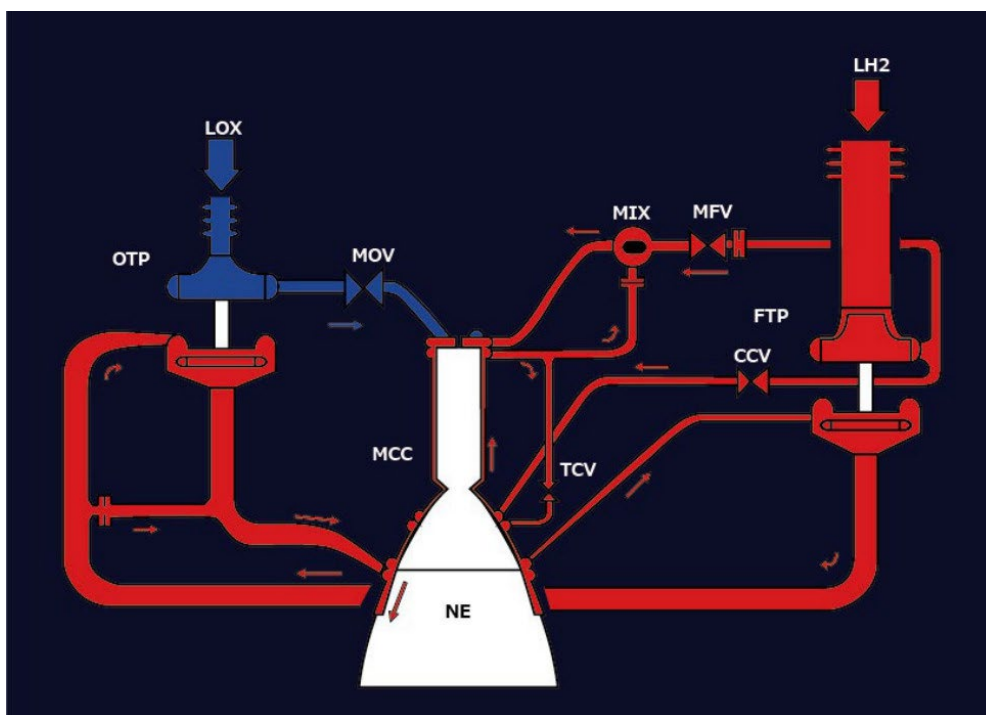
LE-9 is the first application of Japan's expander-bleed technology to a high-thrust first-stage engine.

LE-7A uses a staged-combustion cycle. Propellant first burns in a small preburner. The resulting gas drives the turbopumps, which supply propellant to the main combustion chamber. The turbine exhaust also enters the main chamber and burns there. The name reflects this two-stage combustion process.

Because all propellant contributes to the exhaust, staged combustion is highly efficient. However, it is structurally complex—effectively incorporating a small engine within the main engine—and subjects components to demanding pressures and temperatures. Malfunctions can create a risk of explosion, and controlling the startup sequence is difficult.

In contrast, LE-9's expander-bleed cycle uses liquid hydrogen to cool the combustion chamber and nozzle. The hydrogen absorbs heat, vaporizes and drives the turbopumps. The system is comparatively simple.

Although less efficient than staged combustion, it requires no preburner, reduces the number of components and is less susceptible to abnormal combustion conditions, combining high reliability with low cost. This cycle was first put into practical use in H-II's LE-5A second-stage engine and subsequently in the LE-5B used by H-IIA/B. LE-9 builds on this long-established Japanese technology.



LE-9 engine schematic. English key: LH2—liquid hydrogen; LOX—liquid oxygen; FTP—fuel (liquid hydrogen) turbopump; OTP—oxidizer (liquid oxygen) turbopump; MOV—main oxidizer valve; MFV—main fuel valve. Hydrogen absorbs heat in the combustion chamber cooling passages; heated gas drives the turbopumps. The main propellant flow feeds the combustion chamber.

LE-9 and LE-7A comparison

Item	LE-9	LE-7A
Vacuum thrust	1,471 kN (150 tonne-force); throttle capability	1,100 kN (112 tonne-force)
Mixture ratio	5.9	5.9
Specific impulse	422 s	440 s
Engine cycle	Expander-bleed	Staged combustion
Propellants	LH2/LOX	LH2/LOX
Chamber pressure	10.0 MPa	12.3 MPa
LH2 turbopump speed	Approximately 41,000 rpm	Approximately 41,900 rpm
LOX turbopump speed	Approximately 17,000 rpm	Approximately 18,300 rpm
Overall length	3.75 m	3.7 m
Mass	Approximately 2.4 t	Approximately 1.8 t
Valve actuation	Electric; continuously adjustable operating point	Pneumatic; operating point adjusted using orifices

6.4 New technologies

Three major technologies have been introduced.

High-thrust expander-bleed cycle

The combustion chamber is elongated to increase heat absorption by the hydrogen used for cooling, supplying the energy needed to drive the turbines. Large-part forming techniques, including die forging and flow forming, make this elongated chamber possible.

Throttling using large electrically actuated valves

Electric valves replace conventional helium-actuated valves, simplifying the system and reducing costs. Fine valve-position control also enables precise engine thrust adjustment.

Near the end of first-stage flight, LE-9 is throttled to approximately 66% thrust. As propellant is consumed, vehicle mass decreases and acceleration rises; throttling limits the loads on the payload. The throttle duration varies by configuration: approximately 20 seconds for H3-2X and approximately 60 seconds for H3-30.

Electric control also improves development and acceptance testing. Multiple operating points can be checked during a single firing, greatly increasing the number of demonstrated conditions.

Additive manufacturing

AM, or 3D printing, has been applied to valves, combustion-chamber parts and piping from Test Flight No. 1 onward. Application will expand to injectors and other structures. Conventional complex parts often required numerous welded or bolted components. Producing them as integrated AM parts greatly reduces component count and cost.

6.5 Two-stage development

LE-9 is being developed in two stages. The original plan was to use an engine incorporating new technologies, including AM, on Test Flight No. 1. In 2019, however, firing tests revealed abnormal combustion characteristics in the AM injector and fatigue fracture surfaces in the fuel turbopump's turbine rotor blades.

In October 2019, the qualification plan was revised. Stage 1 would produce a Type 1 engine using a proven machined injector and operating outside resonance regions. Stage 2 would produce a Type 2 engine with an AM injector and design changes to eliminate the resonance regions themselves.

Two further technical issues emerged during Type 1 testing in 2020, forcing another revision. To minimize risk, multiple design options were developed in parallel so that alternative countermeasures could be implemented quickly if an earlier option proved ineffective.

Type 1 was completed, but evaluation of the optimum Type 2 specifications for Test Flight No. 2 and later missions identified further issues requiring design changes and testing. Type 1A was therefore prepared for launches in the interim.

6.6 Type 1A engine

Type 1A supports the initial launches from Test Flight No. 2 onward and bridges the transition to Type 2, which incorporates permanent corrective designs.

Type 1A retains the Type 1 fuel turbopump (FTP) and injector. Its oxidizer turbopump (OTP) uses the permanent design that suppresses nonuniform flow at the turbine inlet. Other principal components use specifications whose functions and performance were confirmed in development tests.

Building on the technology established by Type 1, Type 1A has accumulated flight experience while supporting H3's initial operations.

6.7 Type 2 engine

Type 2 applies permanent corrective designs based on experience with Types 1 and 1A, with a view to stable operations and future series production.

The FTP turbine is redesigned to improve engine performance while suppressing blade vibration. AM is applied to major components, including the injector. The aim is to consolidate an injector previously made from more than 1,000 machined parts into a single AM part. Complex internal passages can be formed integrally,

reducing component count and manufacturing steps while improving performance, reliability and manufacturability.

LE-9 configuration comparison

Component	Type 1	Type 1A	Type 2
FTP	Option 0: increased stiffness and damping	Same as Type 1	Permanent corrective design*1
OTP	Option 1: suppression of nonuniform turbine-inlet flow	Permanent corrective design; same as Type 1	Permanent corrective design; same as Type 1
Injector	Machined	Same as Type 1	Permanent corrective design*2
Other components	Type 1 specifications	Permanent corrective designs, mainly demonstrated in development tests	Permanent corrective designs

*1 Multiple designs are under consideration to increase FTP turbine efficiency. Basing them on previously tested Option 0 and Option 1 designs reduces development risk.

*2 The permanent injector design plans to reduce product cost through additive manufacturing.

7 LE-5B-3 engine

7.1 What is LE-5B-3?

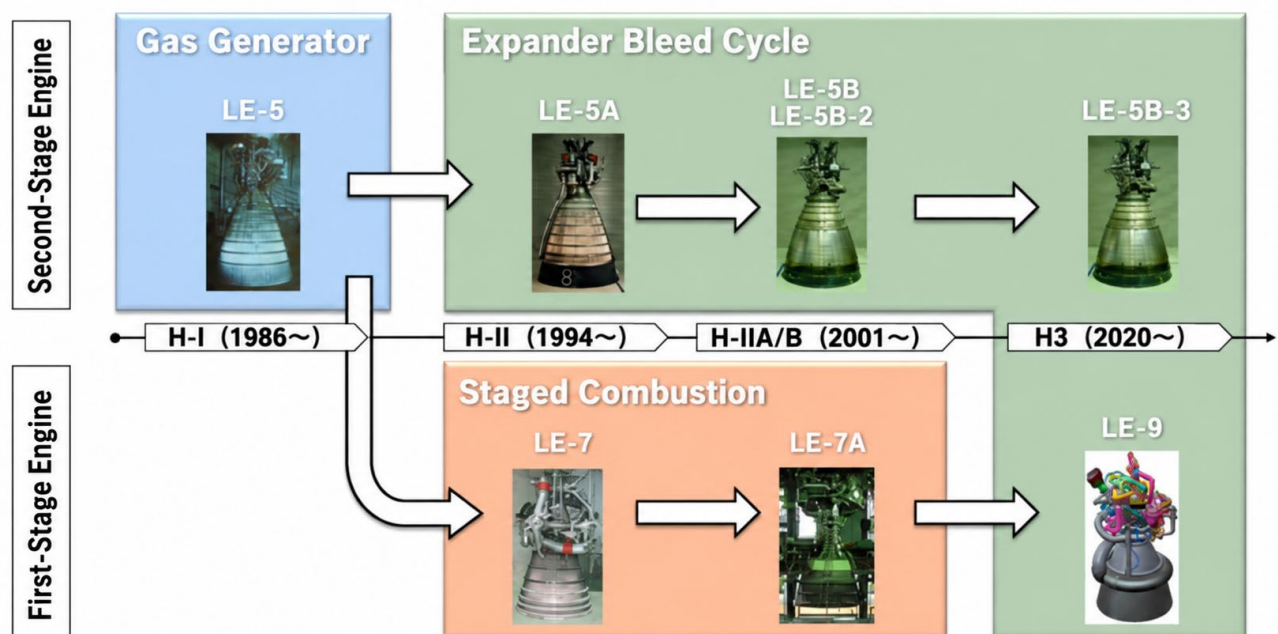
LE-5B-3 is a newly developed H3 second-stage engine that builds on Japan's upper-stage engine heritage. Its predecessor, LE-5B-2, used on H-IIA/H-IIB, can restart multiple times in orbit, as could the LE-5A developed for H-II.

To meet H3 requirements, LE-5B-3 improves propellant efficiency and service life relative to LE-5B-2. Changes include an improved mixer, which combines hot hydrogen gas after combustion-chamber cooling with cold liquid hydrogen supplied directly from the pump, and an improved FTP turbine.

Better mixing reduces combustion-temperature nonuniformity and improves heat transfer in the combustion chamber. Less hydrogen is needed to drive the turbopumps, improving specific impulse—the measure of propellant efficiency—from 446.6 s for LE-5B-2 to 448.0 s.

H3 extends second-stage engine burn time from LE-5B-2's 534 s to 740 s. To prevent high-cycle fatigue damage, the FTP turbine design reduces fluctuations in pressure and temperature, increasing life. Other changes improve reliability and lower cost.

The LE-5B family uses the expander-bleed cycle, in which hydrogen used for combustion-chamber cooling subsequently drives the turbopump turbines. LE-9 uses the same cycle, scaled up to produce much higher thrust.



Engine heritage of LE-9 and LE-5B-3. The upper branch shows first-stage engines; the lower branch shows the LE-5 upper-stage family.

7.2 Specifications

Item	Specification
Vacuum thrust	137 kN
Mixture ratio	5
Vacuum specific impulse	448 s
Cycle	Expander-bleed

Item	Specification
Propellants	LH2/LOX
Expansion ratio	110
LH2 turbopump drive-gas temperature	Approximately 400 K
LH2 turbopump speed	Approximately 52,000 rpm
LOX turbopump speed	Approximately 18,000 rpm
Overall length	Approximately 2.8 m
Mass	Approximately 300 kg
Maximum diameter	1,690 mm
Burn duration at 100% thrust	740 s
Burn duration in idle mode	250 s
Number of ignitions	3

8 Solid rocket booster: SRB-3

8.1 What is SRB-3?



Static firing of the H3 SRB-3 qualification motor, identified in this photograph's source caption as 21 August 2019.

SRB-3 is H3's solid rocket booster, developed and manufactured by IHI Aerospace. The smallest H3-30 configuration uses no boosters. H3-22 uses two for greater capacity; H3-24 uses four for still greater capacity. This flexibility supports diverse satellite missions.

H-IIA used SRB-A. "SRB" stands for Solid Rocket Booster; "A" indicated an improved version of the SRB used on H-II. "SRB-3" reflects both its use on H3 and its position as the third generation of this booster family.

SRB-3 is an SRB-A derivative, retaining the 2.5 m diameter and approximately the same motor-case dimensions. Thrust and burn time differ slightly because the burn profile, including propellant grain geometry, is tailored to each launch vehicle. Overall booster capability is broadly similar.

Unlike SRB-A's movable nozzle, SRB-3 uses a fixed nozzle. Attitude control through engine gimbaling is provided by the two or three LE-9 main engines. SRB-3 is also intended for use as Epsilon S's first stage, improving reliability and reducing costs through shared technology.

8.2 Features and improvements

Domestic motor-case production

A major change is domestic production of the motor case. SRB-A used a filament-wound carbon-fiber-reinforced plastic (CFRP) case to simplify structures and improve performance. Its short development schedule led to a design based on an existing US motor, requiring license payments and overseas manufacturing equipment.

Subsequent development of upper-stage motor cases for M-V and Epsilon established Japanese capabilities to design, develop and manufacture large CFRP cases with stable quality. SRB-3 could therefore be domestically produced, eliminating license fees, lowering costs and increasing freedom in design and material selection.

Improved attachment and separation

The connection between booster and first stage has been fundamentally redesigned. SRB-A on H-IIA/H-IIB used aluminum-alloy adapters at the top and bottom of the case, connected by six large rod-like components: thrust struts and yaw braces. Thrust struts transmitted booster thrust to the core stage; yaw braces transmitted lateral loads.

At separation, small separation rocket motors ignited, yaw braces were cut by pyrotechnic devices almost simultaneously, and thrust struts were cut immediately afterward. The booster then separated from the first stage.

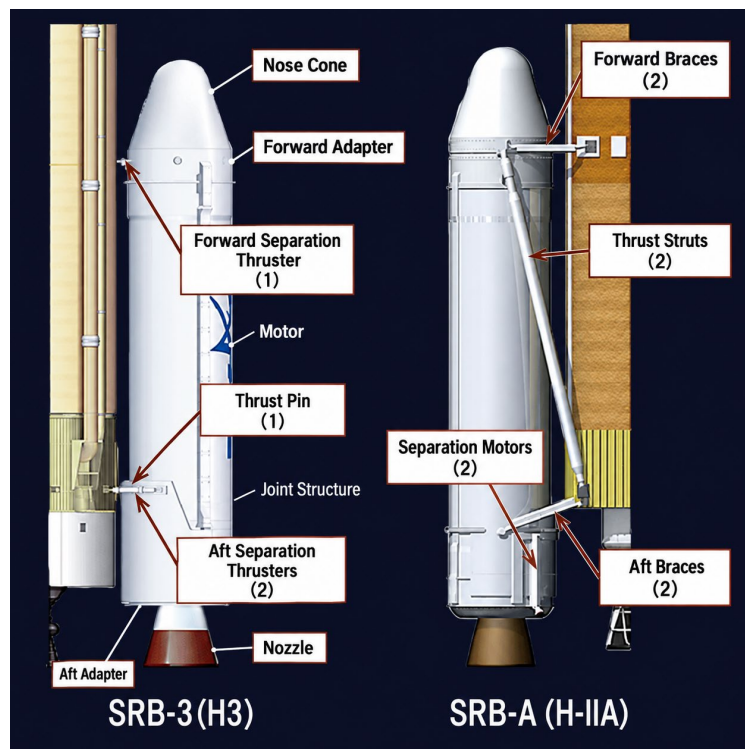
A CFRP motor case is vulnerable to concentrated loads and cannot simply be bolted directly to the core. Six rods distributed these loads. However, each was approximately as thick as a utility pole, adding mass; a failed cut could prevent separation and cause launch failure.

H3 adopts a simpler approach. Booster thrust is transmitted through a single metal cylindrical thrust pin. The pin fits into a receptacle in the first stage rather than being permanently fastened to it. A strong metal attachment structure at the bottom of the booster transfers loads through this pin without loading the CFRP case locally.

Gas actuators, or separation thrusters, at the top and bottom provide attachment and separation. They act as rigid links while attached. At separation, internally generated gas drives pistons outward, pushing the booster away from the first stage.

This eliminates thrust struts and separation rocket motors, reduces attachment points, simplifies the mechanism, lowers mass and cost, and improves performance.

Similar systems have been used on Atlas V boosters and on the smaller solid strap-on boosters (SSBs) formerly flown on H-IIA. Applying the mechanism to a motor as large as SRB-3 was novel internationally. Domestic motor-case design freedom was central to achieving this optimized arrangement.



SRB-3 versus SRB-A attachment and separation. English key: nose cone; motor case; forward separation thruster—one; aft separation thrusters—two; thrust pin—one; forward braces—two; aft braces—two; thrust struts—two; separation motors—two; forward adapter; aft adapter; attachment structure; nozzle. SRB-3 uses a thrust pin and gas actuators; SRB-A uses thrust struts, yaw braces and separation motors.

9 Payload fairing



H3 payload fairing separation and jettison test, 17 December 2019.

9.1 What is a fairing?

The fairing protects the payload from aerodynamic pressure and heating while the launch vehicle flies through the atmosphere.

H3 provides three types: Long (L), Short (S) and Wide (W). S primarily supports medium satellites. L is approximately twice as long as S, at 16.4 m, supporting large or multiple satellites. W was developed for HTV-X, has a diameter of 5.4 m and includes a large late-access door.

Kawasaki Heavy Industries develops and manufactures L and S fairings. Beyond Gravity of Switzerland is responsible for W.

9.2 Improvements to short and long fairings

9.2.1 Shape

H-IIA/B used a straight conical nose. H3 uses a smoothly curved ogive, or double-contour, shape for improved aerodynamic performance.

9.2.2 Automated fiber placement

Automated Fiber Placement (AFP) enables lower-cost CFRP layup, reducing structural-panel manufacturing cost while supporting higher production rates.

9.2.3 Out-of-autoclave bonding

H-IIA/B fairing panels were joined using bolts and nuts. H3 makes extensive use of out-of-autoclave adhesive bonding to reduce both mass and cost.

9.2.4 Designed to sink after splashdown

Conventional fairings could remain afloat after separation and splashdown because of their hollow interiors. To prevent hazards to ships and shoreline drift, they had to be recovered by vessel. H3 fairings allow seawater to enter and sink them, removing the need for recovery and lowering costs.

9.2.5 Sheet insulation

H-IIA fairing insulation, silicone containing glass microballoons, was spray-applied in a dedicated booth. Long drying times could make the booth a production bottleneck. H3 uses preformed silicone-foam sheets bonded to the fairing. Insulation can be installed in the assembly area, removing this production constraint.

9.2.6 Quick-access doors

H3 incorporates access doors that can be closed more quickly, reducing vehicle and payload servicing time.

9.2.7 Synergy across vehicle families

The sinking-fairing design, sheet insulation and quick-access doors were originally developed for Epsilon fairings and adapted for H3. Technologies developed for H3 fairings will also be applied to Epsilon S. These shared developments will support continued, efficient advancement of fairing technology.

10 Propulsion, structures and electrical systems

A launch vehicle consists broadly of three subsystems: propulsion, structures and electrical systems. H3 emphasizes simplification, use of standard components and commonality across all three.

10.1 Propulsion system

The propulsion system comprises engines, propellant tanks, valves, pressure sensors and other equipment that controls engine operation and propellant flow.

Engine simplification includes integrated parts, improved material yield using near-net-shape stock, easier-to-machine geometries and automated processes. Additive manufacturing is one example.

Commercial aircraft and automotive parts are used for valves and sensors. Space-specific parts are expensive because of specialized requirements and low production volumes. Applying reliable, low-cost commercial parts provides substantial benefits. JAXA and private companies jointly assessed their suitability, including parts-selection and safety standards.

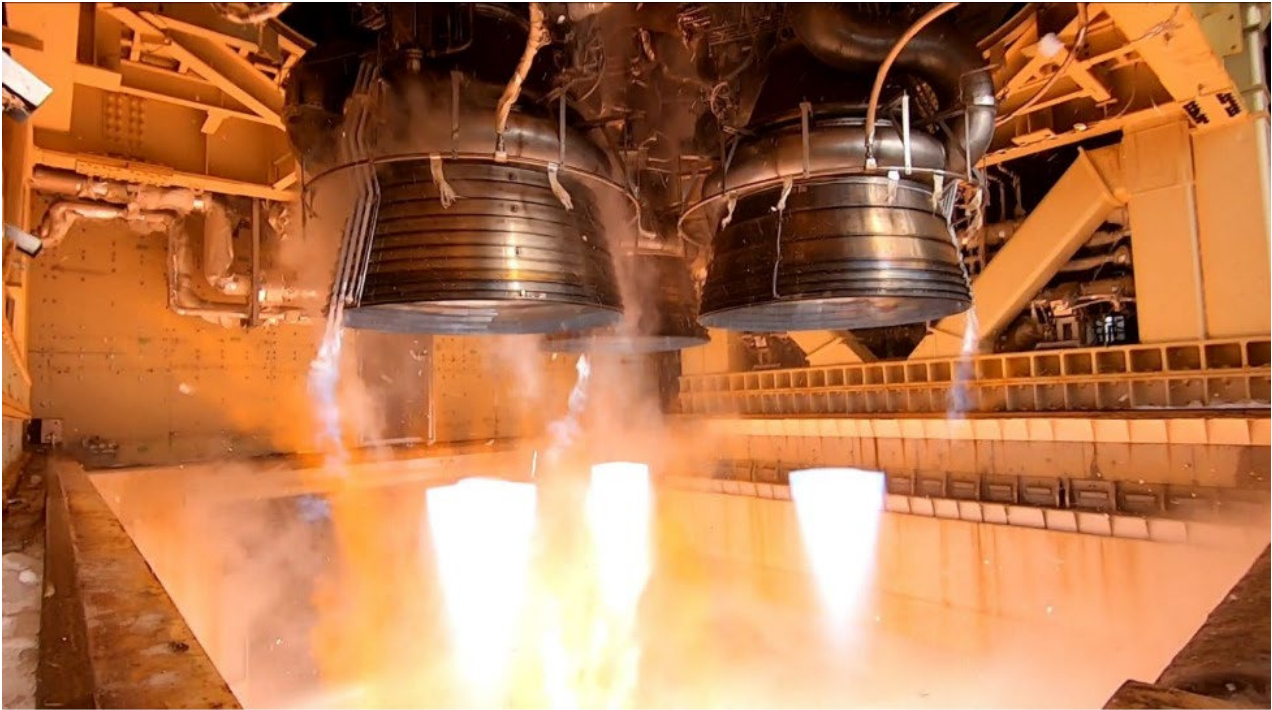
Commonality also extends to power sources: valves and actuators are electrically driven. Eliminating the high-pressure-gas and hydraulic systems previously required simplifies the system, improves reliability and performance, and lowers costs.

Propulsion testing took place at JAXA's Tanegashima Space Center in Kagoshima and Kakuda Space Center in Miyagi. Integrated system tests, including flight-type valves and sensors, were conducted at MHI's Tashiro Test Facility in Akita.

A major first-stage development milestone was the battleship tank firing test (BFT) at Tashiro. Propellant tanks and engines were tested together under ground-simulated flight pressures and temperatures. Except for the tanks, components were the same as flight hardware, and two or three LE-9 engines were fired together in the flight configuration. Eight tests were completed successfully, beginning in January 2019 and, as stated in this section of the source, ending on 13 February 2020.



First-stage battleship tank firing test at MHI's Tashiro Test Facility.

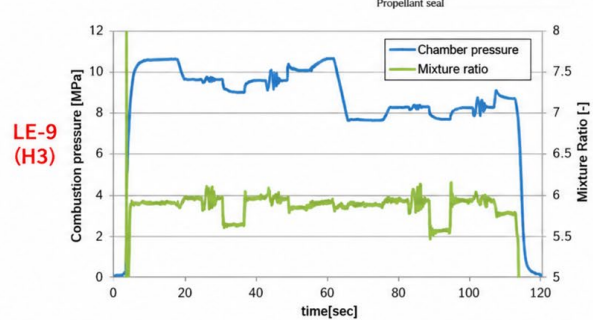
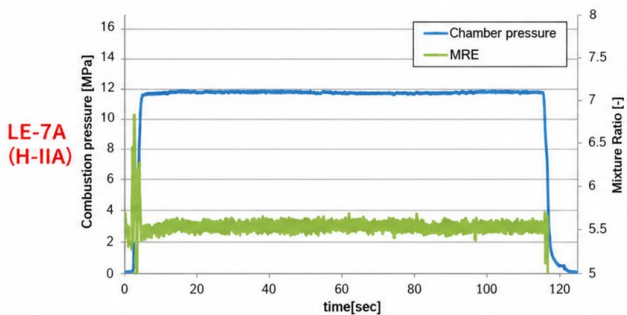
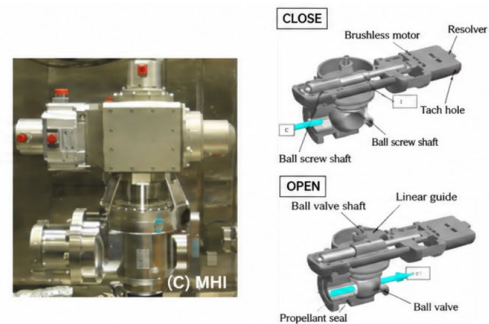


■ Test Process Innovation

- Control to target operating points through feedback
- Development testing: integration of multiple tests (improved efficiency)
- Manufacturing testing: reduced number of tests (no readjustment required)

■ Automated Inspection

- Shorter manufacturing period and launch-site maintenance period



BFT engine firing and electrically actuated valves used on H3. English key: feedback control to a target operating point; integration of multiple development tests for efficiency; fewer manufacturing tests without readjustment; automated inspections; shorter manufacturing and launch-site processing. The plots compare LE-7A and LE-9 combustion pressure and mixture ratio versus time. Electric actuation allows multiple operating points to be checked in one firing. Valve drawing: brushless motor; resolver; tooth brake; ball screw nut/shaft; ball valve shaft; linkage; propellant seal; open/closed positions.

10.2 Structures

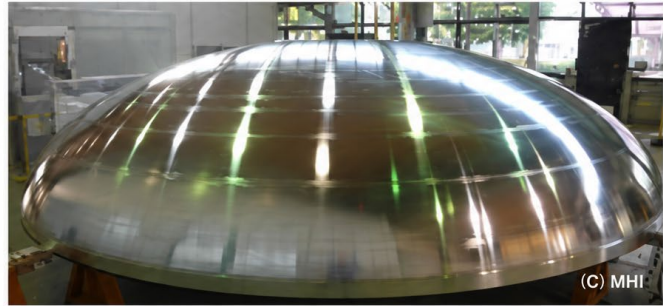
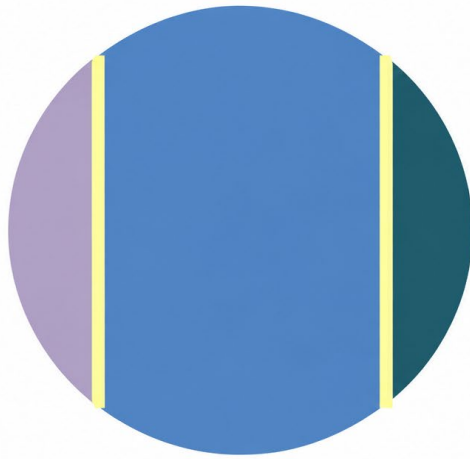
The structural subsystem includes the vehicle's outer structure, fairing, interstage, and supports and attachments for onboard equipment.

Building on existing designs, H3 lowers cost through simpler shapes, fewer specialized materials and automated processes. Propellant-tank dome-forming technology developed for H-IIIB has been extended to produce larger integral domes, reducing part count and cost.

Automated drilling and riveting are used to assemble main structures, including the engine section and interstage. The engine section is common to both two-engine and three-engine configurations, retaining H3's flexibility while controlling cost.

■ Preparation of Material for Tank Dome Forming

- Welding standard-size plates instead of using specially manufactured large-size plates



Propellant-tank dome. English key: preparation of material for dome forming; weld standard-size plates instead of procuring a special large plate. Larger integral formed sections reduce separate parts and joints. © MHI.

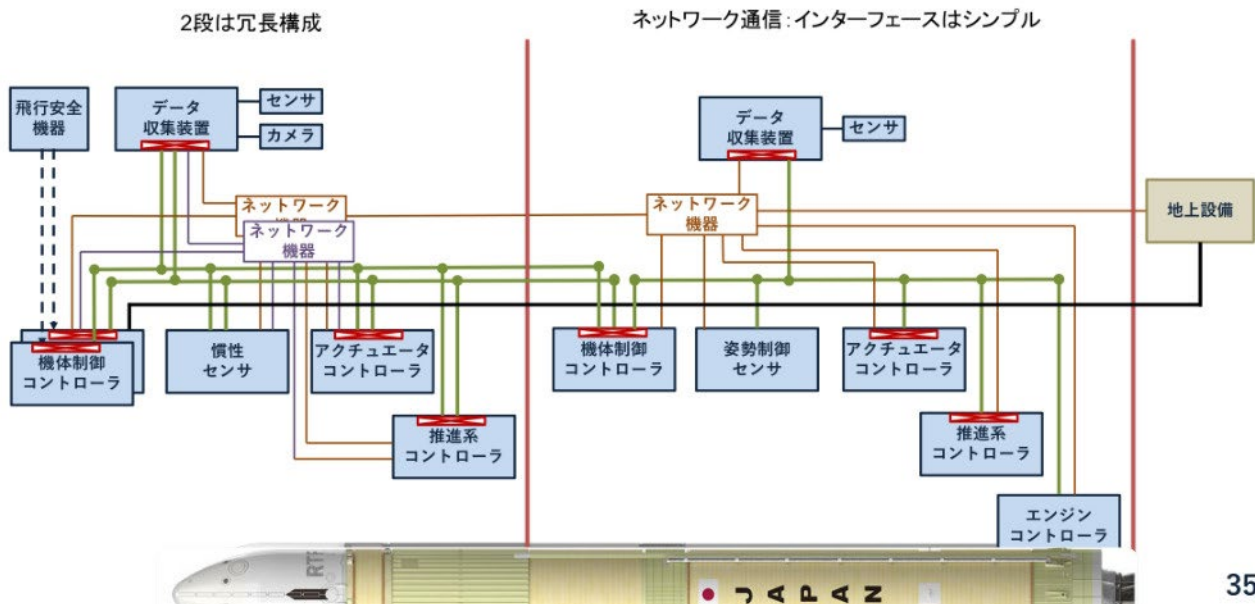
10.3 Electrical system

The electrical system, or avionics, includes attitude-control sensors, computers and communications equipment.

H-IIA/B used direct cable connections between numerous sensors, computers and ground-communications equipment. This required extensive, heavy wiring and substantial connection and inspection work during assembly.

H3 distributes control functions and links avionics through a network. In principle, a network cable replaces numerous individual connections. This reduces mass, simplifies assembly, launch-site inspections and launch operations, lowers costs and facilitates equipment upgrades.

Commercial aircraft and automotive electronic parts are used wherever possible. Suitable part types were selected after radiation-tolerance testing. Development proceeded from breadboard models (BBMs), demonstrating feasibility, through engineering models (EMs).



H3 avionics architecture. English key: second stage has redundant architecture; simple network interfaces; flight-safety equipment; data acquisition equipment; sensor; camera; network equipment; vehicle controller; inertial sensor; actuator controller; propulsion controller; attitude-control sensor; engine controller; ground facilities. The distributed equipment is networked rather than connected by numerous individual cables.

11 Ground facilities (1): launch pad facilities

Engines and vehicle hardware alone cannot reach space. Ground facilities are required for assembly, transport, propellant and electrical supply, and launch.

Facilities around the launch point are called launch pad facilities. Equipment used to track the vehicle by radio or radar and transmit a flight termination command if necessary is categorized as range and flight-safety equipment.

H3 reuses many Tanegashima facilities from H-IIA/H-IIB, with selected facilities newly constructed.

11.1 Takesaki Launch Control Center (LCC)

The H-IIA/B Launch Control Center, commonly called the blockhouse, was near the Vehicle Assembly Building (VAB) inside the launch area. H3's new Takesaki LCC is approximately 3 km from the pad, reducing the burden on personnel.

It is located beside the Takesaki Range Control Center (RCC), which directs launch operations as a whole, improving coordination. Automation has reduced the number of launch-control personnel to approximately one-third to one-quarter of the H-IIA level.



Takesaki Launch Control Center (LCC).

11.2 Movable launcher (ML)

H-IIA/B's movable launcher enabled efficient successive launches. H3 retains this concept, introducing major changes relative to the H-IIB ML. The new launcher is named ML5, continuing the naming tradition of ML1 for H-IIA and ML3 for H-IIB.

11.2.1 Reducing wind-induced motion

Wind passing around the ML masts can create wakes that cause vehicle motion, including during transport. H3 changes the cross-sectional shape of the upper masts and removes the overhead bridge linking the two masts, minimizing this effect.

11.2.2 Larger exhaust opening and retractable launch supports

The ML has an opening for engine and booster exhaust. H-II/B used five small openings; H3 uses one large opening, with three main objectives.

1. Reduce thermal damage. The larger opening limits heat damage to structures. Removing propellant, high-pressure-gas and water piping and fixed supports from the upper deck makes post-launch repairs easier and shortens turnaround.
2. Reduce launch acoustics. The larger opening reduces sound generation and the amount of suppression water required. H3 eliminates the ML-mounted water-deluge equipment and uses only ground-mounted equipment. This also removes the need to connect water piping after the ML reaches the pad.
3. Change how the vehicle is supported. H-IIA/B stood on four supports on top of the ML. H3 uses four movable supports on the sides of the larger opening. This allows the vehicle to sit lower inside the ML, fitting in the existing VAB despite being 6.4 m taller than H-II/B. The supports retract into the ML structure, avoiding exhaust damage and reducing post-launch repairs.



Retracting the launch supports. The supports withdraw into the launcher structure to protect them from exhaust.

These changes substantially shorten repair time and reduce the launch interval from approximately two months to one month or less.

11.2.3 Hold-down system

Unlike H-IIA/B, the H3 lineup includes H3-30, which uses only liquid-propellant engines and no solid boosters.

On H-IIA/B, unignited solid boosters acted as ballast, preventing liftoff after the liquid engines started. Likewise, on H3-22 and H3-24, two LE-9 engines alone do not produce enough thrust to exceed vehicle weight; liftoff occurs only after SRB-3 ignition increases total thrust above that weight.

For H3-30, three LE-9 engines produce more thrust than vehicle weight. A hold-down system must therefore restrain the vehicle until all engines are confirmed to be operating normally.

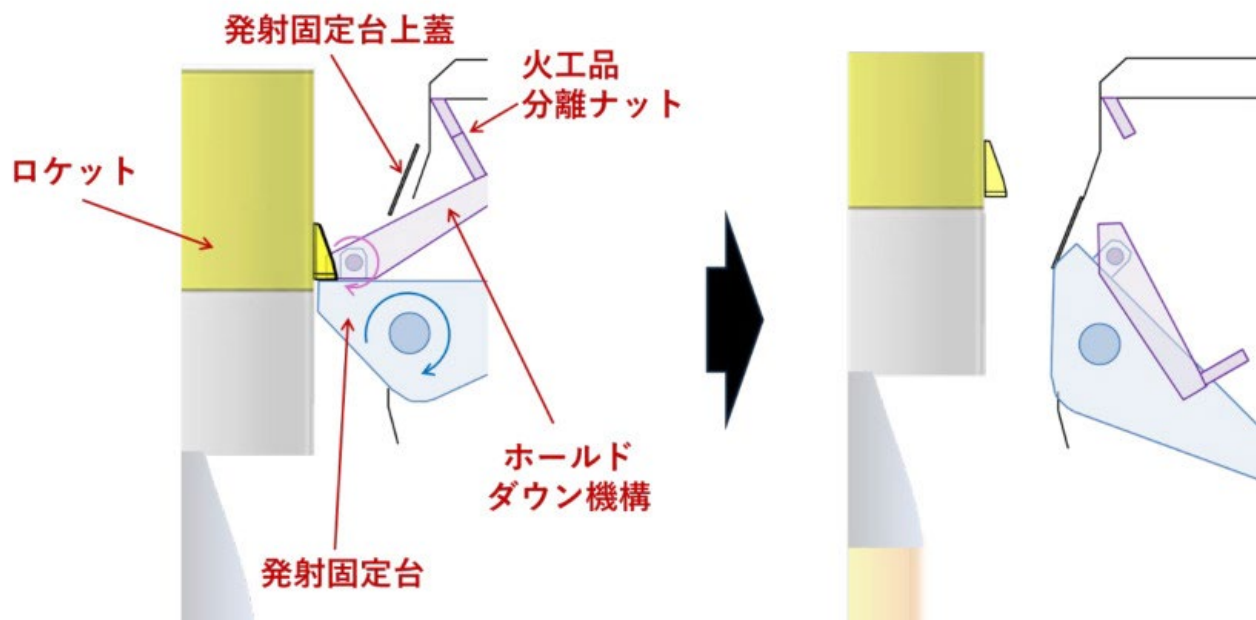
Four large claw-like mechanisms around the first-stage engine area clamp launch fittings on the vehicle. After all three engines ignite and pass health checks, the clamps release. Release timing must be synchronized to

avoid adverse liftoff motion. The mechanisms then retract immediately into the launcher, and protective covers close to prevent exhaust damage.

The system first supported a launch on Flight No. 6, the H3-30 test vehicle.



H3 movable launcher.



Hold-down system operating sequence: restrain vehicle; release after normal engine startup; retract mechanism; close protective cover.

11.2.4 Vehicle restraint system

The vehicle restraint system prevents large wind-induced motion while H3 is on the ground. H3 is larger than H-IIA and particularly sensitive to wind when its tanks are empty. Nonlinear resonance can cause motion to increase sharply above a certain wind speed, potentially leading to overturning. H3-24's four boosters increase mass and lower the natural frequency, making resonance possible at relatively low wind speeds.

The system is positioned between ML5's two masts. Semicircular supports embrace the vehicle, moved vertically by arm-like mechanisms. To engage, the arms rise on either side and the supports rotate around the middle of the first stage. Ground-supplied hydraulics drive the system, which has controls that are designed to tolerate two faults.

During operations, the system supports the vehicle while it moves from the VAB to the pad. After piping connections and propellant loading, the heavier fueled vehicle is less wind-sensitive, and the restraint is released and stowed.

During Flight No. 6's July 2025 CFT, three raising/lowering cycles verified engagement, release and stowage. Operational use began with Flight No. 7 in October 2025.

H3-30 is less susceptible to motion and is held by the hold-down system, so the vehicle restraint is generally not used. Earlier H3-22 and H3-24 operations also used hold-down restraints; the vehicle restraint system removed that need for those configurations.



Vehicle restraint engaged, and after release.

11.3 Movable-launcher transporters: dollies

Two coordinated transporters, commonly called dollies, move the ML and vehicle from the VAB to the pad. The H-IIA/H-IIB dollies were more than 20 years old. For the new system, JAXA defined requirements, MHI established detailed specifications, and Nippon Sharyo designed, developed and manufactured the transporters.

Each is approximately 25 m long, 3 m high and 3 m wide. Working as a pair, they carry up to 1,460 tonnes. Maximum speed is approximately 2 km/h; the roughly 500 m journey takes about 30 minutes.

A driver occupies the cab, but movement is primarily automatic, with sensors following magnets embedded in the ground. To permit connection of pad piping, stopping accuracy is within ± 25 mm longitudinally and

laterally. Level accuracy is maintained within 0.2° , and acceleration during speed changes remains below 0.08 g for gentle, safe transport.

The system provides high reliability, redundancy and maintainability. Automated checks and component-wear management have halved annual maintenance costs.



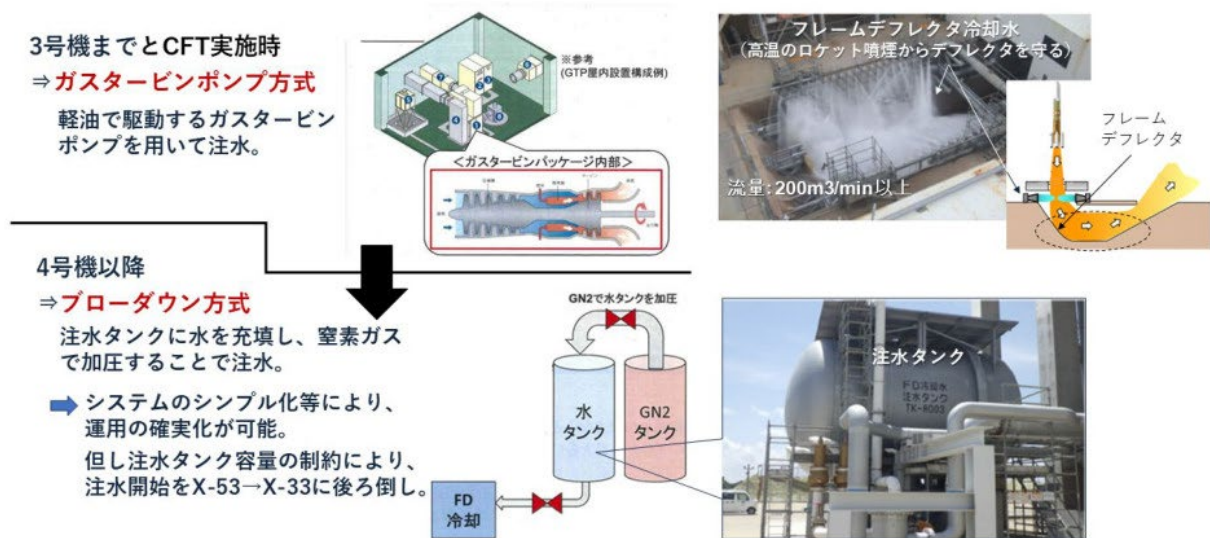
New H3 movable-launcher transporters.

11.4 Cooling-water deluge system

Below the launcher, a flame deflector directs exhaust sideways into flame ducts, which route it safely away. The cooling-water system supplies large amounts of water to protect these structures from high-temperature, high-speed exhaust. Steam also reduces engine noise and acoustic loads transmitted to the vehicle and payload.

Through Flight No. 3, H3 used diesel-fueled gas-turbine pumps. From Flight No. 4 onward, a blowdown system uses nitrogen gas to pressurize water tanks and force water out. This simplifies equipment and improves operational dependability. Water-deluge start moved from 53 seconds before launch to 33 seconds before launch.

※1 フレームデフレクタ：ロケットの噴流の方向を変えるための耐火コンクリートの壁

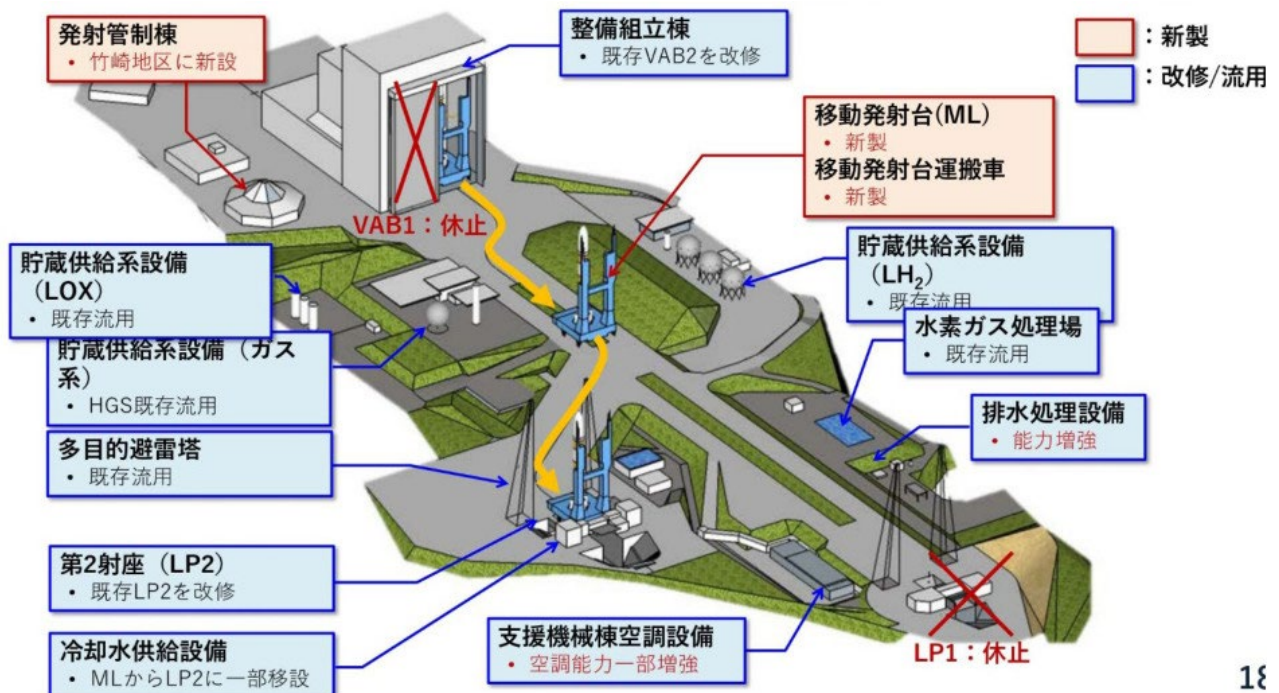


Cooling-water deluge system. The upgraded system uses nitrogen-pressurized water tanks instead of gas-turbine pumps.

11.5 Refurbishment and reuse

H3 reuses the refurbished northern portion of the existing assembly facility, VAB2, and Launch Pad 2 (LP2), previously used for H-IIB. Wastewater treatment and support-building air conditioning have been upgraded.

Many existing facilities have also been refurbished and reused, including liquid-oxygen and liquid-hydrogen storage and supply systems, gas storage and supply systems, and facilities for handling hydrogen gas generated by cooling and evaporation.



18

Launch pad facilities. English key: red—new; blue—refurbished/reused. New: Takesaki LCC, ML5 and transporters. Refurbished/reused: VAB2, LP2, LOX/LH₂ storage and supply, high-pressure-gas supply, multipurpose evacuation tower and hydrogen-gas treatment. Increased capacity: wastewater treatment and part of support-building air conditioning. Cooling-water equipment is partly relocated from ML to LP2. VAB1 and LP1 are marked out of service.

11.6 Addressing aging infrastructure

Many reused facilities are aging: older structures were built approximately 20 years ago, and even newer ones are more than 10 years old. The coastal environment accelerates corrosion of piping and valves. In January 2020, corroded piping caused postponement of H-IIA Flight No. 41.

Regular repairs maintain operations but increasingly constrain the space-development budget. Fundamental renewal is needed to support stable H3 operations and future next-generation launch vehicles.

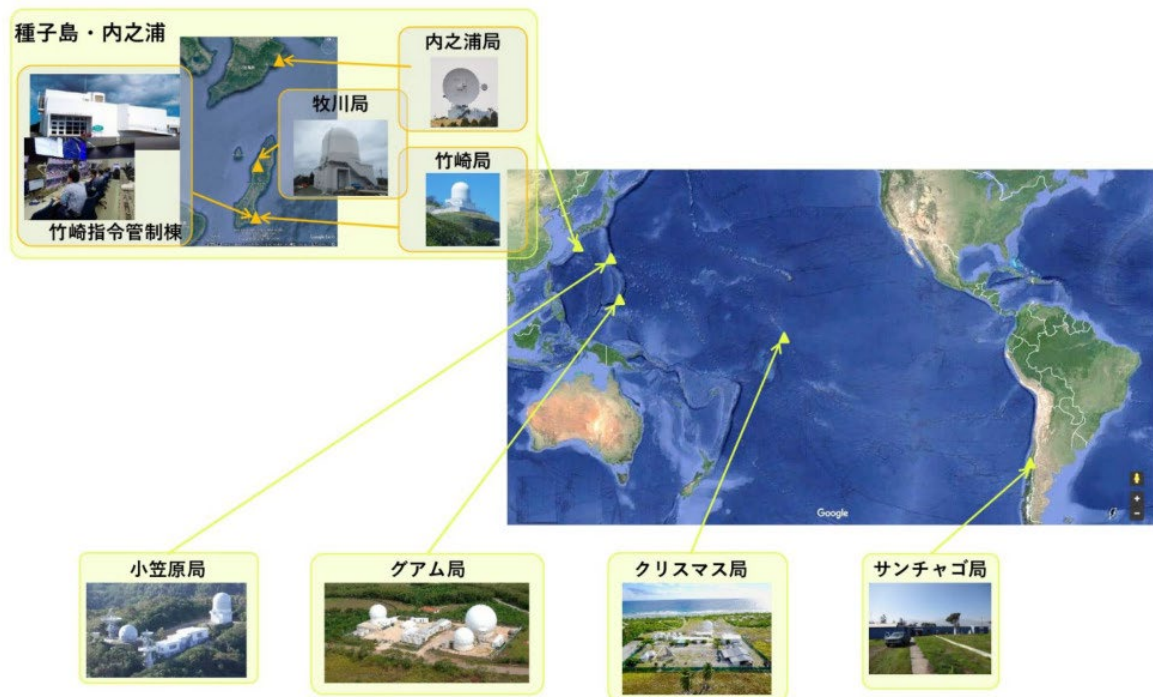
12 Ground facilities (2): range facilities

12.1 Renewal of range and flight-safety systems

After liftoff, the Takesaki RCC continuously monitors H3, assessing flight safety and mission achievement. Tracking stations along the flight path receive and transmit radio signals. Because Earth is curved while radio signals travel approximately in straight lines, stations hand over tracking successively.

JAXA has stations at Tanegashima, Uchinoura, Ogasawara, Guam in the United States, Christmas Island (Kiritimati) in Kiribati, and Santiago in Chile. Some have been operating since H-IIA's first test flight approximately 20 years ago, and their antennas and equipment were aging. H3 development provided an opportunity to renew these long-life facilities in support of flexibility, reliability and low cost.

The most significant change is remote operation of each station from the RCC, reducing station staffing and launch costs. Automated RCC operations also ease personnel workload, improve maintenance efficiency and lower costs. Optimizing performance between H3's onboard equipment and the stations allows smaller antennas and fewer receiving devices.



H3 tracking network: Tanegashima/Takesaki, Uchinoura, Ogasawara, Guam, Christmas Island (Kiritimati) and Santiago.



Takesaki tracking station at Tanegashima Space Center.

12.2 Renewal challenges and the COVID-19 pandemic

Upgrades had to proceed alongside H-IIA/H-IIB operations. At Guam, for example, the new H3 antenna had to fit within limited space already occupied by H-IIA/B antennas.

Final integrated range-system tests took place in late March 2020. COVID-19 restrictions prevented overseas personnel from entering or staying in Guam and Christmas Island. Because the systems supported remote operation, testing was conducted entirely from the RCC with manufacturers' technical support available on call.

After installation, tracking tests were carried out during H-IIA Flights Nos. 41 and 42 and H-IIB Flights Nos. 8 and 9.

13 Ground facilities (3): payload processing facilities

Payload facilities prepare and inspect spacecraft before launch. Tanegashima previously had four such facilities, including STA1. SFA3 was newly constructed to support larger payloads, multiple payloads and higher annual launch rates.

13.1 Third Spacecraft and Fairing Assembly Building (SFA3)

SFA3 enables parallel processing of larger and multiple spacecraft. It is outside the 3 km launch exclusion zone, allowing processing of subsequent payloads to continue while another launch is underway.

Construction began in FY2020 and was completed in FY2023. Operations began with Test Flight No. 2. Existing facilities such as STA1 will continue operating in coordination with SFA3, according to each payload's needs.

Spacecraft built by domestic and overseas manufacturers enter SFA3 after delivery to Tanegashima. Testing, propellant loading and fairing encapsulation take place in the following rooms:

Room	Function
Airlock	Initial receipt of spacecraft and fairings; cleaning of container exteriors, unpacking and transfer to assembly rooms
Fairing assembly room	Spacecraft/fairing integration and preparation for transfer to the VAB
Payload assembly room	Post-delivery assembly, functional checks and propellant loading
Checkout room	Remote control and monitoring

■ 第3衛星フェアリング組立棟(SFA3)の概要

- 人工衛星の大型化、複数化、年間多数機打上げに対応するため新整備
- 射場搬入後、衛星の試験から推進薬充填、フェアリング格納までの一連の作業を行う
- 射点から3kmの警戒区域外に整備、他ミッションの影響を受けない



SFA3外観

■ 打上げまでの流れ（写真はTF1他、別ミッションのもの）



2

SFA3 and the payload processing flow: acceptance/assembly; propellant loading; fairing encapsulation; transport to the VAB; launch.

13.2 Other facilities

First Spacecraft Test and Assembly Building (STA1)

STA1 supports approximately 1-tonne small and medium spacecraft: unpacking and visual inspection after transport, assembly, radio-frequency testing, functional checks and compatibility tests. Spacecraft remain in a test room maintained at Class 10,000 cleanliness and required temperature/humidity conditions. Test equipment in an adjacent checkout room performs functional testing.

Second Spacecraft Test and Assembly Building (STA2)

STA2 supports similar activities for large spacecraft. It can also monitor spacecraft in the fairing assembly building or launch tower and remotely control and monitor hydrazine loading and bipropellant apogee-motor fueling in the fairing assembly building.

The building contains an airlock, spacecraft test room, tool/jig room, measurement preparation room, checkout room and unpacking room. Its 25 m-high test room maintains Class 100,000 cleanliness. Spacecraft delivered in separate units are assembled and inspected there, including alignment measurements and electrical/functional checks, with remote operation from the checkout room.

Fiber-optic cables and antenna equipment connect STA2 with the VAB and ML, allowing checks and monitoring after spacecraft transfer. Antenna links also connect it to Masuda Tracking and Communication Station. An extension completed in March 1997 allows two large spacecraft to be processed simultaneously.

Spacecraft and Fairing Assembly Building (SFA)

SFA includes a 30 m-high, Class 100,000 assembly room, airlock, propellant loading room, control/monitoring room and fueling-equipment room.

Following work in STA1 or STA2, activities include high-pressure leak tests of spacecraft propulsion systems, propellant loading, pressurization, installation of pyrotechnic devices and spacecraft solid motors, and final fairing encapsulation.

Second Spacecraft and Fairing Assembly Building (SFA2)

SFA2 combines STA2 and SFA functions, covering post-delivery testing, propellant loading and fairing encapsulation. It includes an airlock, payload assembly room, fairing assembly room and checkout room.

The payload assembly room is 28 m high and maintains Class 100,000 cleanliness. It supports spacecraft assembly, functional verification, propellant loading, pressurization and inspection. The fairing assembly room is 35 m high, also maintains Class 100,000 cleanliness, and supports spacecraft/fairing integration and preparation for transfer to the VAB.

Work is remotely controlled and monitored from the checkout room. Fiber-optic cables connect SFA2 with the VAB and ML for checks and monitoring after transfer. In 2009, approximately 1,000 m² of additional work area was added to accommodate the large KOUNOTORI (HTV) cargo spacecraft.

14 H3 cost reduction initiatives

To secure international competitiveness and low prices, H3 incorporates cost reduction across design, manufacturing, operations and ground facilities.

14.1 Cost reduction at every stage

Cost was considered from concept through development and future operations. At the concept stage, the program established low-cost concepts, defined price requirements for individual components and examined the entire supply chain, from materials and procurement through manufacturing, inventory, delivery and service.

During development, low-cost designs and technologies were adopted, manufacturing methods improved and design reviews included cost assessments against the original targets. In operations, series production, consolidated procurement and continuing efficiency improvements were planned.

14.2 Fewer components

LE-9's simple expander-bleed cycle and AM parts reduce component count by approximately 20% relative to LE-7A at the component level.

14.3 Takt production

H-IIA was manufactured on a build-to-order basis. H3 balances the time required for parts installation, assembly, inspection and shipment, moving toward line production similar to cars and aircraft.

Takt production divides the vehicle into four segments, with movement and processing repeated in synchronization. This increases manufacturing efficiency, shortens lead time and lowers costs.

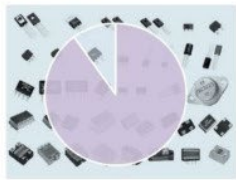
14.4 Automotive components

Conventional valves, sensors and electronics used specialized space components which had low production volumes that made them expensive. H3 uses aircraft and automotive commercial parts wherever possible. Their reliability and mass production offer major benefits in reliability and cost.

Part types were selected through radiation-tolerance assessments. JAXA and private companies jointly evaluated selection criteria and safety requirements.

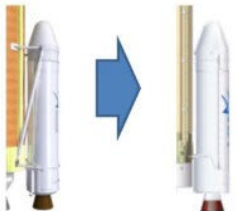
■ 車載電子部品の適用

- 全数の90%



■ シンプルなシステム設計

- 固体ブースタ結合構造の簡素化



■ 新型エンジンLE-9

- 大推力
- オール電化制御
- 部品点数の削減
- 3Dプリンタ



低価格 柔軟性

3Dプリンタ



造形例



24

Low-cost design specifications. English key: automotive electronic parts—90% of the total shown in the source graphic; simple system design—simpler solid-booster attachment; new LE-9—high thrust, all-electric control, fewer components and 3D printing; low cost; flexibility; 3D printer; example printed part. © MHI.

14.5 Additive manufacturing

LE-9 uses AM primarily for piping, valve/injector/combustion-chamber structures and casing parts. An injector supplies and mixes liquid hydrogen and liquid oxygen using approximately 500 coaxial tubular elements. Oxygen flows through each central tube; hydrogen flows through the surrounding annular gap.

Conventional manufacturing required machining separate components for 500 elements, joining them and attaching them to the injector body. AM forms the core integrally, reducing extensive machining and assembly work and potentially cutting total cost by more than 50%.

■ 製造プロセスやサプライチェーンの見直し

■ 加えて、Additive Manufacturing（3D造形）等の低コスト技術を導入



26

Manufacturing cost reduction. English key: review manufacturing processes and supply chain; introduce AM and other low-cost technologies; die forging/flow forming of large chamber stock; choose regenerative-cooled versus radiatively cooled nozzle material according to mass/cost balance; AM for piping, valve/injector/chamber structures and casing parts; turbine exhaust film cooling increases nozzle expansion ratio; electric valves control mixture ratio and thrust; open impeller permits a single-stage hydrogen impeller; integral turbine construction. © MHI.

14.6 SRB-3

As described in Chapter 8, domestic motor-case production, innovative attachment/separation systems and revised propellant formulations, and raw materials reduce SRB-3 costs.

14.7 Lower ground-facility maintenance costs

A flatter ML upper deck reduces post-launch repair work. Automated transporter checks and component-wear management have halved annual maintenance costs.

Tracking stations in Japan and overseas can be operated remotely from the Takesaki RCC, reducing station personnel and launch costs. Automated operation lowers workload and improves maintenance efficiency. Smaller antennas and fewer receivers, enabled by balancing vehicle and tracking-station performance, further reduce maintenance costs.

15 The challenges of launch vehicle development

Like other launch vehicle programs, H3 encountered numerous technical challenges. Development of LE-9 proved particularly difficult. Although the program prepared extensively through LE-X prototype design and component testing, multiple issues still emerged.

15.1 Revising the engine qualification plan

The first major setback occurred in 2019, during repeated firing tests of flight-type development engines intended to collect basic functional and performance data. The AM injector showed abnormal combustion behavior, and FTP turbine rotor blades showed fatigue fracture surfaces resulting from repeated loading.

In October 2019, the program adopted a two-stage qualification plan:

- Type 1: use a proven machined injector and operate outside resonance regions.
- Type 2: use an AM injector and redesign the engine to eliminate the resonance regions themselves.

15.2 First development-plan revision and launch postponement

Type 1 qualification firing tests (QT) began in February 2020. QT uses test engines manufactured with flight-equivalent designs and processes to verify function, performance and life, including demanding conditions that could occur in service. Eight firings totaled 1,098.5 seconds.

An internal inspection after the eighth firing, on 26 May, identified two issues. The development plan was revised to address them, moving the anticipated Test Flight No. 1 launch to FY2021 and Test Flight No. 2 to FY2022.

15.2.1 Openings in the combustion-chamber inner wall

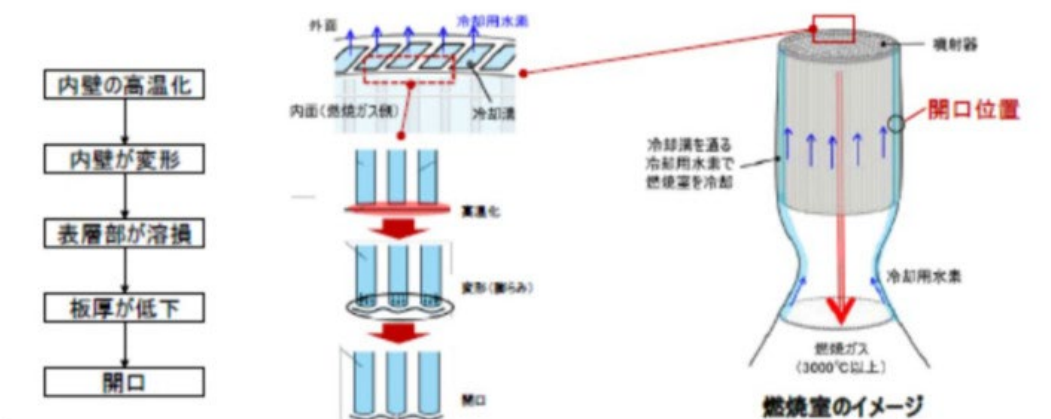
LE-9's chamber has inner and outer walls, with many cooling channels between them. Part of the liquid hydrogen flows through these channels, cooling the chamber and generating turbine-drive gas.

Inspection the day after the eighth firing found 14 openings extending through the inner wall into the cooling channels. The largest were approximately 0.5 mm wide and 10 mm long along the channel direction.

Investigators inferred that the inner wall exceeded its design temperature during the high-temperature operating condition. It deformed, its surface melted or eroded, its thickness decreased and openings developed.

The test used operating temperatures approximately 100°C above normal to verify tolerance of the worst credible conditions arising from manufacturing variation, changing flight acceleration, and changes in propellant quantity and pressure.

Data evaluation and simulation indicated that repeated high-temperature cycles during steady combustion accumulated plastic deformation in one direction, ultimately creating openings. The corrective approach was to operate below the wall-temperature threshold at which deformation progresses significantly, approximately 1,100 K.



Combustion-chamber cooling structure and inner-wall damage mechanism: hot gas; thin chamber wall; hydrogen cooling channels; accumulated plastic deformation and local opening.

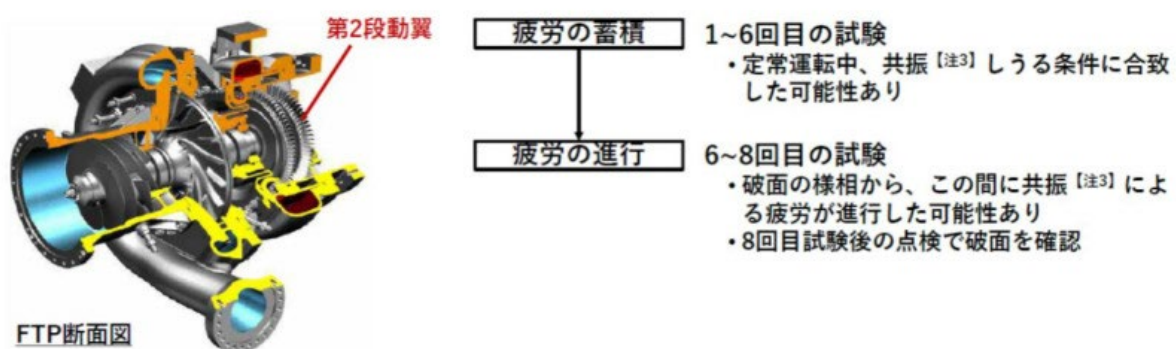
15.2.2 FTP turbine fatigue fracture surfaces

LE-9 has separate turbopumps for hydrogen and oxygen. Fatigue fracture surfaces were found in two of the FTP second-stage turbine's 76 rotor blades.

This issue was assessed as independent of the combustion-chamber wall openings; the two were coincidentally identified at the same time.

Blade-vibration tests directly measured rotor-blade strain during turbopump operation. Combined with detailed inspection and analysis, they indicated that resonance had accumulated fatigue damage. Similar fatigue had appeared in earlier flight-type development tests and had been addressed by avoiding the affected operating region. The new issue occurred in a different region previously considered free of resonance.

The turbine was redesigned to move all structural natural frequencies outside the operating region. The OTP was also redesigned along the same lines as a precaution.



Fuel turbopump and turbine vibration. Rotor blades and the surrounding flow interact; vibration testing measures the blade response.

15.3 Second development-plan revision and launch postponement

Corrective measures for the chamber-wall openings were established. The turbopump modifications showed promise, but validation testing revealed self-excited vibration—flutter—in the first-stage turbine disk, arising from coupling between the blades/disk and the surrounding fluid.

The redesigned OTP also exhibited a vibration response requiring attention, attributed to nonuniform turbine-inlet flow and related effects that had not previously emerged as a problem. The plan was revised again, and Test Flight No. 1 could no longer launch in FY2021.

15.4 Completing Type 1

To minimize risk, multiple countermeasures were developed in parallel, permitting rapid implementation of alternatives. Proven design principles that had addressed resonance were retained where possible to avoid introducing new risks, while more fundamental redesigns were also prepared if incremental changes proved insufficient.

Options were assessed in order of effectiveness, design maturity—including heritage and novelty—and manufacturing schedule. Multiple design teams worked in parallel, and a cross-company Turbopump Development Promotion Office conducted technical evaluations.

Designs were manufactured successively and verified through blade-vibration tests at Tanegashima and Kakuda. Testing began with Option 0, the smallest change to the existing design. Subsequent options were held ready at appropriate stages, from completed parts to finished designs.

Turbopump	Option	Approach
FTP	0	Further machine the turbine that showed partial improvement in October 2021, increasing damping
FTP	1	Increase disk stiffness based on the turbine used in June 2021
FTP	2	Design/manufacture a derivative of Option 1, with further machining incorporating its test results
FTP	3	Study multiple designs in parallel
OTP	0	Increase damping based on the turbine that showed partial improvement in December 2021
OTP	1	Suppress turbine-inlet flow nonuniformity, based on the June 2021 turbine
OTP	2	Fundamentally alter natural frequencies, based on the June 2021 turbine
OTP	3	Study multiple designs in parallel

The selected measures were FTP Option 0 and OTP Option 1. QT turbopumps were then manufactured, engine qualification firings completed, and acceptance firing tests (AT) conducted, completing Type 1.

15.5 Type 1A development and status

After Type 1 was completed, evaluation continued to select optimum Type 2 specifications for Test Flight No. 2 and later vehicles.

From February 2023, FTP Option 2 blade-vibration tests suppressed the nonsynchronous response observed in Option 1, but some resonance responses remained large. Further improvement and testing were required, so Type 1A was prepared for interim launches.

Type 1A provisionally retains the Type 1 FTP and injector, uses the same permanent OTP corrective design as Type 1, and applies permanent designs to other components, principally those already demonstrated in development tests.

Eight Type 1A qualification firings were conducted from 10 August to 12 December 2023. They verified that the assembled engine and major components had flight-suitable function and performance and that manufacturing/inspection equipment and processes were appropriate. The planned data were obtained.

To launch Test Flight No. 2 as early as practicable, QT was split into an initial and final phase, with acceptance testing of its second engine between them. Operating-range verification, more susceptible to rework, was conducted first; life demonstration followed, reducing the risk of late rework requiring another AT.

Test Flight No. 2's first engine was Type 1, using hardware prepared as a backup for Test Flight No. 1. Its second was Type 1A, which completed AT as planned and was integrated into the vehicle.

Type 1A first flew on Test Flight No. 2 and was also used on Flights Nos. 3, 4, 5 and 7. These flights proceeded as planned, building operational experience. Flight No. 8 analysis also found that the first-stage engines and propulsion system functioned normally from ignition to cutoff. Type 1A continues to support H3's initial operations. See Section 6.7 for the engine comparison.

15.6 Type 2 development and status

Type 2 development proceeds alongside Type 1A operations. Permanent corrective designs for the FTP, injector and other principal components are intended to establish specifications suitable for stable operation and series production.

Component tests, blade-vibration tests and technical-data tests have been conducted repeatedly. During 2024–2025, Tanegashima tests collected data on turbine blade responses, engine operating characteristics and major-component function and performance.

Type 2 qualification engine firings began in March 2026, addressing technical data, functional/performance verification and life demonstration. The program incorporates lessons from LE-9 development while establishing the specifications and qualification evidence needed for continued H3 operations.

15.7 Challenges unique to LE-9

Developing an expander-bleed engine at LE-9's thrust level was a world-first undertaking. Its recurring technical challenges reflect that ambitious step.

The chamber's inner wall must be extremely thin to transfer heat efficiently to hydrogen and generate sufficient turbine-drive energy. Heat absorption is central to the cycle. LE-9 uses a wall approximately 0.7 mm thick, close to technological limits, which contributed to the difficulty of preventing wall openings.

The turbine-drive gas has relatively low enthalpy. Extracting high power from it requires a very high turbine expansion ratio of 8.5, with demanding conditions including supersonic gas flow and high inlet pressure.

The integral blade-and-disk design, adopted partly for cost reduction, is also somewhat more prone to resonance than LE-7A's separately manufactured blades fitted into a disk.

An engine with this performance is essential to H3's flexibility, reliability and low cost. These difficulties were therefore unavoidable challenges to overcome.

16 Test flight missions

Vehicle and launch-site construction and ground testing do not complete development. Some aspects can only be understood through flight.

H3 must therefore launch toward its target orbit, acquire and transmit data from liftoff through mission completion, and undergo evaluation and final verification. The results complete development.

The test missions also establish the entire process leading to launch: manufacturing, launch-site processing, payload integration and countdown. After satisfactory final verification, operational service begins, with a future transition to private-sector-led launches by MHI. Launch processes will continue to be refined after the test missions.

16.1 What is verification?

Systems engineering is central to test-flight verification. Launch vehicles are complex systems spanning rocket engineering, mechanical engineering, electrical/electronic engineering, robotics and other disciplines. Systems engineering provides the multidisciplinary approach and tools needed to make such systems succeed.

H3 begins with requirements analysis, followed by integrated-system design encompassing vehicle and facilities, then separate system designs and individual component designs. Decomposing the system in this way reduces risk.

After manufacturing, testing builds upward in the opposite direction: component tests, vehicle and facility system tests, then integrated-system tests. Validation also checks whether the design achieves its intended purpose.

Verification confirms that each design has been realized as intended. Testing every detail exhaustively would exceed available time and cost, so the verification program balances scope with development scale, following the principle “Test as you fly, fly as you test.” Flight testing is the final verification step and also supports validation of the overall design.

■ 要求マネジメントと検証計画

■ 要求分析

■ システムの分割

- ・ 困難は分割せよ
- ・ 機能の割付と設計

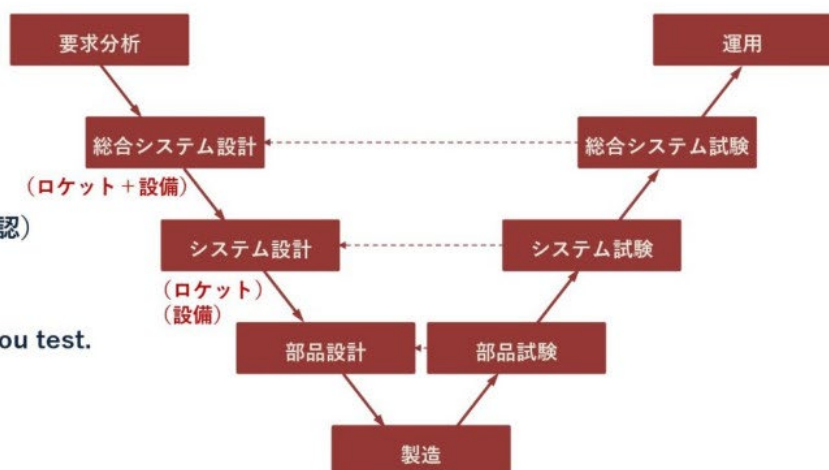
■ 統合と検証

- ・ Verification : 検証
(Validation : 妥当性確認)
- ・ 段階的に統合し、検証
- ・ 開発規模とのバランス
- ・ Test as you fly, fly as you test.
- ・ 最終検証は、試験飛行

■ 運用コンセプト

■ Think the end before the beginning.

- ・ まずはトリセツ



Systems engineering V-model. English key: requirements management and verification planning; requirements analysis; decompose difficult systems; allocate functions and design; integration and verification; staged integration; balance the development scale; “Test as you fly, fly as you test”; final verification through test flight. Design path: requirements analysis → integrated vehicle/facility design → separate system design → component design → manufacturing. Verification path: component

tests → system tests → integrated-system tests → operations. Operational concept: “Think the end before the beginning”—define operating instructions first.

17 Manufacturing H3

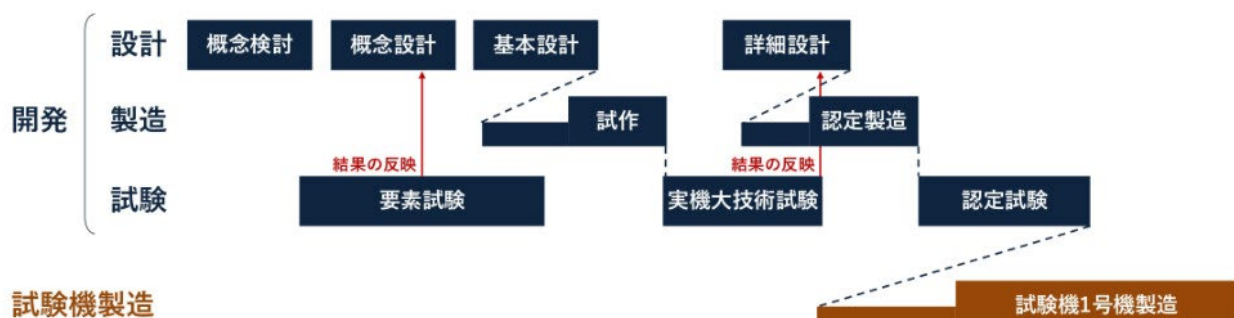
17.1 Coordinating development and manufacturing

Many industrial products enter mass production only after development and production lines are complete. Launch vehicle projects have limited schedules and cannot always wait for one phase to finish before starting the next.

H3 therefore uses fast-tracking: beginning a subsequent process before the preceding one is fully complete. This shortens the schedule but can require rework if an earlier process later reveals a problem.

To manage this risk, detailed decision gates determine when enough work has been completed to safely proceed, even though the preceding phase is not yet finished.

- プロジェクト：有期性（限られた時間）
- 対応策：ひとつの工程が終わる前に、次工程を開始
 - ファスト・トラッキング
 - 手戻りのリスクマネジメント＝きめ細かなゲート



7

Coordinating development and manufacturing through overlapping processes and decision gates.

17.2 Supply chain

Company	Principal locations
MHI	Nagoya, Aichi; Tashiro, Akita
IHI	Mizuho, Tokyo
IHI Aerospace	Tomioka, Gunma
Kawasaki Heavy Industries	Harima, Hyogo

17.3 New technologies introduced by MHI

H3-30S targets a vehicle price half that of H-IIA, under specified conditions in steady operations after manufacturing stabilizes, based on 2014 economic conditions. It also aims to halve the launch interval from two months to one month.

To achieve low cost and flexibility, MHI, the manufacturing prime contractor, has invested in factory expansion and a production approach closer to the assembly lines used for cars and aircraft.

Manufacturing retains H-IIA/H-IIB technology while balancing changes and new technologies needed for competitive pricing. Common principles include simplification, commonality and standard components, with automated assembly, machining and inspection.

AM is applied to complex LE-9 components, including the injector, to lower costs and shorten manufacturing time. Structural and propulsion components build on existing specifications while simplifying shapes, reducing specialized materials and automating processes. Examples include deep-drawn integral tank domes, automated drilling/riveting of main structures, and extensive use of commercial avionics components.

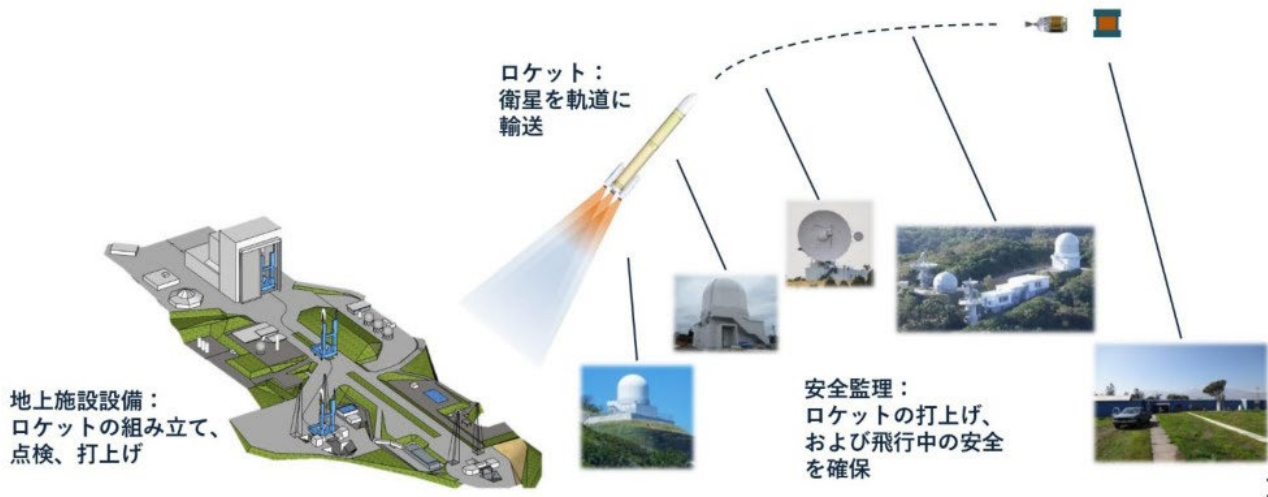
Private-sector ingenuity and manufacturing efforts are essential to achieving H3's objectives.

18 Launch site processing

18.1 H3 integrated system

Ground facilities and launch safety management are as important and complex as the vehicle. The vehicle carries the payload into orbit; ground facilities support assembly, inspection and launch; safety management ensures safe launch and flight.

Together they form the integrated system, or system of systems. If any element is missing or inadequate, launch cannot proceed.



H3 integrated system: launch vehicle, ground facilities and launch safety management.

18.2 Launch site operations

Each system can be tested separately, but integrated testing requires an assembled vehicle, inspections and preparation approaching launch conditions. H3 test missions therefore perform Vehicle On Stand (VOS), followed by three major tests: F-0, special inspections and CFT.

18.3 Vehicle On Stand (VOS)

VOS is the assembly of the vehicle on the movable launcher. Components manufactured at partner factories are delivered to Tanegashima. The first stage is erected on ML5, the second stage attached above it, and SRB-3 boosters attached to the first stage.

After functional checks, the payload fairing containing the spacecraft is attached. Final preparations lead into launch countdown.

18.4 Integrated-system testing

18.4.1 Cryogenic test (F-0): Test Flight No. 1 and Flight No. 3

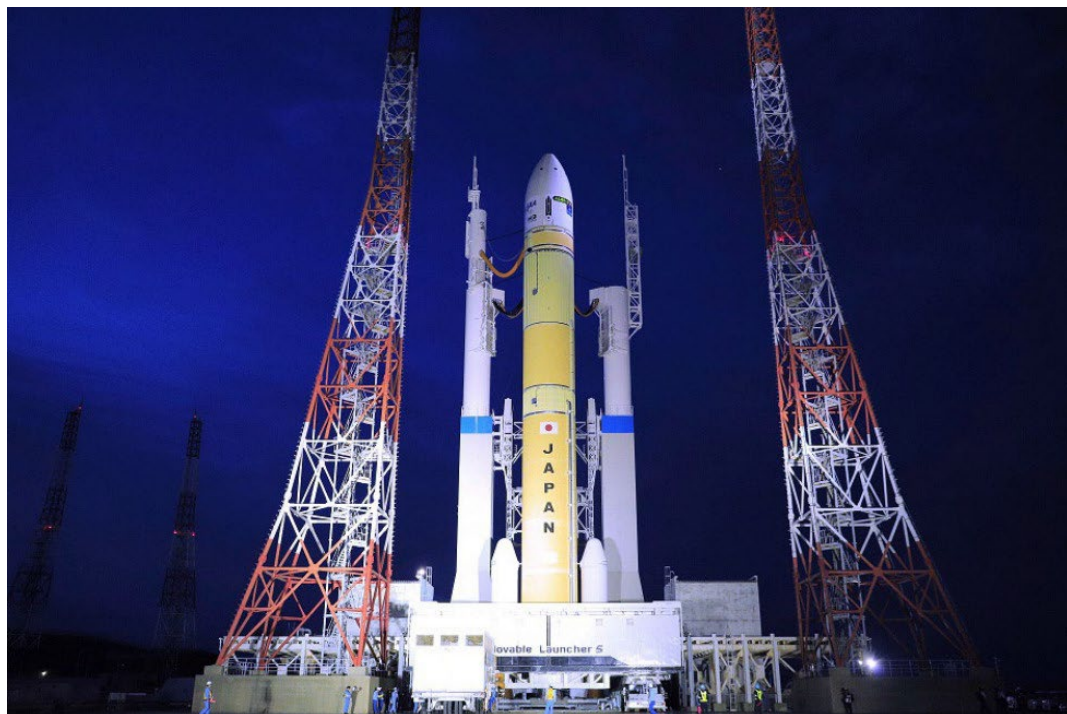
Test Flight No. 1 underwent F-0 on 17–18 March 2021. The vehicle was positioned at the launch pad, filled with liquid hydrogen and liquid oxygen, and taken through a countdown rehearsal up to immediately before engine ignition.

The test verified interfaces among the vehicle, ground facilities and safety-management systems, as well as procedures and operability. It used a fairing from an earlier separation test and test LE-9 engines because the flight engines were not yet complete. Pyrotechnic devices for booster and fairing separation were not installed.

Flight No. 3 underwent F-0 on 29–30 May 2024, as stated in this section of the source, to verify improvements based on the test flights. Data were acquired successfully.



Test Flight No. 1 cryogenic test.



Flight No. 3 cryogenic test.

18.4.2 Special inspections: Test Flight No. 1

Following F-0, special inspections acquired technical data with all stages assembled. Tests included electromagnetic compatibility (EMC), full-vehicle vibration, full-vehicle attitude control, and disconnecting umbilicals—the pipes and cables linking vehicle and launch facilities.

18.4.3 First-stage captive firing test (CFT)

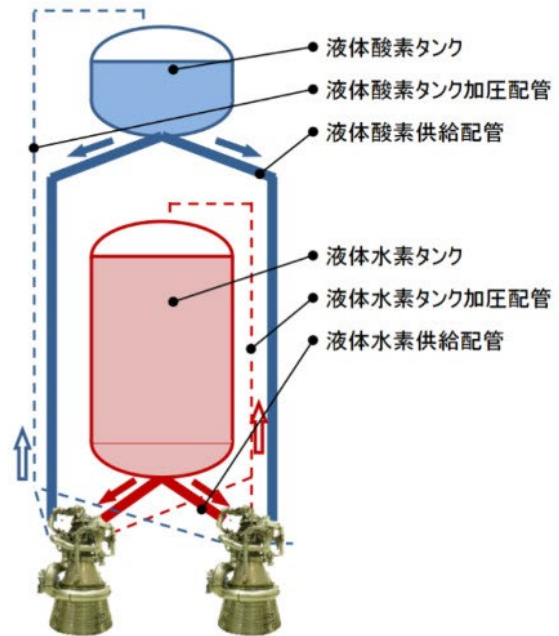
CFT is the final major test before launch. It verifies the first-stage propulsion system: propellant tanks, LE-9 engines, connecting piping and valves.

The actual flight LE-9 engines are installed and fired briefly on the pad. Flight second-stage and fairing hardware are also installed. SRB-3 boosters are omitted because they are not tested in CFT and would increase the consequences of an accident.

As in the March 2021 cryogenic test, the vehicle and facilities perform the sequence leading to launch, confirming functions, procedures and changes arising from F-0. Three second-stage CFTs had already been conducted at Tashiro in July–August 2020.

■ 第1段推進系：飛行中の主な機能

- 加圧された各タンクからエンジンに推進薬（水素/酸素）を供給
- エンジン燃焼により推力を発生
 - ・ ターボポンプにより高圧に推進薬を加圧し、燃焼室に供給
- 推進薬の消費によりタンク圧が下がるためエンジンから取り出されたガスによりタンクを加圧
- 推力方向制御機能により、飛行方向を変更
 - ・ 推力方向制御機能：アクチュエータによりエンジンを首振り



First-stage propulsion functions verified in CFT. English key: liquid-oxygen tank; oxygen-tank pressurization line; oxygen supply line; liquid-hydrogen tank; hydrogen-tank pressurization line; hydrogen supply line. Pressurized tanks feed propellant to the engines; turbopumps raise pressure for injection into the chamber; combustion generates thrust; engine-derived gas maintains tank pressure as propellant is consumed; actuators gimbal the engines for thrust-vector control.

CFT sequence:

- Move the vehicle and launcher from the VAB to the pad; connect electrical cables and propellant loading lines.
- Load liquid propellants and verify vehicle functions under cryogenic conditions.
- Monitor vehicle status through radio links with tracking-station antennas.
- Perform a countdown rehearsal.

Test Flight No. 1's CFT took place on 6–8 November 2022. At 16:30 JST on 7 November, the LE-9 engines ignited for a 25-second firing. Required data were obtained, and detailed analysis confirmed that the test objectives were met.

The test identified numerous improvements, including:

4. First-stage LOX pressurization-line joint leakage after testing. Tank contraction during loading deformed piping and loaded the joint, deforming its seal. During return to ambient temperature, piping recovered its shape, opening a gap through the damaged seal. Affected piping was replaced with redesigned joints, and procedures such as bolt tightening were revised.
5. LOX tank upper-dome relief-valve vibration exceeding its specified environment. Gas flow during pressurization and venting excited the dome and its lightweight relief valve. Environmental requirements

were reassessed, and component vibration testing demonstrated valve tolerance under expected flight conditions.

6. Excessive vibration at second-stage equipment mounting locations. Acoustic energy from the flame-duct outlet excited these areas beyond specified limits. Requirements were revised and tolerance assessed. The inertial sensor, requiring detailed evaluation, underwent additional qualification testing.



Test Flight No. 1 first-stage CFT.

18.4.4 Flight No. 6: H3-30 first-stage CFT

H3-30 also underwent CFT. Earlier H3 configurations used two LE-9 engines; H3-30 requires three engines to generate coordinated thrust without excessive variation. The test verified the entire first-stage propulsion system with all three engines operating together.

Without SRB-3, the vehicle must remain restrained during LE-9 startup. Verification of the hold-down system and its interaction with the ground facilities was therefore important.

On 24 July 2025, three engines fired for the planned 25 seconds. Vehicle and ground-system data were obtained successfully, but during the latter part of the firing, hydrogen- and oxygen-tank pressures did not rise fully to their commanded levels. Correct tank pressure is essential for stable propellant delivery, making this an important issue to resolve before flight.

Countermeasures increased pressurization-gas flow and revised the tank-pressure control plan. A second CFT was conducted on 15 March 2026, as stated in this section of the source, with a longer 50-second burn to obtain additional pressure data.

The test completed as planned. Hydrogen- and oxygen-tank pressurization was controlled normally, with results consistent with predictions. The measures addressing the first test's insufficient pressurization were validated, substantially completing functional and performance verification for H3-30 flight.

- 初回CFTで確認された1段水素/酸素タンク圧昇圧不足事象に対して、取得したデータの詳細評価を踏まえ、下記2点の変更を行った。再CFTでエンジン燃焼時のデータを取得し、その妥当性を検証する。

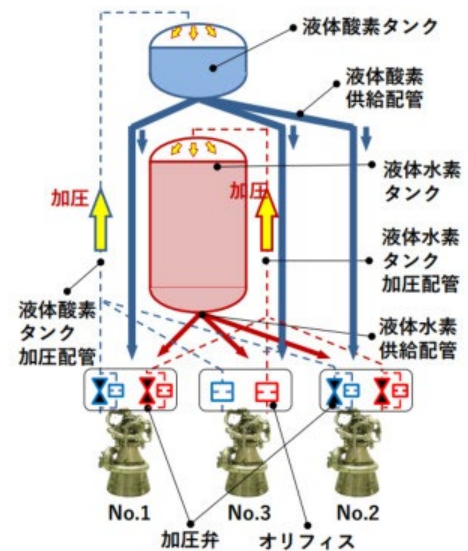
① タンク加圧ガス流量の増加

水素/酸素タンク共に加圧弁の無いNo.3エンジン系統^(※)のオリフィス穴径を変更することで、タンクに供給する加圧ガス流量を増加させる。

② タンク圧制御計画の変更

水素タンクについては22/24形態と共通化していた制御圧力の値を30形態専用に適正化することで、タンク圧制御に必要な加圧ガス流量を低減させる（酸素タンクは対策①のみで対応できるため、本対策は実施しない）。

(※) No.3エンジン系統に加圧弁は無く、常に小流量の加圧ガスを供給する。No.1,No.2エンジン系統は加圧弁があり、加圧ガスを大流量（加圧弁開）/小流量（加圧弁閉）に切り替えて加圧ガスを供給する。



Measures following the first CFT and objectives of the second. Increase pressurization-gas flow by changing the orifice on engine No. 3's hydrogen/oxygen pressurization system, which has no pressurization valve. Engines Nos. 1 and 2 have pressurization valves; No. 3 does not. For the hydrogen tank, optimize the pressure-control setpoint for H3-30 to increase the flow needed for control. The oxygen tank can be addressed by the flow increase alone. Obtain additional data through a longer firing.



Second H3-30 captive firing test.

19 Launch of Test Flight No. 1

19.1 Mission overview

Test Flight No. 1 was intended to provide the final integrated-system flight test while delivering JAXA's DAICHI-3 (ALOS-3) to its designated orbit.

It used H3-22S: two LE-9 engines, two SRB-3 boosters and a short fairing. The two-engine cluster allowed phased verification building on H-IIB experience. Booster count and fairing were selected for ALOS-3's dimensions, mass and target orbit.

19.2 Flight sequence

The following describes the planned sequence, not the outcome of the failed launch.

The vehicle would launch from Launch Pad 2 at Tanegashima, roll to orient its pitch plane to an azimuth of 90.8°, and follow the planned path over the Pacific. SRB-3 separation was planned at approximately T+1 minute 56 seconds, fairing separation at T+3:34, first-stage main-engine cutoff at T+4:58 and stage separation at T+5:05.

The second-stage engine would begin its first burn at T+5:17 and stop at T+16:36. ALOS-3 would separate at T+16:57 into a sun-synchronous subrecurrent orbit approximately 675 km high, inclined 98.1° to the equator.

The second stage would then be deorbited for controlled disposal over the Indian Ocean, avoiding residual orbital debris. H-IIA lacked this capability; H-IIB demonstrated it on Flights Nos. 2–9. H3 incorporated controlled disposal from the outset, reflecting global concern about debris. It will be used on missions meeting conditions such as sufficient performance margin.

Event	Elapsed time	Seconds	Altitude, km	Inertial speed, km/s
Liftoff	00:00:00	0	0	0
SRB-3 separation	00:01:56	116	43	1.5
Fairing separation	00:03:34	214	121	2.1
Main-engine cutoff (MECO)	00:04:58	298	258	3.6
First/second-stage separation	00:05:05	305	275	3.6
Second-stage first thrust buildup (SEL1)	00:05:17	317	302	3.5
Second-stage first cutoff (SECO1)	00:16:36	996	675	7.5
ALOS-3 separation	00:16:57	1,017	675	7.5
Second-stage deorbit thrust buildup (SEL2)	01:47:13	6,433	681	7.5
Second-stage deorbit cutoff (SECO2)	01:47:30	6,450	680	7.1

19.3 Visual design

H3's colors and decals reflect the concept of simple sophistication. The fairing carries JAXA and H3 logos; the JAXA logo is expected to eventually be removed as operations transfer to MHI.

A mission-logo area is also reserved on the fairing. H-IIA/B placed mission logos on the interstage, but H3's changed shape makes that difficult. A black arrow motif on the fairing conveys energetic flight toward space. Its black color is the natural color of the fairing's composite material.

The first stage carries Japan's flag and the word “JAPAN.” Earlier vehicles used “NIPPON”; the more widely recognized English name reflects H3's ambition to provide a global service.



Test Flight No. 1 exterior design and fairing markings: JAXA logo; H3 logo; mission-logo area; black composite arrow motif; Japanese flag and “JAPAN” marking.

20 Test Flight No. 1 failure: investigation and corrective actions

Test Flight No. 1 was initially scheduled for 17 February 2023. A malfunction of the first-stage vehicle controller during pre-liftoff monitoring caused the launch to be aborted. Following investigation and corrective actions, it launched on 7 March, but the second-stage engine failed to ignite. A flight termination command was sent after it was determined that the designated orbit could not be reached.

20.1 Launch abort on 17 February: cause and measures

20.1.1 Overview

Preparations proceeded normally. At approximately T-10 minutes, the final GO/NO GO decision was made and the automatic countdown began. The sequence reached LE-9 start, and the engines ignited and built up normally. Flight Lock In (FLI) was established when automatic checks confirmed the launch conditions: engine thrust equivalent to 90% and normal equipment operating states.

During monitoring between FLI and liftoff, the first-stage controller detected an anomalous signal. Flight-control software stopped the remaining sequence. The SRB-3 ignition command, scheduled immediately after FLI, was withheld by the fail-safe design. The boosters did not ignite. The vehicle, ALOS-3 and ground facilities were undamaged.

20.1.2 Investigation and corrective measures

The investigation used fault tree analysis (FTA), component and flight-hardware tests at Tanegashima, and expert software/circuit reviews.

The inferred cause was a transient electrical-potential change when ground communications and power lines were disconnected immediately after FLI, causing V-CON1 to malfunction.

Electrical disconnection occurs before physical umbilical separation at liftoff. Previously, the communications and power lines were disconnected together. The procedure was changed to introduce a time offset between disconnections, suppressing the transient. Verification tests confirmed effectiveness. Other systems potentially affected by ground-interface transients were checked and modified where necessary.

Final verification data were obtained after rollout to the pad on L-1, the day before launch.

20.2 Launch failure on 7 March: cause and measures

20.2.1 Overview

Test Flight No. 1 launched from Tanegashima at 10:37:55 JST on 7 March 2023. Flight proceeded normally through booster separation, fairing separation, first-stage cutoff and stage separation.

Second-stage thrust buildup, or Second Engine Lock In (SELI), was not confirmed. At T+835 seconds—13 minutes 55 seconds—at 10:51:50 JST, a flight termination command destroyed the vehicle after it was determined that the mission could not be achieved. Analysis indicated that fragments fell within the preplanned first-stage impact area.

JAXA established a special task force on 7 March, headed by President Hiroshi Yamakawa, and an investigation team led by Vice President Yasuhiro Funo. Working with MHI's investigation team, they collected and analyzed data, identified possible causes, developed corrective actions and applied lessons to related systems.

Electrical specialists from JAXA's Space Technology Directorate I, Human Spaceflight Technology Directorate and Research and Development Directorate participated, alongside external experts and retired personnel. Investigation and corrective-action progress was reported to MEXT's Investigation and Safety Subcommittee and made publicly available through online video broadcasts.

Flight results and reference prediction shown in the source figure

Event	Actual elapsed seconds	Reference prediction, seconds
Liftoff	0	0
SRB-3 separation	116	116
Fairing separation	212	211
MECO	296	296
Stage separation	304	303
Second-stage first thrust buildup	Not achieved	316
Flight termination	835	—

20.2.2 Ignition sequence and observed events

The normal second-stage ignition sequence is:

7. V-CON2A/2B detects first/second-stage separation.
8. V-CON2 sends Second Engine Ignition (SEIG) to PSC2.
9. PSC2 forwards SEIG to the Engine Control Box (ECB).
10. ECB commands the Pneumatic Package (PNP).
11. PNP actuates engine valves and the igniter exciter/spark plug.

Equipment	Function
V-CON2	Vehicle flight-control computer. Uses position, velocity and attitude to command engine operation, gas jets and engine gimbaling through subsystem controllers. Two identical units, A and B, provide redundancy.
PSC2	Second-stage propulsion controller. Executes tank-pressure, engine and gas-jet control commands. Contains redundant A and B circuits in one enclosure.
ECB	Determines valve-opening/closing timing during engine startup and shutdown.
PNP	Controls helium supply for valve actuation and operation of the igniter exciter/spark plug.
Exciter	Generates sparks to ignite the engine.

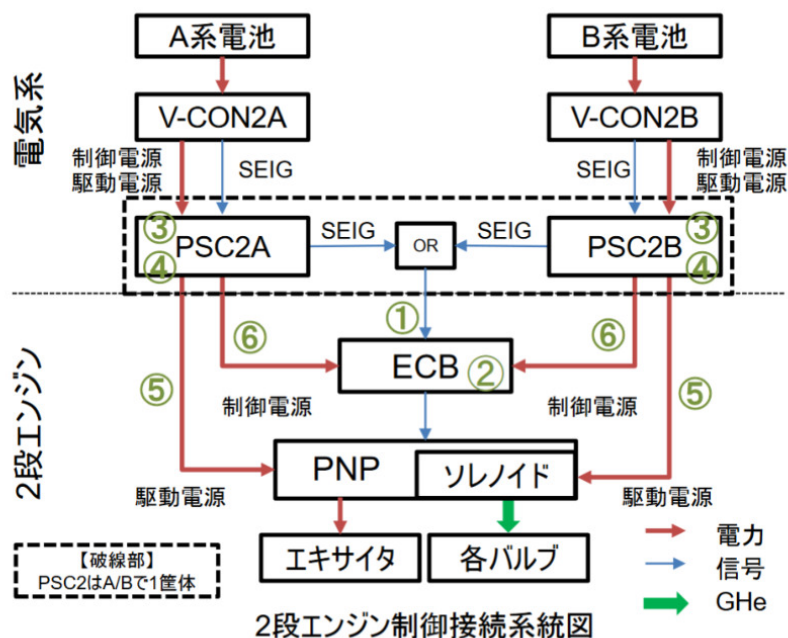
Observed events were:

- Valve control through ECB/PNP was normal up to SEIG.
- PSC2 transmitted SEIG to ECB, and ECB confirmed receipt; operation remained normal to this point.
- Immediately afterward, built-in tests in both PSC2 channels detected abnormal engine-drive voltage/current and disconnected power to downstream equipment.
- The system switched redundancy from A to B.
- PNP drive voltage fell in both channels, while ECB control voltage remained normal.
- Subsequent valve operation was not observed, and the engine did not ignite.

- 2023年3月7日10時37分55秒(日本標準時)に、H3ロケット試験機1号機を打ち上げた。
- 第2段エンジンが着火しなかったことにより、所定の軌道に投入できる見込みがないことから10時51分50秒にロケットに指令破壊信号を送出し、打上げに失敗した。
- ロケットは第1段・第2段分離まで、計画どおり飛行した。



Test Flight No. 1 trajectory and observed sequence. The vehicle flew normally through stage separation; second-stage thrust buildup was not detected; flight was terminated at T+835 s.



Ignition-system events. English key: vehicle controller (V-CON2); propulsion controller (PSC2); engine control box (ECB); pneumatic package (PNP); redundant channels A/B; SEIG command; drive voltage/current fault; power cutoff. The numbered events correspond to SEIG transmission/receipt, fault detection, power interruption and PNP voltage drop, with ECB control voltage remaining normal.

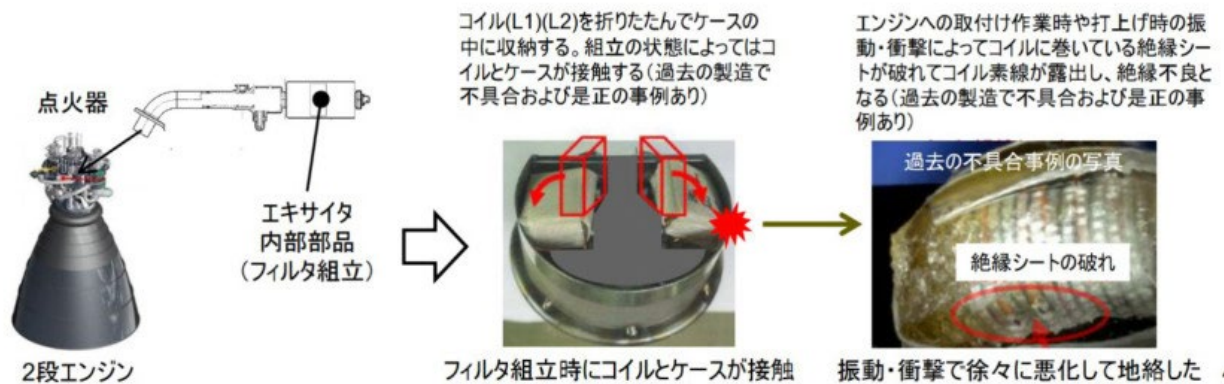
20.2.3 Investigation results

FTA began with the event “second-stage engine did not ignite,” tracing causal relationships and testing possible scenarios. Verification and reproduction tests narrowed the possibilities to three scenarios.

Scenario 1: a minor exciter short became a complete short after SEIG

Manufacturing and handling could leave small clearances between electrical parts, making short circuits or ground faults possible. Launch vibration or stage-separation shock could reduce clearances or abrade insulation, exposing conductors and causing an initially high-resistance fault.

Some PNP devices, including the exciter and solenoids, are first energized at SEIG. If the exciter was already faulted, switching on could produce excessive current instead of ignition. PSC2A would detect the anomaly and disconnect. PSC2B would then power the same exciter, encounter the same short and also disconnect.



Failure scenario based on a ground-fault incident previously recorded during manufacturing: insufficient internal clearance or insulation damage can lead to a fault in the exciter.

Scenario 2: energizing the exciter created an overcurrent condition

In this scenario, the exciter was initially intact but failed and shorted when energized. An internal transformer raises the approximately 30 V input to the voltage needed for sparks. Investigators found that a transistor could have been exposed to voltage above its rating.

Prelaunch tests also operate this circuit. Repeated overvoltage could accumulate damage until the transistor failed during flight. The design was shared with H-IIA/B; 182 transistors of the same type had flown, leading to an assumption that the technology was mature.

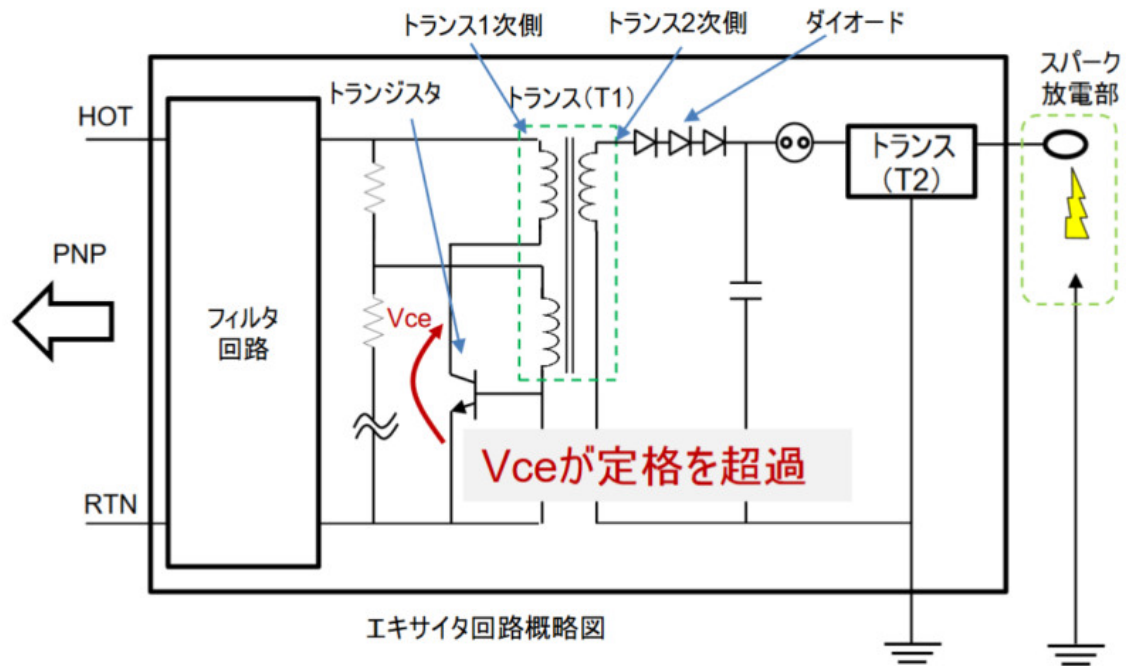
The overvoltage was identified for the first time in this investigation. Measurement equipment available during H-IIA development could not detect it. Tests also showed that transistors can tolerate repeated rating exceedance and that susceptibility to failure varies.

Scenario 3: overcurrent within PSC2A propagated to channel B

PSC2 A and B are generally independent. Batteries supply approximately 60 V, stepped down to approximately 30 V. A constant-voltage diode was provided to protect against excess voltage if the converter malfunctioned.

The diode could function at voltages slightly above 30 V but would itself fail short if a complete converter failure applied the full 60 V. This extreme condition had not been anticipated.

Although A and B were nominally separate, their negative return lines were physically connected. Under certain conditions, current from the shorted diode could pass through the return line and affect channel B.



PSC2 electrical failure scenario: converter failure; full battery voltage; protective-diode short; current propagation through the shared return path to the redundant channel.

20.2.4 Corrective actions

None of the three scenarios could be ruled out. Measures were therefore applied to all three.

Measures for Scenario 1

An insulation sheet could have shifted during filter assembly, leaving the coil touching its case. Installation or launch vibration could progressively abrade enamel insulation, making a ground fault possible. Similar manufacturing incidents had been recorded.

Insulation and inspection were strengthened:

- Reduce the coil insulation sheet from 1.5 wraps to one wrap, improving assembly clearance while maintaining insulation performance.
- Add RTV rubber to coil leads and heat-shrink tubing to cables between connector boards, improving abrasion protection.
- Add X-ray CT inspection to confirm there is no contact between coil and filter case.

These measures were incorporated from H-IIA Flight No. 47, whose successful launch on 7 September 2023 provided flight confirmation.

No.	故障部位		シナリオ	対策
15	エキサイタ	フィルタ組立	組立時にコイル・ケース間の絶縁シートがずれ、コイルとケースが接触。組立作業中の取り扱いや打上げ時の振動・衝撃でコイル表面のエナメル被覆が剥がれ、コイル素線とケースが接触し、SEIG時に地絡。	<絶縁強化および検査強化> ・エキサイタ製造検査にX線CT検査を追加し、コイルとフィルタケースの接触がないことを確認する。 ・コイルの絶縁シートの巻き数を1.5巻→1巻に変更して厚みを減らし、コイルをケースに収納し易くしてクリアランスを改善する。 ・コイルリード線にRTVゴム、コネクタ基板間ケーブルに熱収縮チューブを追加し摩耗に対する保護を強化する。



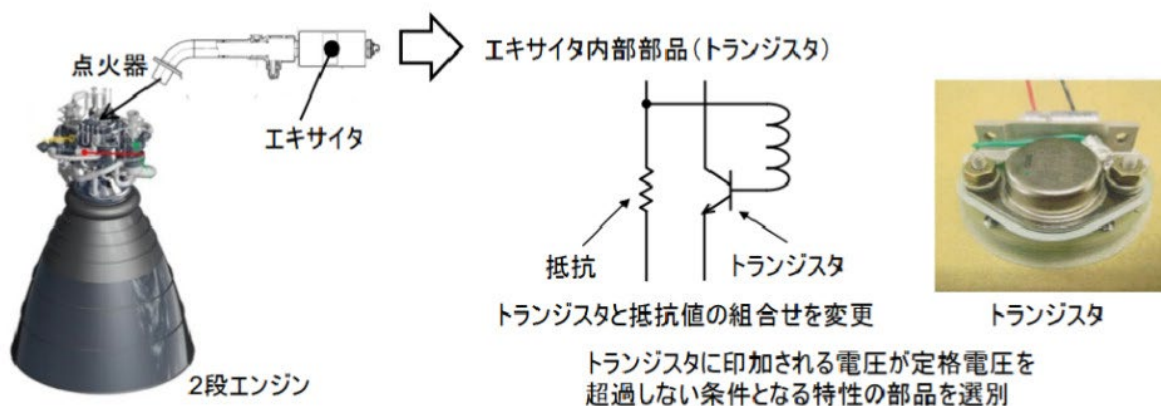
Scenario No. 15: measures for exciter filter assembly faults. Improved insulation, clearance and X-ray CT inspection.

Measures for Scenario 2

The transistor overvoltage was a very brief pulse during circuit oscillation when the exciter generated a spark. Slowing the oscillation slightly reduced peak voltage. This lowered the spark rate, but remained within a range previously demonstrated not to affect ignition.

These measures were also applied from H-IIA Flight No. 47 and confirmed through its successful launch.

No.	故障部位		シナリオ	対策
18	エキサイタ	トランジスタ	エキサイタ点火時にトランジスタの電圧 (V_{ce}) が定格を超過し損傷、短絡による過電流が発生。	<部品選別> トランジスタに印加される電圧が定格内となるトランジスタと抵抗値の組合せとする(従来からエキサイタのスパークレートの調整のために組合せを選別していたが、目的を変えて電圧調整のための選別とするもの)。 電圧を下げる選別によってスパークレートが低下するが、過去の開発試験でエンジン着火に影響がないことを確認した範囲であるため許容可能と評価した。

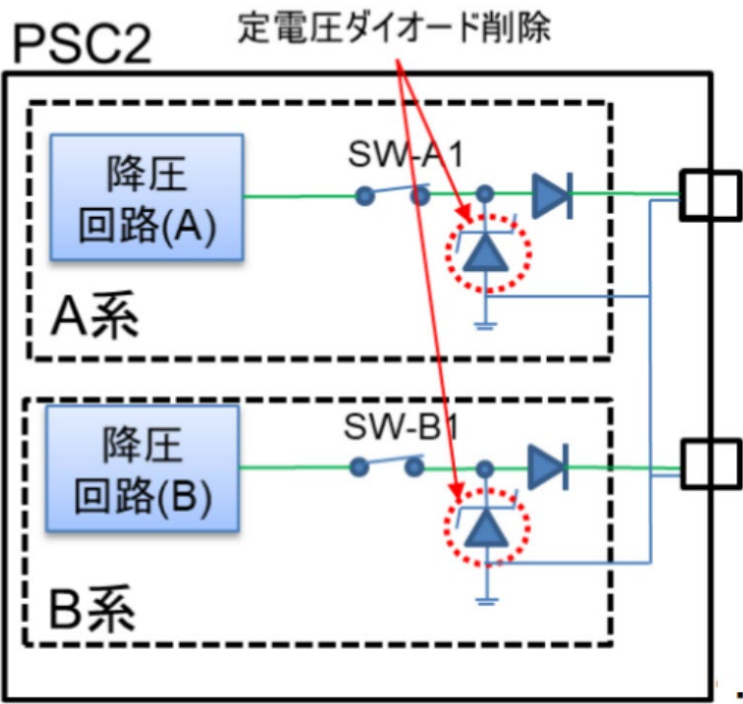


Scenario No. 18: measures for exciter transistor overvoltage. Reduce the transient voltage during spark generation while retaining an acceptable spark rate.

Measures for Scenario 3

The constant-voltage diode was judged inadequate as protection because it failed when full battery voltage was applied. Tests showed that downstream devices—ECB, PNP and exciter—could withstand battery voltage. The diode was therefore unnecessary and could itself cause failure, so it was removed.

This circuit was newly developed for H3 and absent from H-IIA. Flight demonstration of the measure was therefore performed on Test Flight No. 2.



PSC2 corrective circuit. Remove constant-voltage diodes in both channels. English key: step-down circuit A/B; channels A/B; switches SW-A1/SW-B1; constant-voltage diode removed.

20.2.5 Underlying factors and corrective actions

Repeated “why” analysis was applied to the three scenarios to identify underlying contributors.

Scenario	Underlying factor	Action
Exciter short/ground fault	Reliance on heritage since H-II, and an assumption that narrow component clearances would not change after manufacture, meant manufacturing, inspection and design measures were not introduced.	Examine heritage equipment for latent failure potential associated with manufacturing difficulty and similar issues.
Exciter overcurrent after energization	When applying electrical equipment whose designs and processes predated H-II, the program did not check sufficiently for gaps in component-compatibility assessment.	Reassess component suitability evaluations for long-used heritage equipment and identify deficiencies.
PSC2A fault propagating to B	Tolerance of protective devices under possible system-fault conditions had not been fully verified.	Check the design and verification of normally inactive protective devices whose failure could cause mission-critical faults.

H3 had drawn on approximately 20 years of reliability improvements following H-IIA Flight No. 6's failure. However, these devices had designs and manufacturing processes established more than 30 years earlier, before H-II, with almost 200 flights through H-IIA/B. The program had not sufficiently revisited their latent fault potential.

“Why” analysis systematically identifies accident and malfunction factors. After organizing the sequence of events, it repeatedly asks why each problem or error occurred, pursuing causes until effective recurrence-prevention measures can be established.

20.2.6 Reliability improvements

The investigation also identified broader improvements: richer H3 measurement data, improved redundant-channel switching logic, and stronger electrical-system development for future launch vehicles.

More comprehensive measurement data

Limited telemetry made fault isolation time-consuming and contributed to the inability to identify one definitive cause. Downlink capacity is constrained, so earlier data sets primarily covered essential flight-safety and postflight evaluation items, generally common to every mission.

The revised approach optimizes data within transmission limits to make overcurrent and overvoltage events easier to identify and isolate during prelaunch verification and flight.

Improvement	Rate	Application	Purpose
PSC2 engine-drive voltage	8 Hz to 32 Hz	Permanent	Isolate overvoltage-related events with finer timing resolution
V-CON2A/2B power-bus current	64 Hz to 256 Hz	Permanent	Isolate overcurrent events and affected locations
High-speed current monitor between PSC2 and PNP	512 Hz	Test Flight No. 2 only	Verify corrective-action effectiveness
Engine-drive voltage between PSC2 and PNP, covering exciter and selected valves	512 Hz	Test Flight No. 2 only	Verify corrective-action effectiveness

Special Test Flight No. 2 measurements were conditional on prelaunch verification with the modified flight hardware that other measurement items would not be affected. The earlier 8 Hz sampling interval was longer than the interval between SEIG transmission and receipt, leaving too little information on events within that period; 32 Hz improved detail.

Improved redundancy-switching logic

Vehicle-side V-CON2 and PSC2 circuits and the power supply to engine-side ECB/PNP have redundant A/B channels. A healthy channel should continue supplying power if the other fails. However, a shared downstream fault such as Scenarios 1 or 2 can cause both supplies to switch off.

The program reconsidered failure modes, channel-isolation methods and detection conditions, distinguishing overvoltage from overcurrent.

Overvoltage indicates a possible PSC2 power-source fault. Excess voltage could damage healthy downstream equipment, so the affected channel is disconnected promptly, as before.

Overcurrent at normal voltage suggests a load-side fault rather than a power-source fault. Instead of disconnecting immediately, the delay was extended from 8 milliseconds to 1 second. This improves the possibility of continuing the mission through a brief fault that clears quickly, increasing robustness.

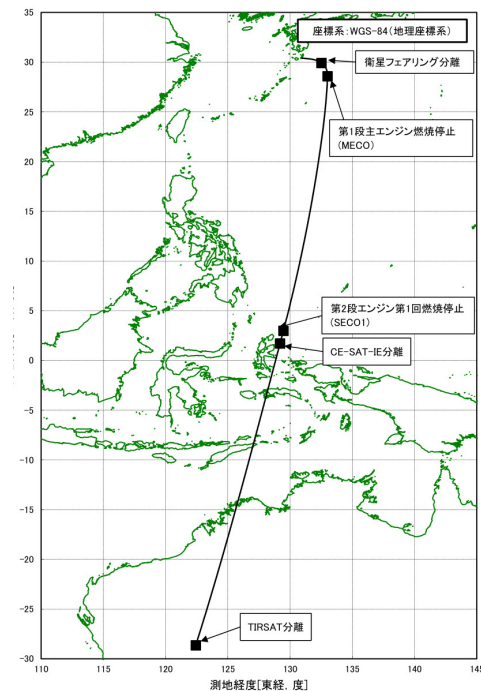
21 Launch of Test Flight No. 2

21.1 Mission overview

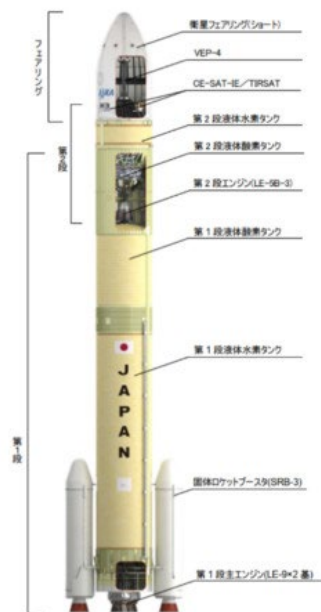
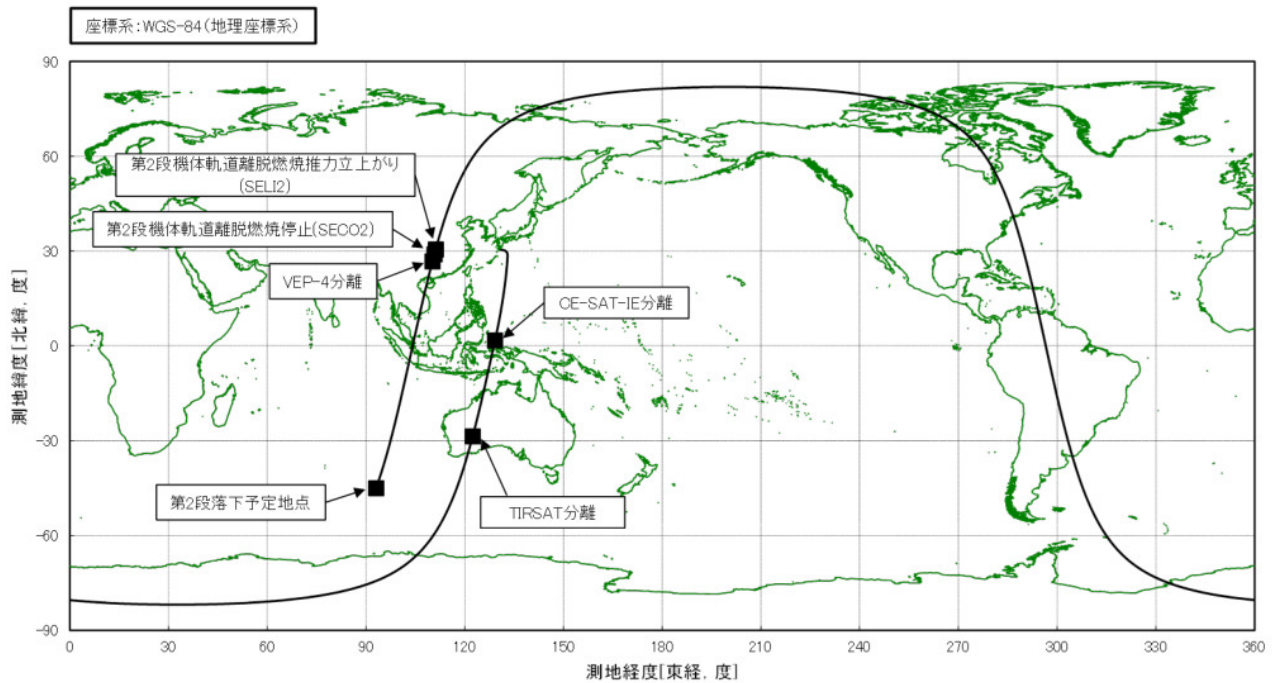
Test Flight No. 2 launched on 17 February 2024. Its primary mission was to achieve H3 orbital insertion and validate the development program through flight demonstration.

Following Test Flight No. 1's failure and loss of ALOS-3, the plan considered the consequences of another payload loss while prioritizing an early demonstration to minimize effects on subsequent launches.

The vehicle retained the H3-22S configuration to maximize reuse of Test Flight No. 1 mission analysis. It carried VEP-4, with mass properties equivalent to ALOS-3, and followed the same trajectory through the first second-stage cutoff. Two secondary satellites, CE-SAT-IE and TIRSAT, were also offered orbital launch opportunities.



Flight path from liftoff to CE-SAT-IE/TIRSAT separation. English key: WGS-84 geographic coordinates; geodetic latitude, degrees north; geodetic longitude, degrees east; fairing separation; main-engine cutoff (MECO); second-stage first cutoff (SECO1); CE-SAT-IE separation; TIRSAT separation.



TF2(H3-22S)

TF1とTF2の比較

	TF1	TF2
機体形態	H3-22S	H3-22S
ペイロード	ALOS-3	VEP-4 小型副衛星(CE-SAT-IE) 小型副衛星(TIRSAT)
衛星フェアリング	ショートフェアリング	ショートフェアリング
第1段 (LE-9)	Type1 × 2基	Type1 × 1基 Type1A × 1基
固体ロケットブースタ (SRB-3)	2本搭載	2本搭載
第2段 (LE-5B-3)	1基搭載	1基搭載 TF1不具合対応として エキサイタ改修品搭載
搭載機器	H3-22S対応の 各機器を搭載	H3-22S対応の 各機器を搭載 TF1不具合対策として PSC2改修品搭載

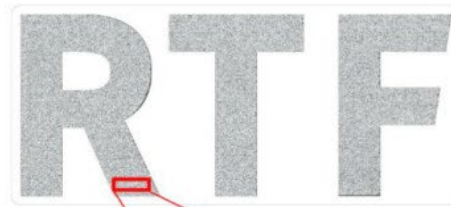
赤字はTF2での主要変更点

Flight path through controlled second-stage disposal and comparison with Test Flight No. 1. Map key: WGS-84; latitude/longitude; SELI2; SECO2; VEP-4 separation; CE-SAT-IE separation; TIRSAT separation; planned second-stage impact point. Vehicle comparison: both use H3-22S, short fairing, two boosters and one LE-5B-3. TF1 carries ALOS-3 and two Type 1 LE-9 engines; TF2 carries VEP-4, CE-SAT-IE and TIRSAT, one Type 1 and one Type 1A LE-9, modified exciter and modified PSC2 as failure countermeasures. Red text in the source marks TF2 changes.

21.2 Vehicle configuration

H3-22S: two LE-9 engines, two SRB-3 boosters and a short fairing.

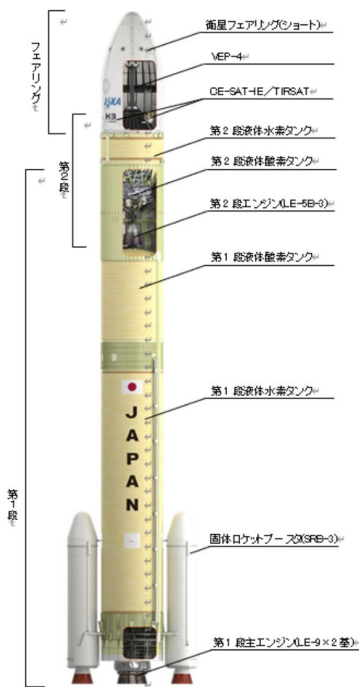
Support messages collected through a dedicated website and X, formerly Twitter, were printed within the letters “RTF” and attached to the fairing. Of 2,931 submissions received from 10–30 November, 2,845 were included after excluding inappropriate content. RTF means “Return To Flight.”



ロケット無事に打ち上がってください希望を乗せて夢へと翼つ
力は失敗の積み重ねです。次こそ成功します。がんばれ H3 ロ
ち上ることを祈っています。 打ち上げ関係者の皆様、頑張
てくれます。頑張ってください！ H3 は日々の研究、積み上げ
も落胆しました。それは私が宇宙開発に興味が出てきたからで
とを期待しています。打上げ再挑戦！広い宇宙へ飛び立つ H3
型ロケットで宇宙への新たな扉を開け祝成功！必ず宇宙へ届く
援します。頑張れ H3！頑張れおとん！皆様の強い信念が必ず
はずっと皆さんと H3 ロケットを応援しています。頑張れ～！

Return To Flight fairing graphic containing public support messages.

21.3 Principal specifications



Test Flight No. 2 vehicle layout. From top to bottom: short payload fairing and payload; payload attachment structure; second-stage hydrogen/oxygen tanks and LE-5B-3 engine; interstage; first-stage hydrogen/oxygen tanks; two SRB-3 boosters; two LE-9 engines. The corresponding dimensions and masses are given below.

Item	Specification			
Vehicle	H3 Test Flight No. 2			
Overall length	Approximately 57 m			
Fully loaded mass	Approximately 422 t, excluding spacecraft mass			
Guidance	Inertial guidance			

Item	First stage (LE-9)	SRB-3, two boosters	Second stage (LE-5B-3)	Short fairing
Length, m	Approx. 37	Approx. 15	Approx. 12	Approx. 10.4
Outer diameter, m	Approx. 5.2	Approx. 2.5	Approx. 5.2	Approx. 5.2

Item	First stage (LE-9)	SRB-3, two boosters	Second stage (LE-5B-3)	Short fairing
Mass, t	Approx. 240	Approx. 152.4, combined	Approx. 28	Approx. 1.8
Maximum propellant mass, t	224.5	134.4, combined	24.6	—
Vacuum thrust, kN*	Approx. 2,942, two engines	Approx. 4,600, two boosters	Approx. 137	—
Burn time, s	Approx. 300	Approx. 110	Approx. 694	—
Propellants	LH2/LOX	Composite solid propellant	LH2/LOX	—
Feed system	Turbopumps	—	Turbopumps	—
Attitude control	Engine gimbaling	—	Engine gimbaling and gas jets	—
Principal electronics	Guidance/control equipment	—	Guidance/control equipment; radio-navigation equipment; telemetry transmitter; flight termination equipment	—

*Solid motor thrust is specified at maximum thrust.

21.4 Flight sequence

The vehicle launched with VEP-4 and two secondary satellites from Tanegashima's Launch Pad 2, oriented its pitch plane to azimuth 90.8° and followed the planned Pacific flight path.

SRB-3 separation was planned at approximately T+1:56, fairing separation at T+3:34, first-stage cutoff at T+4:58 and stage separation at T+5:05. The second-stage first burn began at T+5:17 and ended at T+16:36. CE-SAT-IE separated at T+16:57 into an approximately 675 km-high sun-synchronous subrecurrent orbit, inclined 98.1°. A TIRSAT separation signal followed at T+25:17.

The second stage then performed controlled disposal over the Indian Ocean to prevent orbital debris. After the deorbit burn, VEP-4's separation mechanism was demonstrated. Stopper bolts retained the payload frame on the stage, so VEP-4 reentered with it.

The following table reproduces the source's flight plan, whose stated planned launch time is 09:22:55 JST.

Event	Elapsed time	Seconds	Altitude, km	Inertial speed, km/s
Liftoff	00:00:00	0	0	0.4
SRB-3 separation	00:01:56	116	43	1.5
Fairing separation	00:03:34	214	121	2.1
MECO	00:04:58	298	258	3.6
Stage separation	00:05:05	305	275	3.6
SELI1	00:05:17	317	302	3.5
SECO1	00:16:36	996	675	7.5
CE-SAT-IE separation	00:16:57	1,017	675	7.5
TIRSAT separation	00:25:17	1,517	682	7.5
Deorbit thrust buildup (SELI2)	01:47:13	6,433	679	7.5
Deorbit cutoff (SECO2)	01:47:39	6,459	678	7.1
VEP-4 separation, retained by stopper bolts	01:48:19	6,499	676	7.1

21.5 Payloads

21.5.1 Vehicle Evaluation Payload-4 (VEP-4)

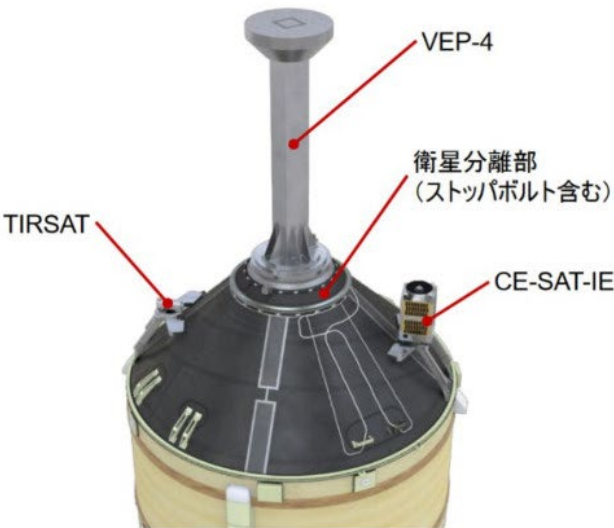
Item	Specification
Name	Vehicle Evaluation Payload-4 (VEP-4)
Upper section	Approximately 1.2 m diameter × 0.4 m
Support section	Approximately 0.4 m × 3.5 m high
Mass	Approximately 2.6 t
Manufacturer	MHI

VEP-4 reproduced ALOS-3's mass properties to verify H3 performance. It also demonstrated the primary payload separation mechanism after the second-stage deorbit burn, while remaining attached by stopper bolts.

サイズ	上部 : φ約1.2m × 約0.4m 支柱部 : □約0.4m × 高さ約3.5m
質量	約2.6トン



VEP-4外観



VEP-4および小型副衛星搭載図
(衛星フェアリング内)

VEP-4 layout: upper mass-simulating section; support column; payload separation interface; retained payload frame.

21.5.2 Small secondary satellites

The rideshare program offers accessible, timely launch and operational opportunities for small satellites built by private companies and universities. It broadens Japan's space activities and contributes to education and workforce development.

Secondary payloads must not affect the primary H3 flight-demonstration mission. JAXA could omit them if they threatened the main launch; both satellites were ultimately placed into their planned orbits.

JAXA previously offered small-satellite rideshares on H-IIA to provide sustained orbital demonstration opportunities and acquire technical knowledge. In December 2019, it signed a basic agreement with Space BD to transfer the H-IIA/H3 small-satellite rideshare business. Initial plans for Test Flight No. 2 were abandoned in May 2022 because of schedule changes and payload-capacity issues.

At the 75th meeting of the Committee on Space Development and Utilization on 24 May 2023, the direction toward a vehicle evaluation payload was confirmed, together with consideration of secondary satellites where they would not affect reliable acquisition of performance data. Analysis showed that one 50 kg-class satellite and one 3U satellite could be accommodated.

A Request for Information conducted from 5–12 June 2023 received four candidate submissions. Relative assessment of technical issues and schedule risk selected CE-SAT-IE in the 50 kg class, and the 3U TIRSAT. The selection was reported to the committee's 76th meeting on 27 June.

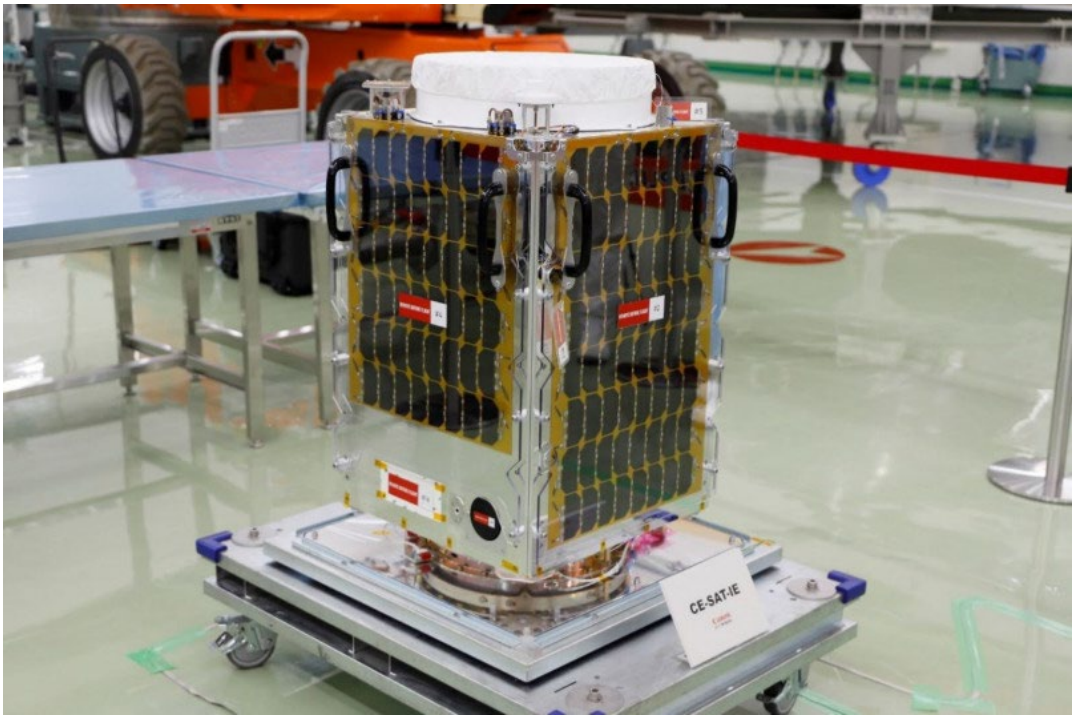
CE-SAT-IE

Pronunciation: “C-E-Sat-One-E.”

Item	Specification
Developer	Canon Electronics Inc.
Dimensions	Approximately 50 × 50 × 80 cm
Mass	Approximately 70 kg
Primary optics	40 cm-aperture reflecting telescope with one EOS R5
Secondary optics	One PowerShot S110

Mission activities:

- Earth imaging at 0.8 m ground resolution using the primary optics, including still images and 8K video.
- Demonstration for remote-sensing business and astronomical imaging.
- Contribution to JAXA geospatial-information and disaster-management activities, and acquisition of images supporting public safety and security.
- Imaging of celestial bodies and space objects with suitable spatial/temporal resolution, exposure, signal-to-noise ratio and coverage.
- Satellite-bus demonstration and in-space verification of a new mission computer, communications equipment, internally developed optics, sensors and actuators.
- Orbit-determination exercises using laser ranging with JAXA's Mt. FUJI equipment. JAXA acquires and evaluates ranging data; orbit determination and related evaluations are performed jointly.
- A deorbit device.



CE-SAT-IE.

TIRSAT

Pronunciation: “T-I-R-Sat.”

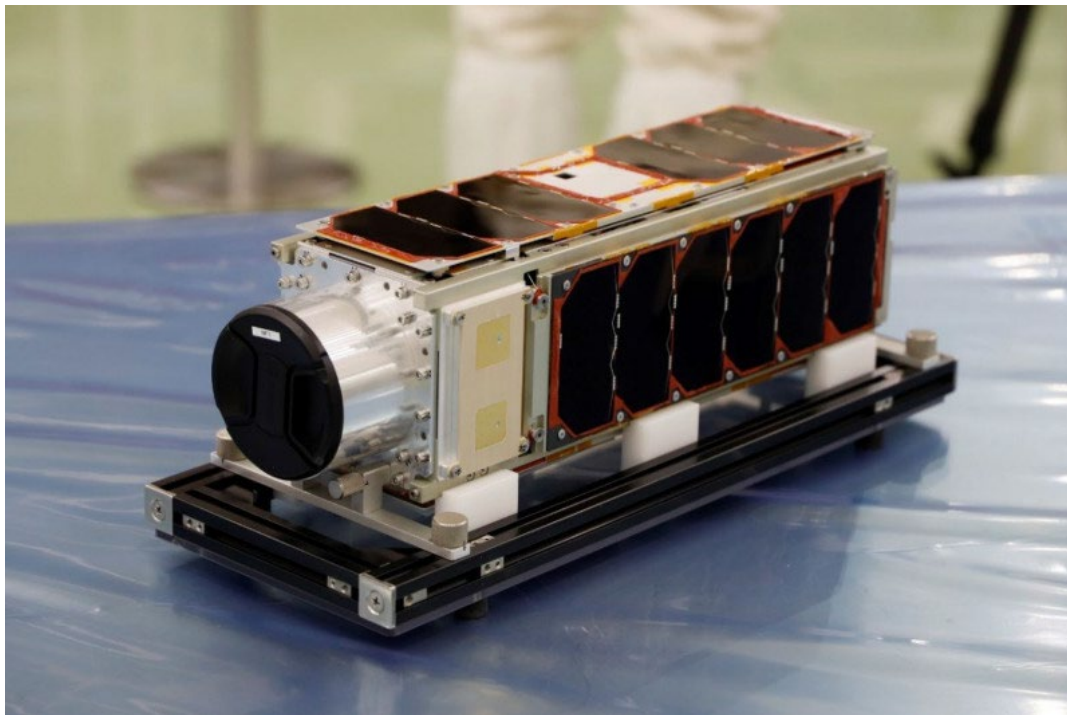
Item	Specification
Developers	Japan Space Systems; Seiren Co., Ltd.; Vision Sensing Co., Ltd.; ArkEdge Space Inc.
Dimensions	Approximately 12 × 12 × 38 cm
Mass	Approximately 5 kg

The COVID-19 pandemic exposed risks of global supply-chain disruption and the need for rapid information collection. Satellite remote sensing can help monitor industrial activity in major production regions. Thermal-infrared sensors detect factory heat sources and can be used to infer operating status.

TIRSAT demonstrates a thermal-infrared camera developed through the Ministry of Economy, Trade and Industry, using an uncooled sensor on a small satellite bus. It tests the effectiveness of this information-gathering method in space for future crises.

The camera observes long-wave infrared wavelengths suited to measuring near-ambient-temperature targets. It supports a high-sensitivity mode at 8–14 μm and a limited-band mode at 10.5–12 μm . Its uncooled sensor can achieve approximately 96 m ground resolution from 600 km altitude, comparable to available thermal-infrared satellite data.

The 3U bus—nominally 10 × 10 × 30 cm—provides power, attitude control and communications. It carries S-band and X-band equipment, using high-speed X-band for image downlink.



TIRSAT.

21.6 Payload separation tests

Test Flight No. 2 demonstrated the PAF operation that Test Flight No. 1 could not perform. Following the deorbit burn, VEP-4 separated from its clamp interface but remained retained by stopper bolts and reentered with the second stage.

The PAF accepts standard clamp bands from Japanese and overseas manufacturers to support varied payloads. Each secondary satellite also had its own deployer.

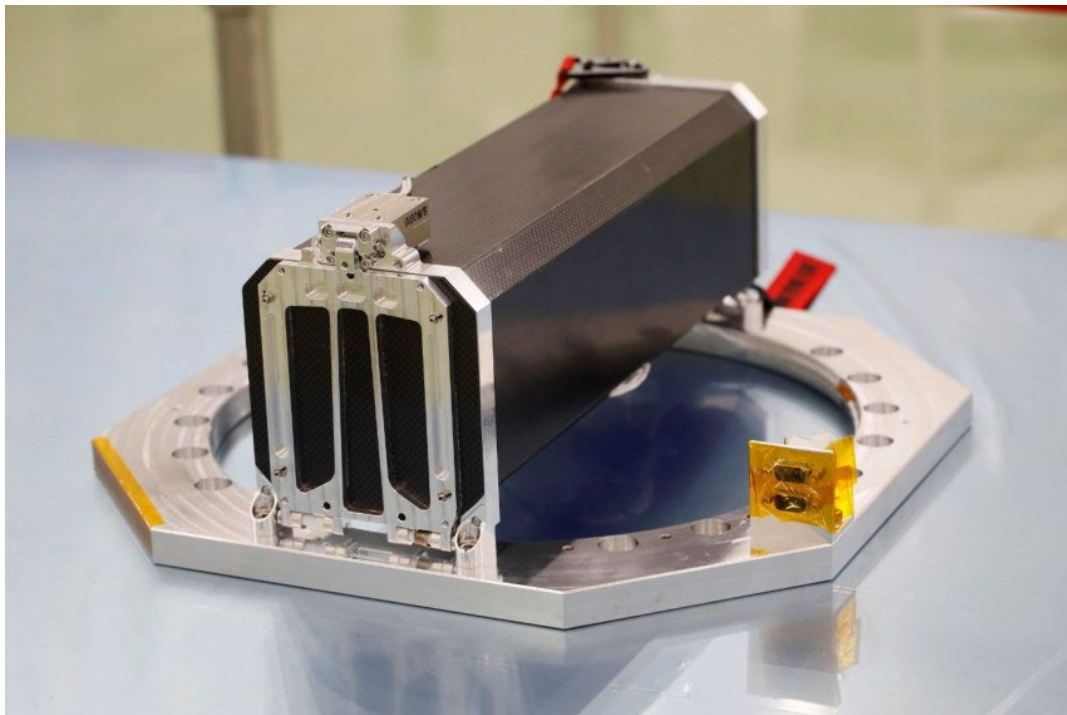
21.6.1 Simple PAF15M

Kawasaki Heavy Industries' nonpyrotechnic separation mechanism enables low-shock deployment. Two approximately 40 cm-diameter circular frames are joined like a sandwich: the upper frame attaches to the satellite and the lower to the vehicle. A band holds them together.

On receipt of a separation command, a pin puller releases the band and springs deploy the satellite. Avoiding pyrotechnics reduces shock compared with conventional PAFs. Simple PAF15M deployed CE-SAT-IE on this mission.

21.6.2 3U pod

Developed by Orbital Engineering, the 3U pod is a box-shaped deployer containing the satellite. Its lid opens without pyrotechnics for low-shock deployment, and its CFRP structure is lightweight. It deployed TIRSAT.



Nonpyrotechnic 3U satellite deployer.

22 Flight results: Nos. 3–5 and 7

22.1 Flight No. 3

- Launch: 1 July 2024, 12:06:42 JST.
- Payload: Advanced Land Observing Satellite-4, DAICHI-4 (ALOS-4).
- Result: Success.
- Mission details: https://fanfun.jaxa.jp/countdown/alos4-h3/files/h3_f3_presskit-a_240619.pdf

Flight No. 3 used the H3-22 configuration demonstrated on Test Flight No. 2. First-stage flight included throttling.

22.2 Flight No. 4

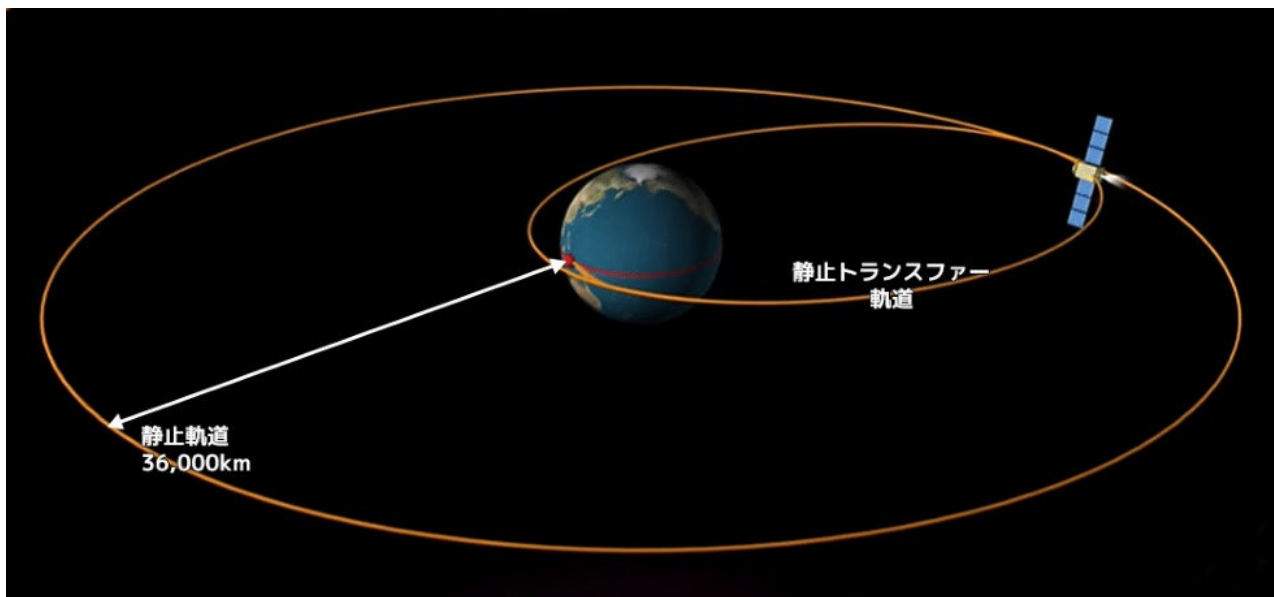
- Launch: 4 November 2024, 15:48:00 JST.
- Payload: X-band defense communications satellite KIRAMEKI-3.
- Result: Success.
- Mission details: https://fanfun.jaxa.jp/countdown/h3f4/files/h3f4_presskit_20241028.pdf

Flight No. 4 placed the Ministry of Defense's KIRAMEKI-3 into H3's first GTO. It used H3-22S, as did Flights Nos. 1–3, and acquired data after payload separation for future long-coast missions.

22.2.1 GTO insertion

Like weather and communications satellites, this payload operates in geostationary orbit, approximately 36,000 km above the equator. Its orbital period matches Earth's rotation, so it appears stationary from the ground and can continuously observe approximately the same region.

The launch vehicle first places the satellite into an elliptical GTO. The satellite then uses its own propellant to reach geostationary orbit.



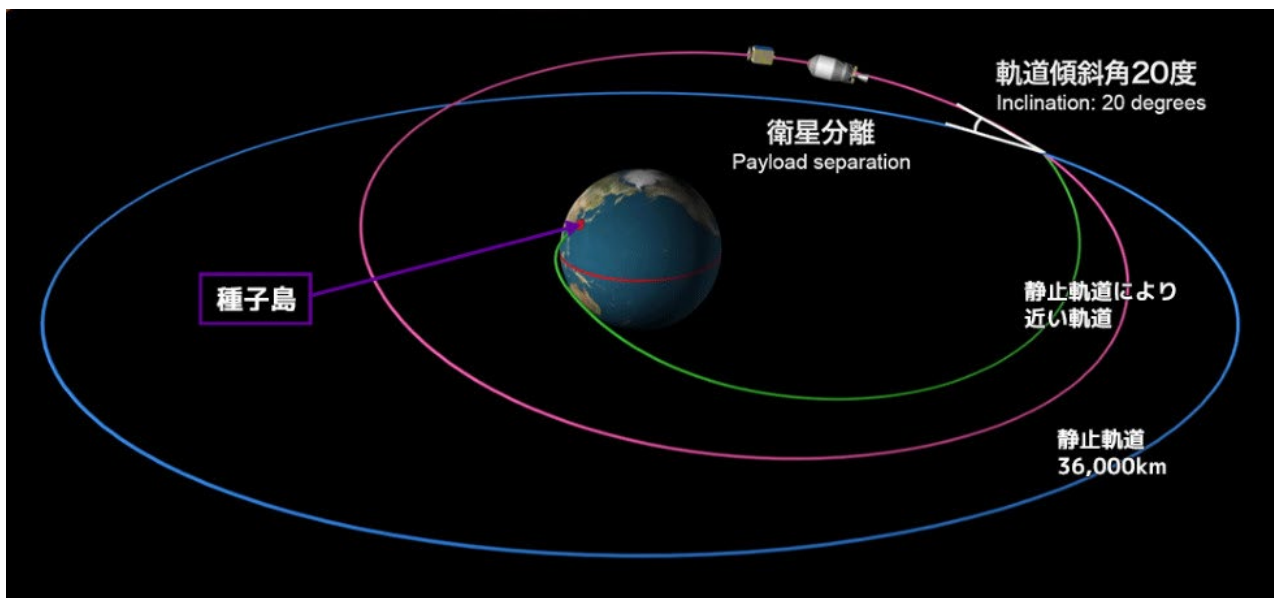
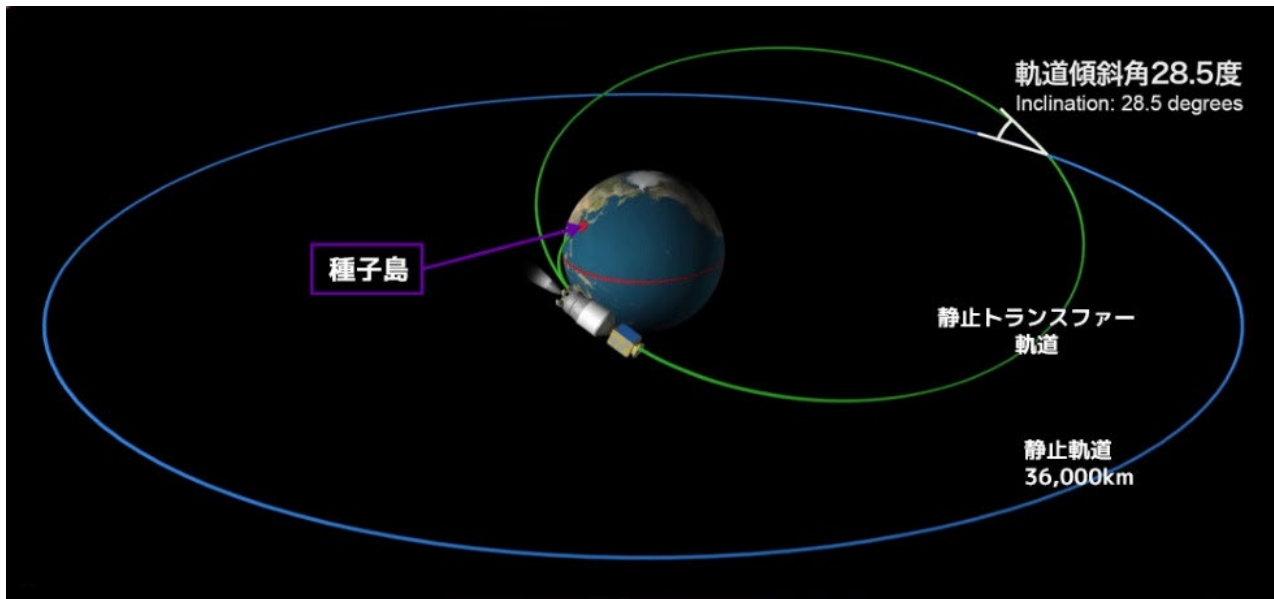
Geostationary orbit and transfer orbit: Earth; equatorial geostationary orbit; elliptical GTO; satellite-powered transfer to geostationary orbit.

22.2.2 Data for future long-coast missions

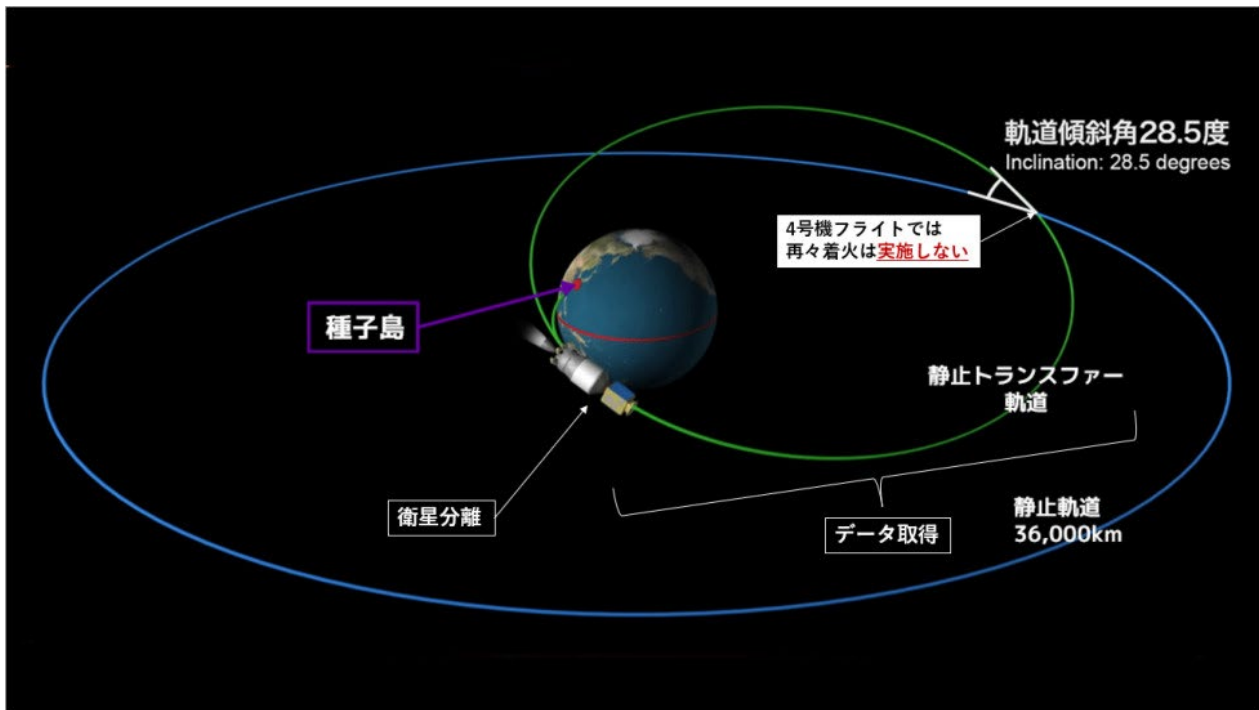
Tanegashima's latitude produces an initial orbital inclination of approximately 28.5°, placing it at a performance disadvantage compared with an equatorial launch site.

A long coast followed by another second-stage ignition can place the satellite closer to geostationary orbit, reducing inclination to approximately 20° and increasing orbit altitude. This reduces the satellite's propellant requirement.

The technology was developed in the H-IIA Upgrade Project. Flight No. 4 collected data for future H3 application. Since nearly all propellant was used to insert KIRAMEKI-3, it did not perform the additional ignition; it acquired coast-phase data beforehand. A high-gain antenna supported the longer communication distance.



Orbit insertion from Tanegashima and with a long-coast mission. English key: initial inclination approximately 28.5°; subsequent burn near apogee; higher transfer orbit; reduced inclination approximately 20°; lower satellite ΔV requirement.



Flight No. 4 data acquisition concept: coast-phase measurements to support future long-coast GTO missions, without an additional engine ignition on this flight.

22.3 Flight No. 5

- Launch: 2 February 2025, 17:30:00 JST.
- Payload: MICHIBIKI No. 6.
- Result: Success.
- Mission details: https://fanfun.jaxa.jp/countdown/h3f5/files/h3f5_presskit_20250131.pdf

Flight No. 5 delivered MICHIBIKI No. 6 to GTO using H3-22S. First-stage throttling was performed as on Flights Nos. 3 and 4. After separation, data were acquired for future long-coast GTO missions, as on Flight No. 4.

22.4 Flight No. 7 mission overview

- Launch: 26 October 2025, 09:00:15 JST.
- Payload: First HTV-X cargo spacecraft, HTV-X1.
- Result: Success.
- Mission details: https://fanfun.jaxa.jp/countdown/htv-x1_h3f7/files/h3f7_presskit.pdf

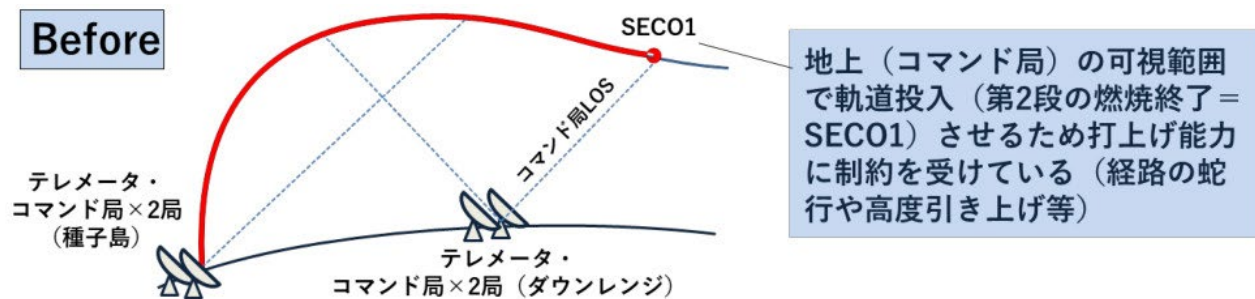
Flight No. 7 placed HTV-X1 into an ISS-bound orbit. It used H3-24W with a dedicated wide fairing and four boosters, marking the first flight of H3-24, the highest-capacity H3 configuration.

It also demonstrated autonomous flight safety and TDRS communications for future missions.

22.4.1 Autonomous flight-safety demonstration

Autonomous flight-safety technology is planned to improve payload capability for HTV-X and GTO missions. During later phases of second-stage flight, an onboard system independently assesses vehicle or trajectory anomalies and can interrupt flight through engine shutdown or vehicle destruction, without a ground command.

Flight No. 7 carried a computer with newly developed autonomous flight-safety software. It collected demonstration data only and was not connected to the flight termination chain.

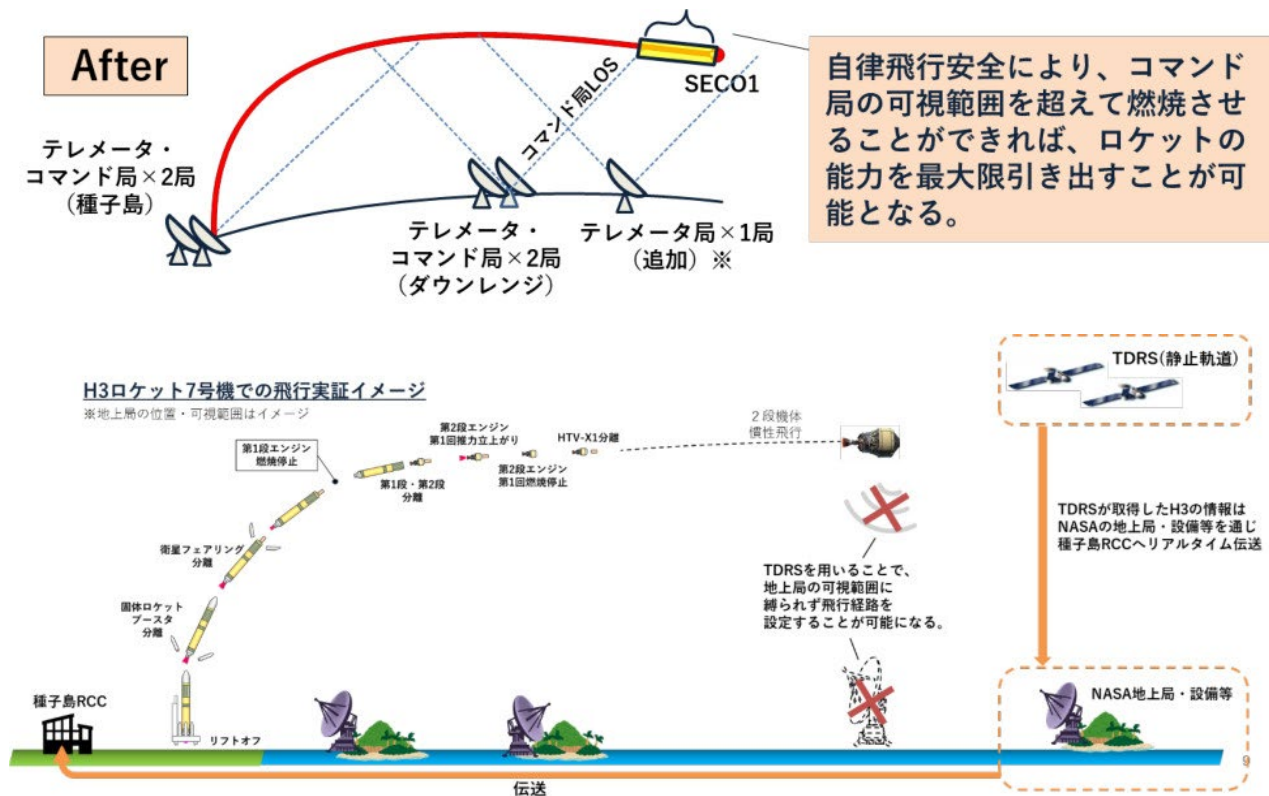


Conventional ground-command flight safety and the autonomous-system concept. Ground monitoring/commands can be replaced in applicable flight phases by onboard evaluation; Flight No. 7 demonstrated the logic without enabling flight termination.

22.4.2 TDRS flight demonstration

Lunar and planetary missions such as Martian Moons eXploration (MMX), the world's first planned sample return from a Martian moon, often traverse regions without ground-station coverage. NASA's Tracking and Data Relay Satellite (TDRS) service is planned to relay vehicle telemetry there.

Flight No. 7 carried a new TDRS-compatible transmitter and antenna on the second stage. After HTV-X1 separation, it demonstrated data acquisition through TDRS over the Atlantic Ocean.



Autonomous flight-safety concept and TDRS communications demonstration. English key: launch vehicle; TDRS relay satellite; ground station; relay service; telemetry transmission; post-HTV-X1-separation demonstration over the Atlantic.

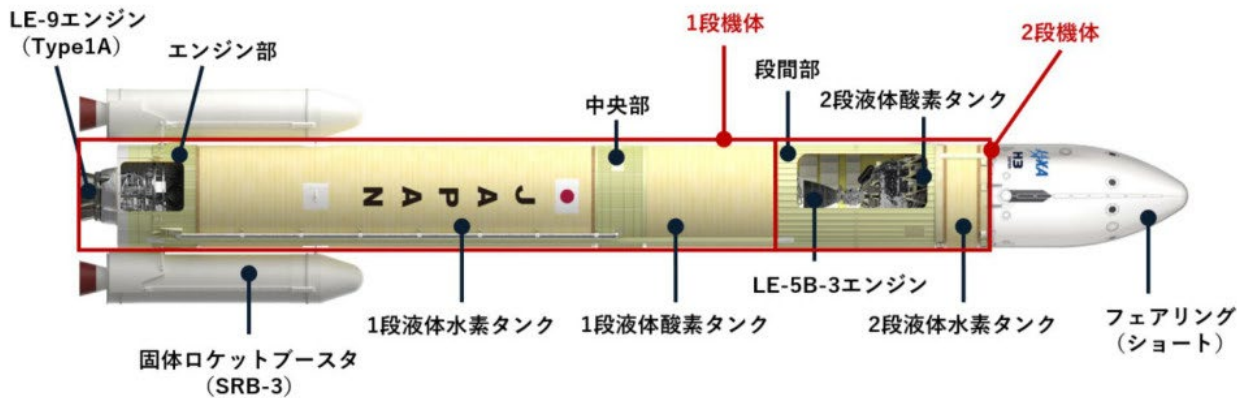
23 Flight No. 8

23.1 Mission overview

Flight No. 8 was planned to place MICHIBIKI No. 5 of QZSS into its designated quasi-zenith orbit. It used H3-22S, as did Test Flight No. 1 through Flight No. 5, with first-stage throttling as on Flights Nos. 3–5.

23.2 Vehicle configuration

H3-22S: two LE-9 engines, two SRB-3 boosters and a short fairing.



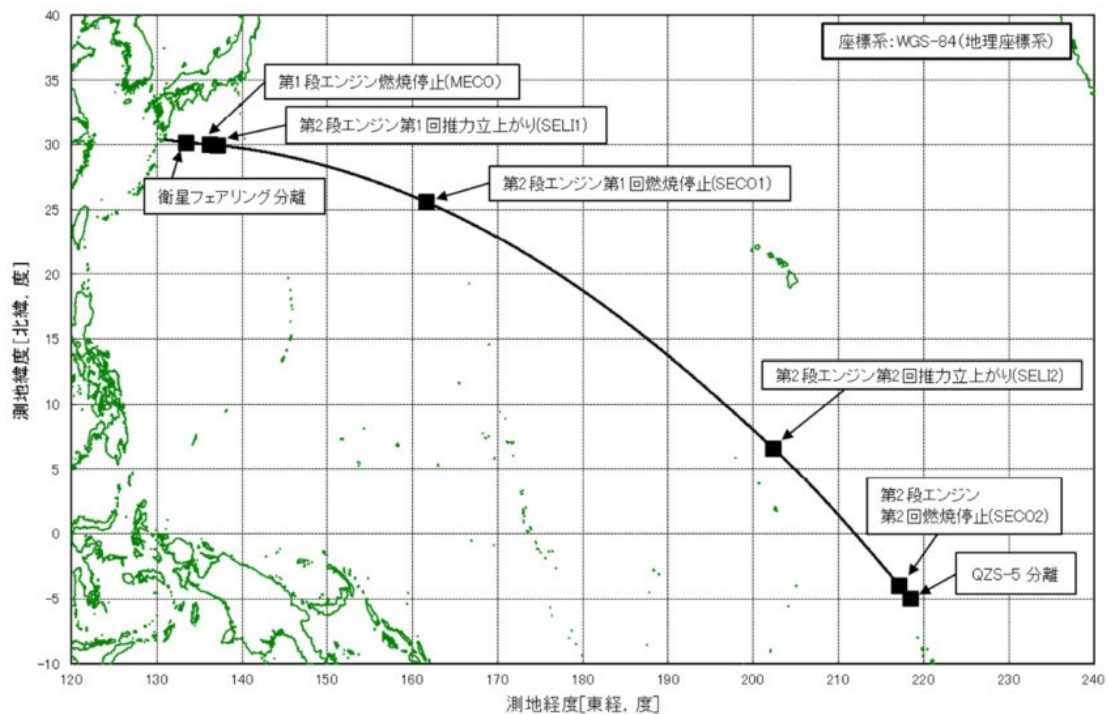
Flight No. 8 H3-22S layout: short fairing; payload attachment structure; second-stage tanks/LE-5B-3; interstage; first-stage tanks; two SRB-3 boosters; two LE-9 engines.

23.3 Planned flight sequence

Flight No. 8 launched MICHIBIKI No. 5 from Tanegashima's Launch Pad 2. The planned sequence oriented the pitch plane to azimuth 90° and followed a Pacific flight path.

Planned event	Approximate time after launch
SRB-3 separation	00:01:56
Fairing separation	00:03:45
First-stage cutoff	00:04:59
Stage separation	00:05:07
Second-stage first burn starts	00:05:20
Second-stage first burn ends	00:12:48
Second-stage second burn starts, after coast	00:24:46
Second-stage second burn ends	00:29:05
MICHIBIKI No. 5 separation	00:29:26

The planned separation orbit had a perigee altitude of approximately 370 km, an apogee altitude of approximately 35,586 km and an inclination of 36.0°. After mission completion, the second stage's perigee would be lowered to shorten its orbital lifetime and ensure natural reentry within 25 years, reducing orbital debris.



Planned flight path from liftoff to MICHIBIKI No. 5 separation. English key: liftoff; booster/fairing separation; MECO; stage separation; first second-stage burn; coast; second burn; payload separation.

23.4 Launch failure and investigation



H3 Flight No. 8 lifts off.

23.4.1 Launch abort on 17 December 2025: cause and measures

Flight No. 8 was scheduled to launch MICHIBIKI No. 5 at 11:10:00 JST on 17 December 2025. At T-16.8 seconds, the cooling-water deluge system detected an anomaly and the automatic countdown stopped in an emergency abort. No damage occurred to the vehicle, satellite or ground equipment.

The nitrogen-pressurized water system protects the flame deflector and ducts from hot exhaust and suppresses acoustics through steam generation.

The anomaly was isolated to the flame-deflector cooling circuit. The flame-duct deluge circuit operated normally, but the deflector circuit supplied only approximately 150 m³/min instead of the specified 200 m³/min.

The direct cause was insufficient opening of a manually operated valve controlling nitrogen pressurization of the water tank. Inadequate gas flow produced insufficient tank pressure and cooling-water flow.

The underlying issue arose from a procedural change intended to improve workability. Previously, valve opening was verified by measuring a specified length with a scale. Because the location was narrow and visual checks after work were difficult, Flight No. 8 introduced a verification jig. Although intended to check the same position, it was used at a different position, resulting in a smaller valve opening.

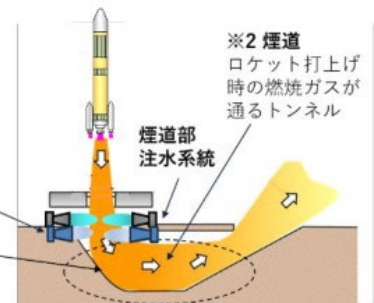
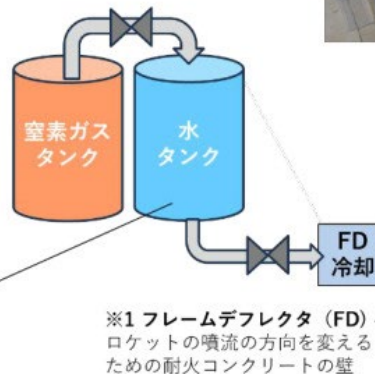
The launch procedure reverted to the proven method, and a verification test confirmed the required water flow. Future changes to procedures or tools would require prior verification, attendance during initial use and proper incorporation into work instructions, strengthening quality assurance.

(参考) 冷却水注水設備とは

窒素ガスで水タンクを加圧して、高温のロケットの噴流に注水する設備。水で冷却することで、フレイムデフレクタ※1や煙道※2を守る。また、発生した水蒸気により、ロケットエンジンが発する大きな音響を吸収し、衛星やロケットを守る。

※冷却水注水設備には、フレイムデフレクタ (FD) 冷却水系統および、これと同等の仕組みを持つ煙道部注水系統がある。

今回は煙道部注水系統の作動は良好、FD冷却水系統の注水量が規定より少なかったのが中止の原因である。



Cooling-water deluge equipment: nitrogen supply; manually adjusted valve; pressurized water tanks; flame-deflector cooling circuit; flame-duct deluge circuit.

23.4.2 Launch failure on 22 December 2025: cause and measures

Launch overview

Flight No. 8 launched at 10:51:30 JST on 22 December 2025. Following booster separation, fairing separation, first-stage cutoff and stage separation, the second-stage engine performed its first burn and reached the designated Earth parking orbit.

The second burn failed to build up normally and stopped prematurely, preventing insertion of MICHIBIKI No. 5 into its intended orbit. A payload separation signal was transmitted, but the separation switches did not confirm separation. The satellite is believed to have fallen within the first-stage impact area, as did the first stage. No damage reports were identified.

JAXA established the H3 Flight No. 8 Special Task Force that day, headed by President Hiroshi Yamakawa. An investigation team performed technical analysis, evaluation and corrective-action studies in cooperation with MHI. JAXA satellite specialists also participated, supported by the Cabinet Office's QZSS Strategy Office and Mitsubishi Electric.

What happened in flight?

The anomaly initially observed was the failure of the second-stage engine to start up normally for its second burn. However, detailed analysis of flight data indicated that the initiating event had occurred earlier in the mission.

At the time of fairing separation, part of the payload attachment structure is believed to have experienced structural failure. This failure is also considered likely to have damaged the second-stage hydrogen-tank pressurization line. Flight data showed decreases in temperature at the payload separation interface and within the Payload Support Structure (PSS), occurring almost simultaneously with a drop in hydrogen-tank pressure.

Although the pressurization valve was open, tank pressure did not recover, and the expected temperature change associated with hydrogen-gas flow into the tank was not observed. Based on these findings, investigators understand that the structural failure damaged the pressurization line and prevented adequate pressurization of the hydrogen tank for second-stage operation.

The first-stage engine continued to generate thrust following the structural failure, maintaining a compressive load between the launch vehicle and the satellite. As a result, the satellite is believed to have initially remained with the vehicle under this compressive load. After first-stage engine cutoff, this compressive force disappeared, and the satellite is inferred to have unintentionally separated from the launch vehicle around the time of stage separation. By second-stage engine ignition, the second stage is believed to have been flying without the satellite.

Several observations support this interpretation. Imagery obtained during stage separation showed an object resembling the satellite. In addition, although the thrust produced during the first burn of the second-stage engine was approximately 20 to 35 percent lower than planned, the burn duration increased by only about 5 percent, and the second stage reached its parking orbit. This behavior is consistent with a reduction in second-stage mass resulting from the loss of the payload.

Acceleration and temperature measurements at the payload separation interface remained nominal until first-stage engine cutoff, then showed behavior consistent with severed electrical wiring at the time of stage separation. Part of the payload attachment structure is believed to have remained attached to the departing satellite. Wiring associated with the separation switches is also considered to have been severed at that time, explaining why commanded separation was not confirmed.

The investigation therefore infers that a payload attachment abnormality began immediately after fairing separation and that the satellite subsequently separated unintentionally from the launch vehicle during flight. The anomaly observed during the second burn was therefore a consequence of this failure sequence, rather than the initiating event itself.

■ H3ロケット8号機の飛行状況（現時点の推定）



Flight No. 8 sequence inferred from telemetry and imagery: fairing separation and attachment-structure damage; pressurization-line damage; payload retained temporarily under first-stage thrust; payload detachment around stage separation; second stage flies without payload; abnormal second burn.

Why did it happen?

FTA systematically examined a broad range of possible causes against flight data, test results and manufacturing records. Together with reproduction tests, it indicated that pre-existing delamination introduced during PSS manufacturing was extremely likely to have propagated under fairing-separation shock and caused structural destruction.

PSS is the structure supporting the satellite on the vehicle. Its panels combine CFRP skins with aluminum honeycomb cores. The satellite is attached to the second stage through PSS and the payload attach fitting (PAF). A PSS bulkhead separates the payload compartment from the space forward of the second-stage hydrogen tank.

PSS manufacturing produces four circumferential panel segments and then joins them. Narrow CFRP splice strips bridge adjacent panel joints on both inner and outer surfaces, bonded under heat at atmospheric pressure. H-IIA PSS panels used mechanical fasteners; H3 changed to adhesive joining for lower mass and cost.

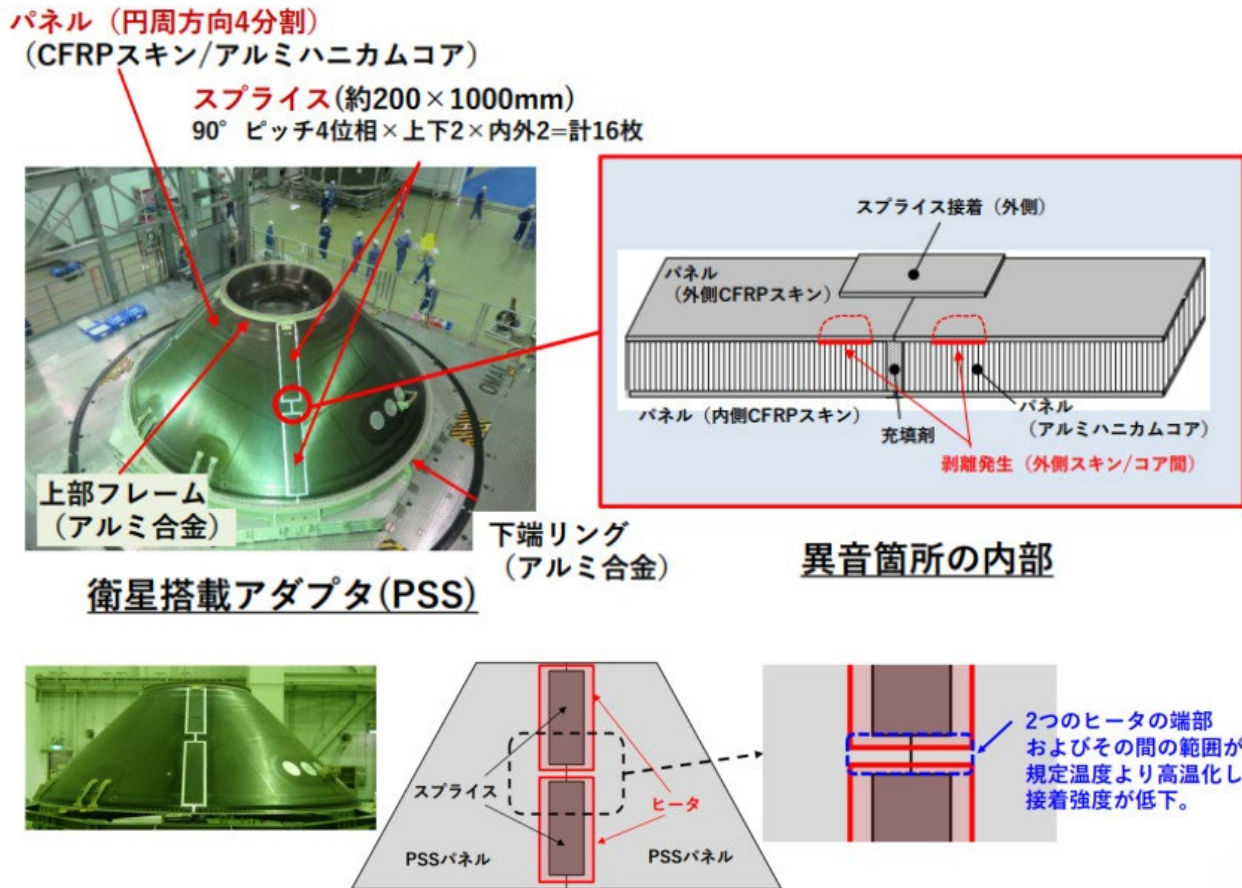
Tap tests on several manufactured PSS units detected abnormal sounds near splice locations. Detailed inspection of representative cut sections found delamination between CFRP skins and honeycomb cores beyond the expected extent.

The condition of Flight No. 8's actual affected area was not recorded. However, similar delamination in PSS units made by the same process strongly suggests that its PSS also entered flight with internal delamination. Fairing-separation shock could then cause local skin buckling or rapid delamination growth, leading to destruction.

The principal manufacturing contributor is considered to be unexpectedly high temperature during part of the splice-bonding process established in development. A reproduction test using flight-type panels created similar delamination during the first heating step, used for drying before bonding.

Between two heaters, panel temperature rose more than anticipated, lowering skin-to-core bond strength. Air inside the honeycomb expanded during heating, producing a peel load on the skin. The weakened bond could not resist that load, allowing skin separation.

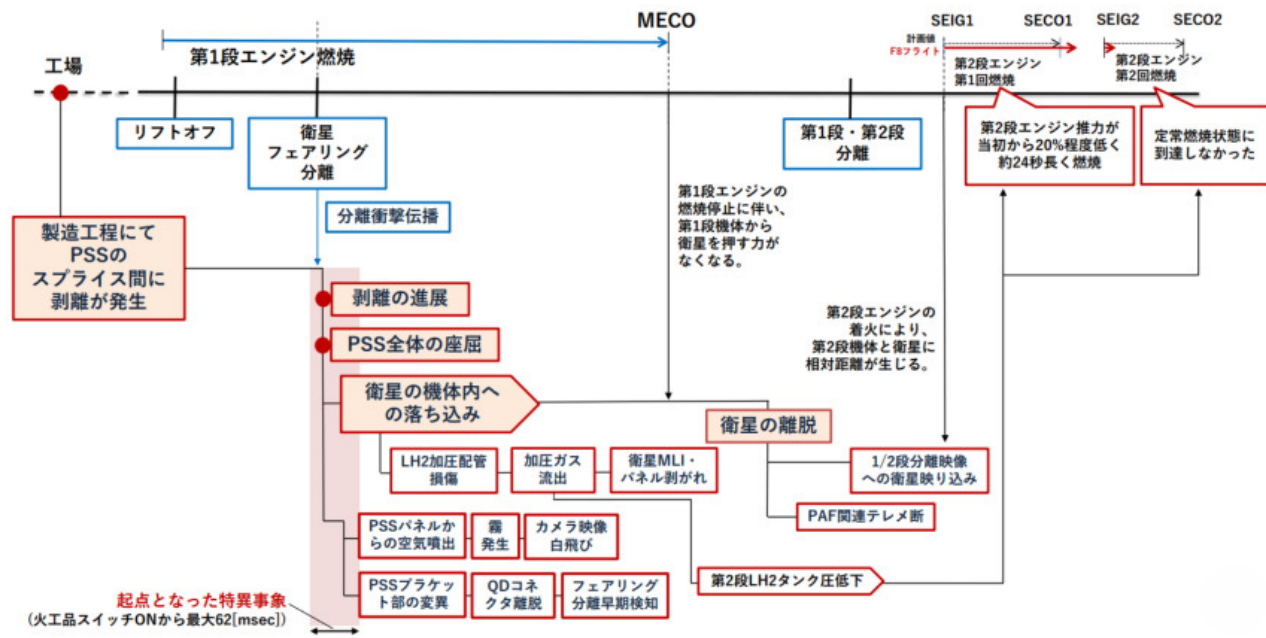
Investigators distinguish delamination between splices from delamination beneath splices. Delamination between splices is considered extremely likely to have been the principal failure cause. Delamination beneath splices was smaller and localized around honeycomb joints, with a low assessed propagation risk.



PSS construction and inferred delamination mechanism. English key: CFRP skin; aluminum honeycomb core; splice strip; splice bonding; two heaters; excessive panel temperature; reduced adhesive strength; expanding trapped air; peeling load; skin/core delamination. Delamination between splice regions is distinguished from smaller delamination beneath the splices.

FTA retained some factors that could not be completely excluded as contributors acting in combination with internal PSS delamination. Flight hardware cannot readily be recovered, so events must be reconstructed from limited telemetry, imagery, ground tests and manufacturing records.

There is no direct flight evidence that these other factors occurred, and they were not judged independent principal causes or causes equivalent in significance to PSS delamination. Additional data on subsequent missions were planned to deepen understanding and confirm the investigation's assessment.



Timeline of observed and inferred events: fairing separation; PSS damage; hydrogen pressurization loss; first-stage cutoff; stage separation and inferred payload detachment; second-stage first burn; coast; abnormal second burn.

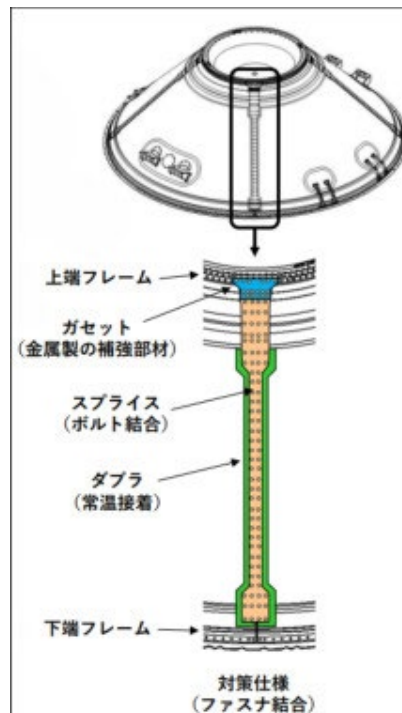
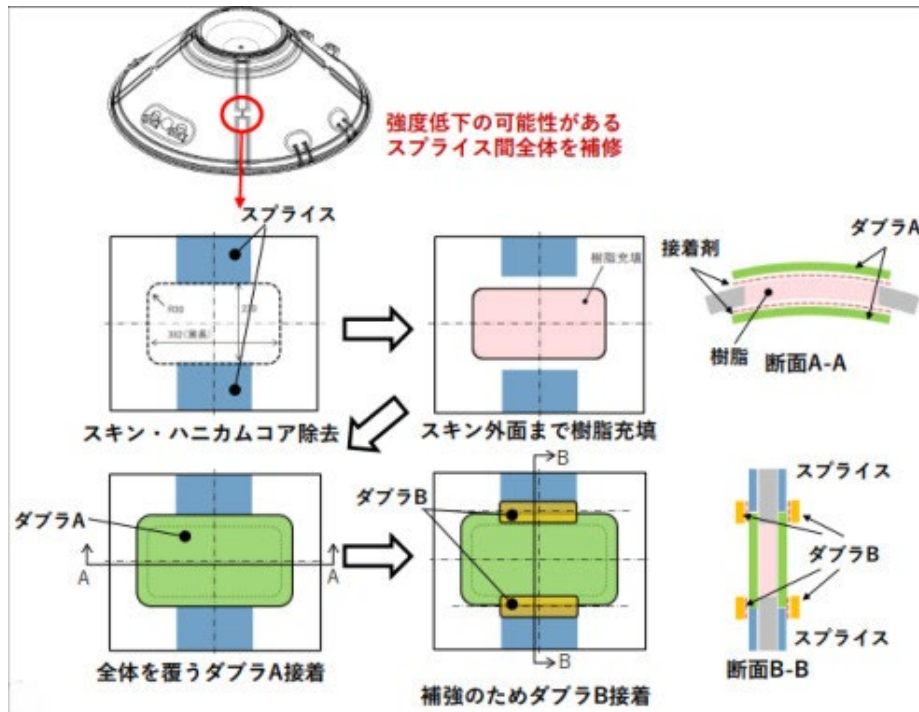
Corrective actions for return to flight

The basic policy was to eliminate the initial PSS delamination considered extremely likely to be the principal cause. Two approaches were examined: repairing already-manufactured PSS units, and replacing bonded panel joints with mechanical fasteners.

The repair approach removes skin from all delaminated and weakened-bond areas, fills from the honeycomb core to the outer skin surface with resin, and bonds CFRP over the entire repaired region. The aim is to leave no initial delamination and restore required strength.

The fastener approach replaces adhesive panel joining with bolts and nuts. This removes thermal loading from splice bonding and structurally prevents the associated delamination initiation and growth.

Both were assessed for technical feasibility, mass impact and implementation on flight hardware, and both can provide required strength when properly applied.



Corrective approaches: left—repair of delaminated/weakly bonded areas by skin removal, resin filling and CFRP covering; right—mechanical fastener joining of PSS panels, eliminating splice-bonding heat exposure.

Flight No. 6, the first return-to-flight mission and H3-30 test, used the repair approach, with required inspections, repairs and strength verification. Its smaller payload mass and lower center of gravity limited PSS loads to about one-third of standard design conditions, allowing adequate strength margin.

For important missions immediately following Flight No. 6, the program selected fastener joining to remove the delamination risk at its origin. Repairs require complete detection and treatment of initial delamination; fasteners eliminate the relevant thermal process and its associated initiation/growth factors, supporting stable quality.

Flight No. 6's added PSS strain measurements supported the validity of the repair and of the strength-design methodology shared with the fastened design. They also confirmed that the potential combined-factor events not completely excluded before that launch did not occur.

A permanent approach has not yet been selected. The program will choose the optimum method, including repair or fastener joining, according to each mission, payload and manufacturing implications.

Applying the findings to related hardware

Fairings use a splice-bonding process similar to PSS. Additional inspections of all fairing splice regions will therefore be performed on all vehicles for the time being. Flight No. 9's fairing was confirmed free of delamination.

A broader review covered all new H3 materials and manufacturing technologies, and every subsystem and component, whether made by the prime contractor or partners, checking that process designs were supported by sufficient component tests and basic data. No additional issues requiring corrective action were identified.

Underlying factors and recurrence prevention

Analysis proceeded in two stages: first, technical factors in H3's development process that allowed the fault to remain undetected through launch; second, organization-wide factors in project management, knowledge management, risk management, personnel and capability.

Two development-process factors were identified:

12. Program objectives and manufacturing-equipment constraints led to adoption of a difficult new technology. Risks were identified, but tests of another configuration considered higher-risk were assumed to cover them, so the PSS fault was not identified.
13. Qualification tests showed the planned behavior, so neither post-test nondestructive inspection nor destructive testing was conducted. The fault remained undetected throughout development.

The organization-wide analysis also identified a need to improve risk-identification methods.

Future measures include a whole-program review of risks early in development and when specifications change, comprehensive identification of potential faults, and planning of necessary prototype tests. Developments with many new elements will involve more specialists with practical development experience, stronger independent reviews, and broader checks grounded in technical knowledge and experience. Post-qualification destructive tests and other inspections will be performed reliably to prevent faults first emerging in flight.

24 Flight results: No. 6 (H3-30 test flight) and No. 9

24.1 Flight No. 6 mission overview



H3 Flight No. 6, the H3-30 test vehicle.

- Launch: 12 June 2026, 09:53:59 JST.
- Payloads: VEP-5 and six secondary satellites—PETREL, STARS-X, BRO-22, VERTECS, HORN-L and HORN-R.
- Result: Success.
- Secondary payload details: https://fanfun.jaxa.jp/countdown/h3f6/files/sub_payload_presskit.pdf
- Mission details: https://fanfun.jaxa.jp/countdown/h3f6/files/h3f6_presskit.pdf

Flight No. 6 tested H3-30, which lifts off using three LE-9 engines without SRB-3. It verified three-engine liftoff, including hold-down operation, first-stage flight and vehicle control, completing the principal planned H3 lineup.

As the first launch after Flight No. 8's failure, it also incorporated checks and measures for PSS and the fairing and acquired added flight data to support the investigation and strengthen confidence in subsequent missions. A newly developed small-satellite adapter supported the secondary payloads.

24.1.1 Measures following Flight No. 8

Flight No. 6 used a repaired PSS after completing qualification tests and load testing of the flight unit.

All fairing splice areas received additional inspection. No delamination was found, so repair was unnecessary. Additional splice insulation increased margin against aerodynamic heating.

Added data supported the failure assessment and later missions, and may help evaluate a future return to adequately corrected splice bonding:

- PSS strain measurements along each axis on inner and outer surfaces, directly confirming flight loads under combined vacuum and static loading and comparing them with analysis.
- Improved data for factors that could not be fully resolved in Flight No. 8 FTA because of limited telemetry precision and coverage. Added sensors included high-frequency accelerometers for fairing-separation

shock; fairing/PSS pressure sensors for venting; and temperature sensors inside the fairing, PSS and interstage for signs of propellant or other leakage.

- Longer recording by the camera inside the fairing to confirm conditions at MECO and stage separation. Flight No. 8 recorded only an external camera at stage separation, so payload detachment was not directly observed.

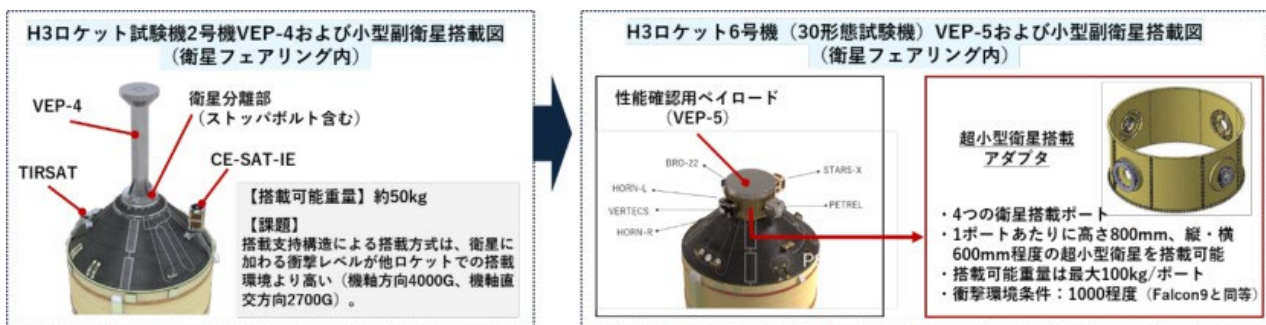
Some additional measurements will continue on later flights.

24.1.2 Small-satellite adapter: reduced shock environment

Test Flight No. 2's secondary satellites were mounted close to the fairing separation interface, exposing them to greater shock than typical mounting environments on other launch vehicles.

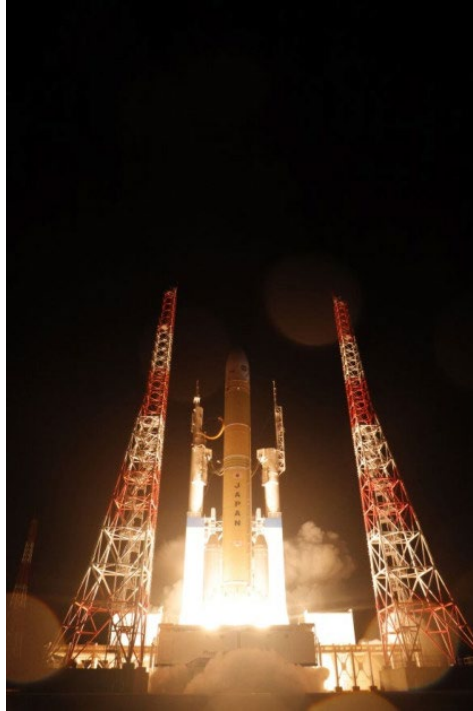
A new ring-shaped small-satellite adapter improves the shock environment and increases accommodation capacity to meet demand for multiple small-satellite launches and demonstrations.

On Flight No. 6, it was installed between PSS and PAF, with payloads occupying all four ports, providing flight demonstration.



Flight No. 6 payload arrangement and small-satellite adapter. English key: VEP-5; PAF; PSS; PETREL; STARS-X; BRO-22; VERTECS; HORN-L; HORN-R. Earlier TF2 mounting: approximately 50 kg payload capability; shock requirements 4,000 g along the vehicle axis and 2,700 g transverse. New ring adapter: four satellite ports; approximately 800 mm-high × 600 mm × 600 mm payload envelope per port; maximum 100 kg per port; shock environment approximately 1,000 g, comparable to Falcon 9. The ring is between PSS and PAF.

24.2 Flight No. 9 mission overview



H3 Flight No. 9.

- Launch: 11 August 2026, 04:23:31 JST.
- Payload: QZSS MICHIBIKI No. 7 (QZS-7).
- Result: Success.
- Mission details: https://fanfun.jaxa.jp/countdown/h3f9/files/h3f9_presskit_20260730.pdf

Flight No. 9 used H3-22, as did Flights Nos. 1–5 and 8, and performed first-stage throttling. The first operational use of the Redundant Integrated Navigation System (RINS) and the second autonomous flight-safety demonstration, following Flight No. 7, both produced satisfactory results.

24.2.1 Measures following Flight No. 8

Flight No. 9 used a fastener-joined PSS, eliminating the high-temperature risk of splice bonding and preventing the associated skin/core delamination. Fastener joining will be the baseline for important missions in the near term.

Additional inspection of all fairing splice regions found no delamination, so no repair was required. Added insulation increased aerodynamic-heating margin.

Following Flight No. 6, added flight data included strain measurements from the first fastener-joined PSS and fairing strain data to characterize events. Results were satisfactory.

25 Flight No. 10

25.1 Mission overview

Flight No. 10 will place the Martian Moons eXploration (MMX) spacecraft onto a Mars transfer trajectory. This is H3's first planetary exploration launch and the first flight of the H3-24L configuration.

It is the second launch of H3-24, following Flight No. 7, and the first use of the long fairing. Although the long fairing uses the same technology as the short fairing, its more demanding loads required dedicated qualification tests, including strength and separation/jettison tests.

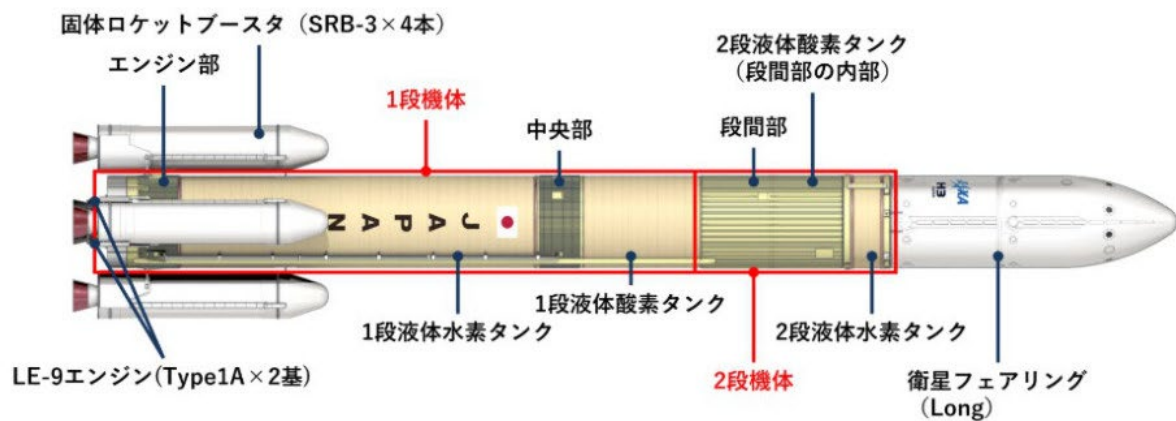
Fairing comparison

Item	Short	Long	Wide
Outer diameter	5.2 m	5.2 m	5.4 m
Length	10.4 m	16.4 m	16.5 m
Encapsulation method	Complete fairing lowered over payload from above	Complete fairing lowered over payload from above	Two half-shells joined from either side
Sinks after splashdown	Yes	Yes	No; recovered
Separation mechanism	Clamshell opening	Clamshell opening	Parallel opening

Flight No. 10 also makes the first operational use of TDRS telemetry relay, demonstrated on Flight No. 7 (Section 22.4.2). Ground stations cannot see the vehicle from the start of the second second-stage burn through MMX separation, so TDRS relay will support monitoring during that period.

25.2 Vehicle configuration

H3-24L: two LE-9 engines, four SRB-3 boosters and a long fairing.



Flight No. 10 H3-24L layout: long fairing; MMX and payload attachment structure; second-stage hydrogen/oxygen tanks and LE-5B-3; interstage; first-stage hydrogen/oxygen tanks; four SRB-3 boosters; two LE-9 engines.

25.3 Principal specifications

Item	Specification
Vehicle	H3 Flight No. 10
Overall length	Approximately 63 m
Fully loaded mass	Approximately 575 t, excluding spacecraft mass
Guidance	Inertial guidance

Item	First stage (LE-9)	SRB-3, four boosters	Second stage (LE-5B-3)	Long fairing
Length, m	Approx. 37	Approx. 15	Approx. 12	Approx. 16.4
Outer diameter, m	Approx. 5.2	Approx. 2.5	Approx. 5.2	Approx. 5.2
Mass, t	Approx. 240	Approx. 304.8, combined	Approx. 28	Approx. 2.7
Maximum propellant mass, t	224.5	268.8, combined	24.6	—
Vacuum thrust, kN*	Approx. 2,942, two engines	Approx. 9,200, four boosters	Approx. 137	—
Burn time, s	Approx. 300	Approx. 110	Approx. 694	—
Propellants	LH2/LOX	Composite solid propellant	LH2/LOX	—
Feed system	Turbopumps	—	Turbopumps	—
Attitude control	Engine gimbaling	—	Engine gimbaling and gas jets	—
Principal electronics	Guidance/control equipment	—	Guidance/control equipment; radio-navigation equipment; telemetry transmitter; flight termination equipment	—

*Solid motor thrust is specified at maximum thrust.

25.4 Flight sequence

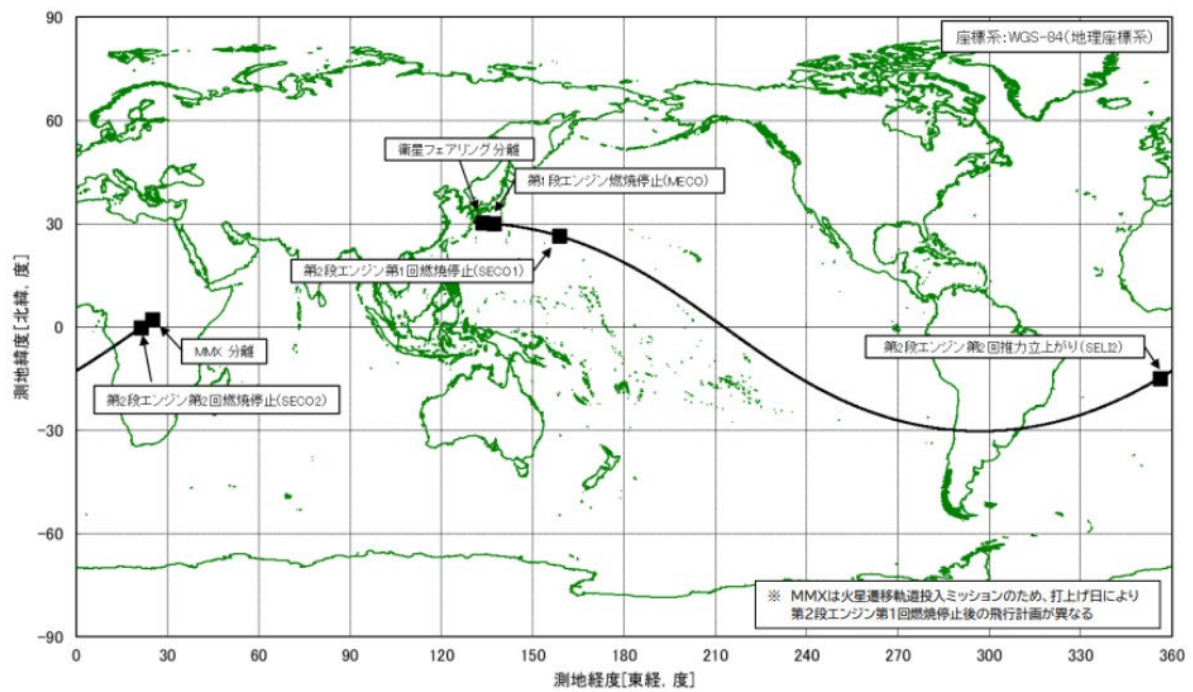
Flight No. 10 will launch MMX from Launch Pad 2 at Tanegashima. Soon after liftoff, the vehicle will orient its pitch plane to azimuth 90° and follow the planned trajectory over the Pacific.

The four SRB-3 boosters will separate in two pairs, at approximately T+1 minute 56 seconds and T+1 minute 58 seconds. The fairing will separate at approximately T+3 minutes 17 seconds. First-stage cutoff will occur at approximately T+4 minutes 52 seconds, followed by stage separation at T+4 minutes 59 seconds.

The second-stage engine's first burn will begin at approximately T+5 minutes 12 seconds and end at T+10 minutes 40 seconds. Following a coast, its second burn will begin at approximately T+1 hour 3 minutes 57 seconds and end at T+1 hour 10 minutes 21 seconds. MMX will separate onto the designated Mars transfer trajectory at approximately T+1 hour 11 minutes 12 seconds.

Event	Approximate elapsed time
First pair of SRB-3 boosters separates	00:01:56
Second pair of SRB-3 boosters separates	00:01:58
Fairing separation	00:03:17
First-stage main-engine cutoff	00:04:52
Stage separation	00:04:59
Second-stage first burn begins	00:05:12
Second-stage first burn ends	00:10:40
Second-stage second burn begins	01:03:57
Second-stage second burn ends	01:10:21
MMX separation	01:11:12

Because MMX is a Mars-transfer mission, the flight plan after the first second-stage cutoff varies with launch date. The sequence above is for a launch on 20 October 2026 JST.



Flight path from liftoff to MMX separation for the 20 October 2026 JST launch case. English key: liftoff; booster and fairing separation; MECO; first/second-stage separation; first second-stage burn; parking-orbit coast; second second-stage burn; MMX separation onto a Mars transfer trajectory.

26 Future outlook

H3 has successfully flown H3-22S on Flights Nos. 2–5 and 9, H3-24 on Flight No. 7, and demonstrated H3-30 on Flight No. 6.

Planned missions include government spacecraft such as Information Gathering Satellites and QZSS satellites, lunar and planetary explorers including the Lunar Polar Exploration mission and MMX on Flight No. 10, and scientific satellites.

Even after development is complete and operational service begins, manufacturing, assembly, testing and launch processes will continue to be refined.

The program intends to transition to private-sector-led launches by MHI and participate in commercial launches for domestic and overseas customers. In 2018, MHI secured an H3 launch order from UK mobile satellite communications provider Inmarsat. In 2026, MHI also signed a launch-services contract for ispace's new ULTRA lander and rideshare launch contracts with multiple Japanese and overseas companies.

The commercial market has changed considerably since H3 development began. Falcon 9's low prices have intensified competition. ULA's Vulcan, Blue Origin's New Glenn and Europe's Ariane 6 entered the market around the same period as H3, increasing price pressure.

Demand has also diversified beyond individual large geostationary satellites to multiple small-satellite constellations and deep-space missions to the Moon and Mars. Reuse and larger launch vehicles are expected to drive further price and market changes.

H3 must continue to improve and evolve to respond to these conditions while sustaining Japan's mainstay launch capability. This will preserve independent access to space and support further growth in space utilization in Japan and worldwide.

Appendix 1 Evolution of Japanese launch vehicles

Solid-propellant rockets: beginning with Pencil

Japan's solid-propellant rocket development began with the Pencil Rocket, led by the late Hideo Itokawa at the University of Tokyo's Institute of Industrial Science.

Only 1.8 cm in diameter and 23 cm long, Pencil was extremely small. Japan did not yet have radar rocket-tracking technology, so experiments fired it horizontally. A public firing on 12 April 1955 marked the beginning of Japanese solid-rocket development and the country's pursuit of spaceflight.

Technology advanced progressively until rockets reached space. In 1970, Lambda-4S launched Japan's first satellite, OHSUMI.

The larger Mu (M) series then supported major advances in Japanese space science from the 1970s through the 2000s, including X-ray astronomy, space plasma physics and solar physics. The final Mu vehicle, M-V, achieved world-leading solid-rocket payload and control capability and earned international recognition.



Hideo Itokawa and the Pencil Rocket.



Lambda-4S launches OHSUMI, Japan's first satellite.

Liquid-propellant launch vehicles: technology introduced from the United States

Alongside solid-rocket development, practical space applications such as satellite broadcasting and weather observation expanded in the 1960s. Japan prepared to enter these fields, establishing the National Space Development Agency of Japan (NASDA) in 1969.

Many practical satellites required geostationary orbit, creating a need for liquid-propellant vehicles with greater capacity and controllability than solid rockets. Although domestic development was considered, limited technology and the need for early capability led Japan to introduce US technology.

Japan developed and operated N-I, followed by N-II and H-I. While learning from the United States, it also researched its own technologies and gradually introduced them, including second-stage engines.



Maiden launch of N-I.

Toward independent Japanese launch vehicles

In 1994, Japan successfully developed H-II, its first fully indigenous liquid-propellant launch vehicle. All elements, including the first-stage main engine previously dependent on US technology, were domestically produced. Combining them with Japan's solid-rocket expertise dating back to Pencil gave Japan an independent large launch vehicle.

H-IIA followed, improving reliability and reducing cost. H-IIB increased capacity to launch HTV cargo spacecraft. Epsilon inherited Japan's solid-rocket technology while pursuing lower costs.

H3 carries forward this liquid- and solid-rocket heritage, tradition and independence, while aiming to provide a more user-friendly launch vehicle.



H-II, Japan's first fully indigenous large launch vehicle.

Appendix 2 Synergy with Epsilon

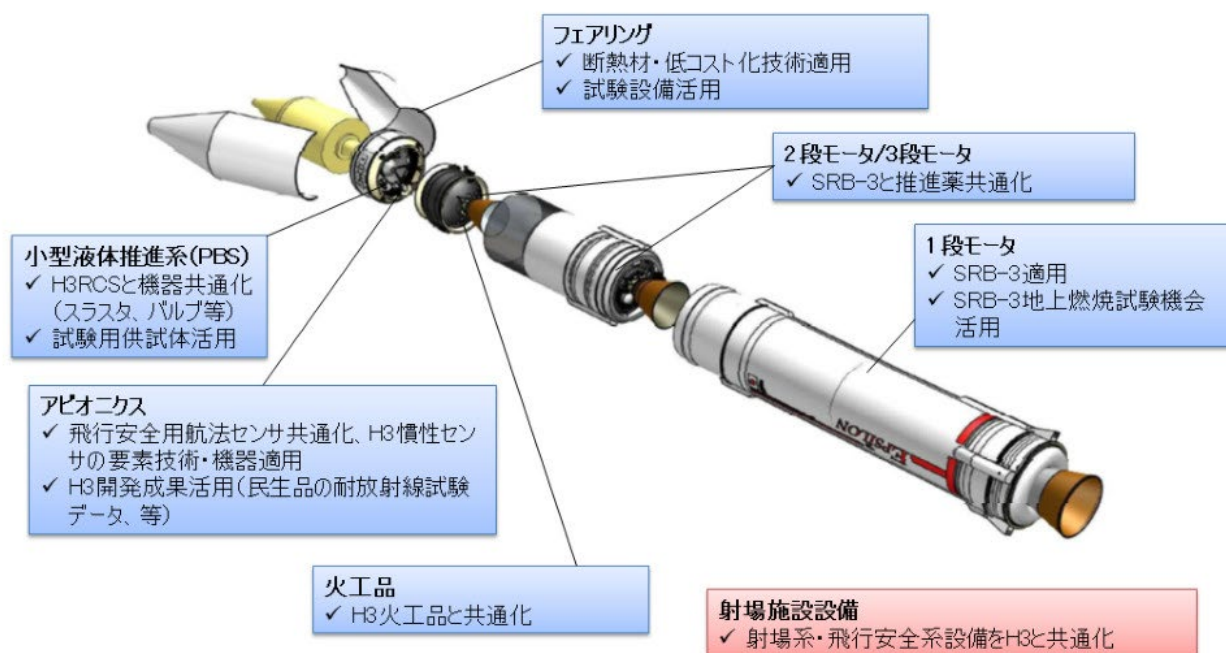
A major feature of H3 development is synergy with Epsilon S, Japan's small solid-propellant launch vehicle.

Epsilon's first test flight succeeded in 2013. An enhanced version with greater payload capability entered service on Flight No. 2 in 2016. Six Epsilon vehicles have been launched. Epsilon S is under development for further capacity gains and lower costs.

Epsilon S maximizes common technology and components with H3. For example, SRB-3 and the Epsilon S first-stage motor share motor-case, propellant and thrust-profile designs as far as possible. SRB-3 has a fixed nozzle, while Epsilon S adds thrust vector control (TVC).

Fairings, second- and third-stage motors, the Post Boost Stage (PBS) small liquid propulsion system, avionics, pyrotechnics and launch facilities also use shared components or H3 technologies.

Static firings of the Epsilon S second-stage motor, E-21, were conducted at Noshiro Rocket Testing Center on 14 July 2023 and Tanegashima Space Center on 26 November 2024. Both experienced abnormal combustion and motor explosions. An investigation team headed by the directorate's director was established for the second static-firing anomaly and continues to investigate causes and corrective measures.



H3/Epsilon S commonality. Fairing: insulation/low-cost technology and test facilities. First-stage motor: SRB-3 and its ground-firing test opportunities. Second-/third-stage motors: propellant commonality with SRB-3. PBS: components shared with H3 RCS, including thrusters and valves, and test-fluid commonality. Avionics: common navigation sensors for flight safety; H3 inertial-sensor technologies/equipment; H3 development results such as radiation-test data for commercial parts. Pyrotechnics: shared H3 devices. Launch facilities: shared range and flight-safety equipment.